NASA SP-45

MERCURY PROJECT SUMMARY

INCLUDING RESULTS OF THE

FOURTH MANNED ORBITAL FLIGHT

MAY 15 AND 16, 1963

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Manned Spacecraft Center

PROJECT MERCURY

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NATIONAL AERONAUTICS
AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
PROJECT MERCURY

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ERRATA—NASA REPORT SP–45

Page 129—For the first five lines in left-hand column, substitute the following:

The United Kingdom also assisted in the selection of communication sites in Africa and in the South Pacific and continued to provide support in the operation and maintenance of certain communication facilities.

The Republic of Nigeria provided land near the city of Kano, assisted in the construction of the station and ground communication facilities, and provided continued support during the operational phase.

The Republic of Zanzibar provided land and assisted in the establishment of the station and ground communication facilities.

The Government of Spain provided land on Grand Canary Island and established the Instituto Nacional de Técnica Aeronáutica (INTA) as the Spanish agency to participate in the implementation and operation of this facility.

The Government of Mexico provided land and participated in the implementation of the station near Guaymas, Mexico. In a joint effort, the Mexico-United States Commission for Space Tracking Observation was formed to provide coordination for the construction and operation of this station.

In establishing stations as a joint effort with the various participating countries, every effort was made to make maximum possible use of local resources and people, to permit free access to the sites, and to establish a basis for continued cooperation throughout the program.

Pages 369–381—The following names were inadvertently omitted from Appendix E.

Aikenhead, Bruce A.
Armstrong, Neil
Baker, Charles
Baker, Thomas F.
Bathurst, Raymond
Bavely, James C.
Bayer, Philip J.
Berry, Lt. Col. S. L.
Brackett, Ernest W.
Breneman, G. Jean
Brockett, H. R.
Buckley, Edmond C.
Chiabotta, R. L.
Clark, Lt. Col. Raymond L.
Conger, Dean
Connor, Lt. Col. Joseph
Cox, Hiden T.
Decherd, Mary C.
Dembling, Paul G.
Dick, Louise
Donlan, Charles J.
Draim, Lt. Cmdr. John E.
Dunkleman, Dr. Lawrence
Evans, Peggy R.
Ferrando, James A.
Ferrando, Rita P.
Frutkin, Arnold W.
Gaever, A. Les
Gamble, Dr. Allen O.
Gesell, Ralph T.
Giannini, William F.
Gibbs, Col. Asa B.
Gill, Dr. Jocelyn R.
Glahn, Earl W.
Glennan, Dr. T. Keith
Goett, Dr. Harry J.
Hand, Ben R.
Haney, Paul
Hare, Linda J.
Harlan, Charles S.
Hartley, Robert M.
Hayes, Capt. Richard
Heaton, Lt. Col. Donald J.
Hemperly, John O.
Henry, Maj. Richard C.
Holmes, D. B.
Holmes, Jay
Horner, Richard E.
Hunter, Willson H.
Hyatt, Abraham
Jenkins, Thomas E.
Johnson, Clinton T.
Johnson, Harold G.
Johnson, Jennie M.
Keany, J.
Kehlet, Alan B.
Kier, Cmdr. Kenneth
Krieger, Robert L.
Kuettner, Dr. Joachim P.
Land, Vinton T.
Leech, Ted B.
Love, James T.
McCaskill, Patsy L.
McGee, Bernard J.
McTigue, John
Neal, James L.
Ostrander, Maj. Gen. Donald R.
Overton, John D.
Petersen, Cdr. Forrest N.
Porter, Thomas J.
Powers, Col. John A.
Rabb, L.
Reed, Carol C.
Rice, Charles N.
Ricker, Harry H., Jr.
Roadman, Brig. Gen. Charles
Salmassy, Omar K.
Sanders, Newell D.
Schmidt, Richard A.
Seamans, Dr. Robert C.
Silverstein, Abe
Soulé, Hartley A.
Stack, John
Stelter, Laverne R.
Stockwell, Ephriam
Thompson, Carolyn B.
Thompson, Floyd L.
Truszynski, G. M.
Turner, Lt. Col. W. R.
von Braun, Wernher
Walker, Joseph
Wallace, Robert F.
White, Dr. Stanley C.
Williams, J. J.
Wilson, Charles
Wolhart, Walter O.
Wood, Clotaire
Wyatt, D. D.
Young, Col. Robert P.
Zimmerman, C. H.

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FOREWORD

This document presents a summary of the planning, preparation, experiences, and results of Project Mercury and includes the results of the fourth United States manned orbital flight conducted on May 15 and 16, 1963, are also included. The papers are grouped into four main technical areas: The space-vehicle development, mission support development, flight operations, and mission results. The performance discussions contained in the various papers for the concluding Mercury mission form a continuation of the information previously published for the three manned orbital flights and the two manned suborbital flights. Although this document, to a limited degree, summarizes the results of the previous manned flights, the formal postflight reports published for these earlier missions should be consulted for greater detail.

The material presented in this document has been prepared in a short period of time. It reflects the close cooperation and intense efforts of the authors, the staff editors, and the printers, all of whom are to be commended for their dedicated efforts.

KENNETH S. KLEINKNECHT,
Manager, Mercury Project.

W. M. BLAND, JR.,
Deputy Manager, Mercury Project.

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1. PROJECT REVIEW

By WALTER C. WILLIAMS, *Deputy Director for Mission Requirements and Flight Operations, NASA Manned Spacecraft Center;* KENNETH S. KLEINKNECHT, *Manager, Mercury Project, NASA Manned Spacecraft Center;* WILLIAM M. BLAND, JR., *Deputy Manager, Mercury Project, NASA Manned Spacecraft Center; and* JAMES E. BOST, *Chief, Engineering Operations Office, Mercury Project Office, NASA Manned Spacecraft Center*

Summary

The United States' first manned space flight project was successfully accomplished in a 4⅔ year period of dynamic activity which saw more than 2,000,000 people from many major government agencies and much of the aerospace industry combine their skills, initiative, and experience into a national effort. In this period, six manned space flights were accomplished as part of a 25-flight program. These manned space flights were accomplished with complete pilot safety and without change to the basic Mercury concepts. It was shown that man can function ably as a pilot-engineer-experimenter without undesirable reactions or deteriorations of normal body functions for periods up to 34 hours of weightless flight.

Directing this large and fast moving project required the development of a management structure and operating mode that satisfied the requirement to mold the many different entities into a workable structure. The management methods and techniques so developed are discussed. Other facets of the Mercury experience such as techniques and philosophies developed to insure well-trained flight and ground crews and correctly prepared space vehicles are discussed. Also, those technical areas of general application to aerospace activities that presented obstacles to the accomplishment of the project are briefly discussed. Emphasis is placed on the need for improved detail design guidelines and philosophy, complete and appropriate hardware qualification programs, more rigorous standards, accurate and detailed test procedures, and more responsive configuration control techniques.

Introduction

The actual beginning of the effort that resulted in manned space flight cannot be pinpointed although it is known that the thought has been in the mind of man throughout recorded history. It was only in the last decade, however, that technology had developed to the point where man could actually transform his ideas into hardware to achieve space flight. Specific studies and tests conducted by government and industry culminating in 1958 indicated the feasibility of manned space flight. Implementation was initiated to establish a national manned space-flight project, later named Project Mercury, on October 7, 1958.

The life of Project Mercury was about 4⅔ years, from the time of its official go-ahead to the completion of the 34-hour orbital mission of Astronaut Cooper. During this period, much has been learned about man's capabilities in the space environment and his capabilities in earthbound activities which enabled the successful accomplishment of the objectives of the Mercury Project in this relatively short period. It is the purpose of this paper to review the more significant facets of the project beginning with the objectives of the project and the guidelines which were established to govern the activity. As in any form of human endeavor, there are certain signs which serve as the outward indication of activity and progress. For the Mercury Project, these signs were the major full-scale flight tests. These tests will be reviewed with particular emphasis on schedule, the individual mission objectives, and the results from each mission. Then, the organization with which management directed the activities of Project Mercury will be explained, particularly with respect to those internal interfaces between major segments of NASA and those external interfaces with contractors and other governmental departments. The resources expended during the project will be explained with discussions on manpower and cost. In addition, the major results of the project will be discussed as will those areas which presented severe obstacles to technical progress.

This paper is primarily a review; greater detail in many of the areas discussed can be obtained by reference to other papers in this document and to the documents listed in the bibliography.

Objectives and Guidelines

The objectives of the Mercury Project, as stated at the time of project go-ahead, were as follows:

(1) Place a manned spacecraft in orbital flight around the earth.

(2) Investigate man's performance capabilities and his ability to function in the environment of space.

(3) Recover the man and the spacecraft safely.

After the objectives were established for the project, a number of guidelines were established to insure that the most expedient and safest approach for attainment of the objectives was followed. The basic guidelines that were established are as follows:

(1) Existing technology and off-the-shelf equipment should be used wherever practical.

(2) The simplest and most reliable approach to system design would be followed.

(3) An existing launch vehicle would be employed to place the spacecraft into orbit.

(4) A progressive and logical test program would be conducted.

More detailed requirements for the spacecraft were established as follows:

(1) The spacecraft must be fitted with a reliable launch-escape system to separate the spacecraft and its crew from the launch vehicle in case of impending failure.

(2) The pilot must be given the capability of manually controlling spacecraft attitude.

(3) The spacecraft must carry a retrorocket system capable of reliably providing the necessary impulse to bring the spacecraft out of orbit.

(4) A zero-lift body utilizing drag braking would be used for reentry.

(5) The spacecraft design must satisfy the requirements for a water landing.

It is obvious by a casual look at the spacecraft (fig. 1-1) that requirements (1), (3), and (4) were followed as evidenced by the escape tower, the retrorocket system that can be seen on the blunt end of the spacecraft, and the simple blunt-body shape without wings. Items (2) and (5) have been made apparent by the manner in which the astronaut has manually controlled the attitude of the spacecraft during orbital maneuvers, retrofire, and reentry, and by the recovery of the spacecraft and astronauts after each flight by recovery forces which included aircraft carriers and destroyers.

Basically, the equipment used in the spacecraft was derived from off-the-shelf equipment or through the direct application of existing technology, although some notable exceptions were made in order to improve reliability and flight safety. These exceptions include:

(1) An automatic blood-pressure measuring system for use in flight.

(2) Instruments for sensing the partial pressures of oxygen and carbon dioxide in the oxygen atmosphere of the cabin and suit, respectively.

Some may argue with the detailed way in which the second basic guideline of simplicity was carried out; however, this guideline was carried out to the extent possible within the volume, weight, and redundancy requirements imposed upon the overall system. The effect of the weight and volume constraints, of course, resulted in smaller and lighter equipment that could not always be packaged in an optimum way for simplicity.

Redundancy probably increased the complexity of the systems more than any other requirement. Because the spacecraft had to be qualified by space flight first without a man onboard and then because the reactions of man and his capabilities in the space environment were unknown, provisions for a completely automatic operation of the critical spacecraft functions were provided. To insure reliable operation, these automatic systems were backed up by redundant automatic systems.

The third guideline was satisfied by an adaptation of an existing missile, the Atlas. The modifications to this launch vehicle for the use in the Mercury Project included the addition of a means to sense automatically impending catastrophic failure of the launch vehicle and provisions to accommodate a new structure that would form the transition between the upper section of the launch vehicle and the spacecraft. Also, the pilot-safety program was initiated to insure the selection of quality components.

Application of the fourth guideline is illustrated by the major flight schedule which is discussed in the next section.

Major Flight Schedules

Planned Flight Test Schedule

The Mercury flight schedule that was planned early in 1959 is shown in figure 1–2. These are the major flight tests and include all those scheduled flight tests that involved rocket-propelled full-scale spacecraft, including boilerplate and production types. The planned flight test program shows 27 major launchings. There are three primary types of tests included in these, one type being the research-and-development tests, another being primarily flight-qualification of the production spacecraft, and the third being the manned orbital flight tests. In addition, the tests with the Mercury-Redstone launch vehicle provided some early ballistic flights for pilot training. Involved in the planned flight-test program were four basic types of launch vehicles, the Little Joe, the Mercury-Redstone, the Mercury-Jupiter, and the Mercury-Atlas.

Four Little Joe flights and two of the Atlas powered flights, termed Big Joe, were planned to be in the research and development category to check the validity of the basic Mercury concepts.

The qualification program was planned to use each of the four different launch vehicles. The operational concept of the qualification program provided for a progressive build-up of flight-test system complexity and flight-test conditions. It was planned that the operation of all

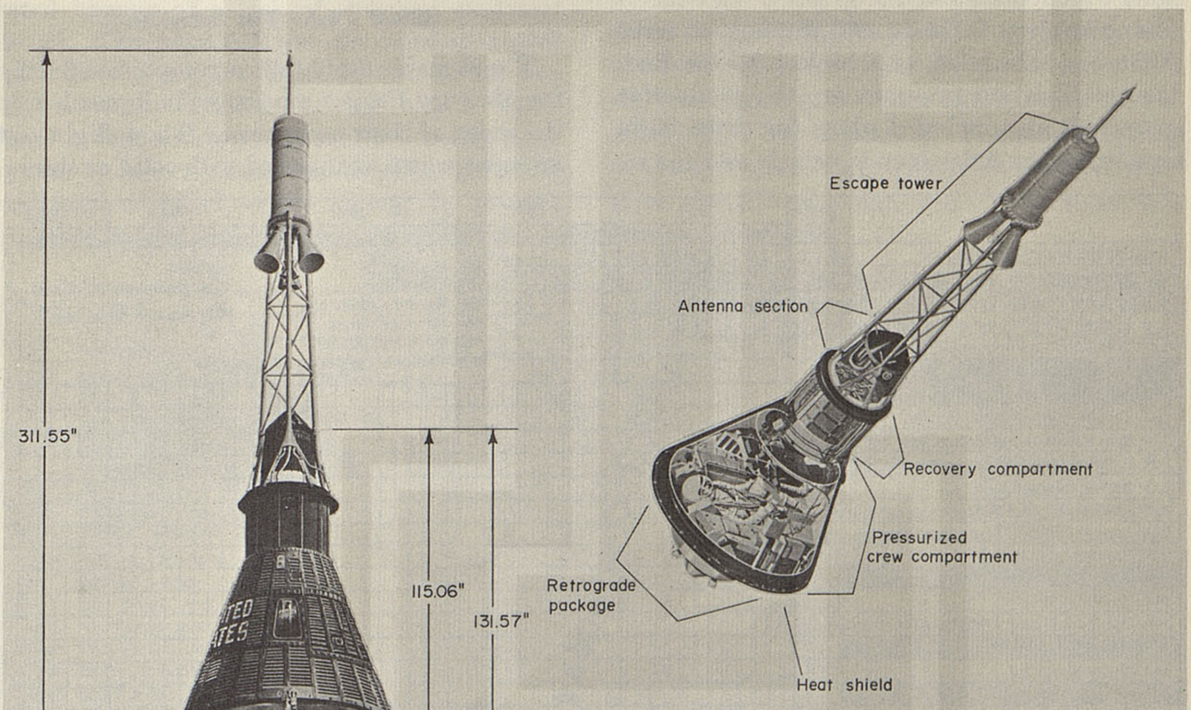


FIGURE 1–1.—General view of spacecraft.

hardware items would be proven in those environments to which they would be subject in both normal and emergency conditions associated with attainment of the planned mission conditions. One qualification flight test was planned with the use of the Little Joe launch vehicle. This test was planned to qualify the operation of the production spacecraft in a spacecraft-abort situation at the combination of dynamic pressure, Mach number, altitude, and flight-path angle that represented the most severe condition anticipated for the use of this system during an orbital launch. There were eight flight tests planned with the use of the Redstone launch vehicle. The first two were intended to be unmanned tests used to qualify the production spacecraft and to qualify the production-spacecraft launch-vehicle combination. The remaining six Mercury-Redstone flights were to be used to train and qualify Mercury astronauts for later orbital flights. Two flight tests were planned in which the Jupiter launch vehicle was to be used. The first one of these was to be made to qualify the production spacecraft for those flight conditions which produced the greatest load factor during reentry. The second Jupiter powered flight was scheduled as a backup to the first. The qualification program for the production spacecraft also included plans for three flight tests using the Atlas launch vehicle and the remainder of the flights were expected to be used for manned orbital flight if the flight qualification achieved at the time so warranted.

This flight-test plan was developed and proposed in early 1959 as a test plan that represented a completely trouble-free preparation and flight-test program. According to this schedule, the first manned orbital flight could have occurred as early as April of 1960. This flight-test schedule represents planning that was done before experience was gained in the production of spacecraft flight hardware and, particularly important, before any experience had been gained in the preparation of space flight equipment for manned flight.

The planned flight test schedule (fig. 1–2) presents some missions that are shaded. This shading indicates that these particular missions were eliminated during the course of the program because the requirement either was not necessary or was satisfied by some other means. In addition, it should be noted that the objectives of some of the other missions were altered to fit the situation as the project advanced.

Actual Flight Test Schedules

The 25 major flight tests accomplished during the Mercury Project are shown in figure 1–3, in the order of their occurrence. Those flight-test missions which are marked with solid circles in-

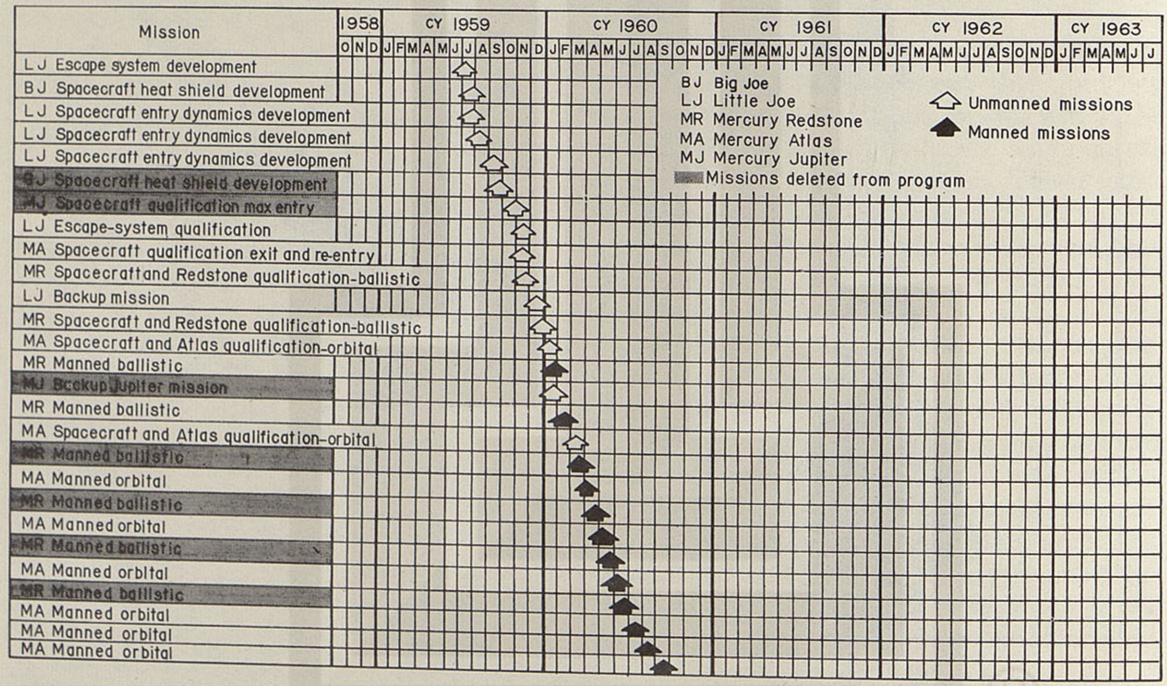


Figure 1–2.—Planned flight schedule as of January 1959.

dicate the missions that were added to this schedule as a result of lessons learned during some of the preceding flight tests or because of extensions to the basic mission objectives as in the case of the last two missions, MA–8 and MA–9.

Little Joe 1.—The flight test program was initiated with the Little Joe 1 research-and-development mission that was scheduled for July of 1959. The actual launch attempt came in the following month, on August 21, at the NASA launch site, Wallops Station, Va. A nearly catastrophic failure occurred at a time late in the launch countdown as the vehicle battery-power supply was being charged. At this time, the escape-rocket sequence was unintentionally initiated and the spacecraft was separated from the launch vehicle and propelled into the air as in a pad-abort sequence. The escape sequence was accomplished correctly, though initiated by a fault. The tower was jettisoned properly, the drogue parachute was deployed as it should have been, but the main parachute deployment circuitry was not activated because of a lack of sufficient electrical power. The spacecraft was destroyed on impact with the water. The cause of the failure was determined by detailed analyses to be a "backdoor" circuit which permitted the launch-escape system to be activated when a given potential had been supplied to the battery by ground charging equipment. The launch vehicle, though fully loaded with six solid-propellant rocket motors, was left undamaged on the launcher.

Big Joe 1.—Spacecraft checkout for the launch of Big Joe 1 was accomplished at the Cape Canaveral launch site starting in June of 1959. The primary purpose of the flight was to investigate the performance of the ablation heat shield during reentry, as well as to investigate spacecraft reentry dynamics with an instrumented boilerplate spacecraft. Other items that were planned for investigation on this flight were afterbody heating for both the exit and reentry phases of flight, drogue and main parachute deployment, dynamics of the spacecraft system with an automatic control system in operation, flight loads, and water-landing loads. Recovery aids, such as SOFAR bombs, radio beacons, flashing light, and dye markers, had been incorporated. This spacecraft was not equipped with an escape system. The mission was accomplished on September 9, 1959. Because of the failure of the Atlas booster engines to separate, the planned trajectory was not followed exactly, but the conditions which were achieved provided a satisfactory fulfillment of the test objectives. The landing point of the spacecraft was about 1,300 nautical miles from the lift-off point, which was about 500

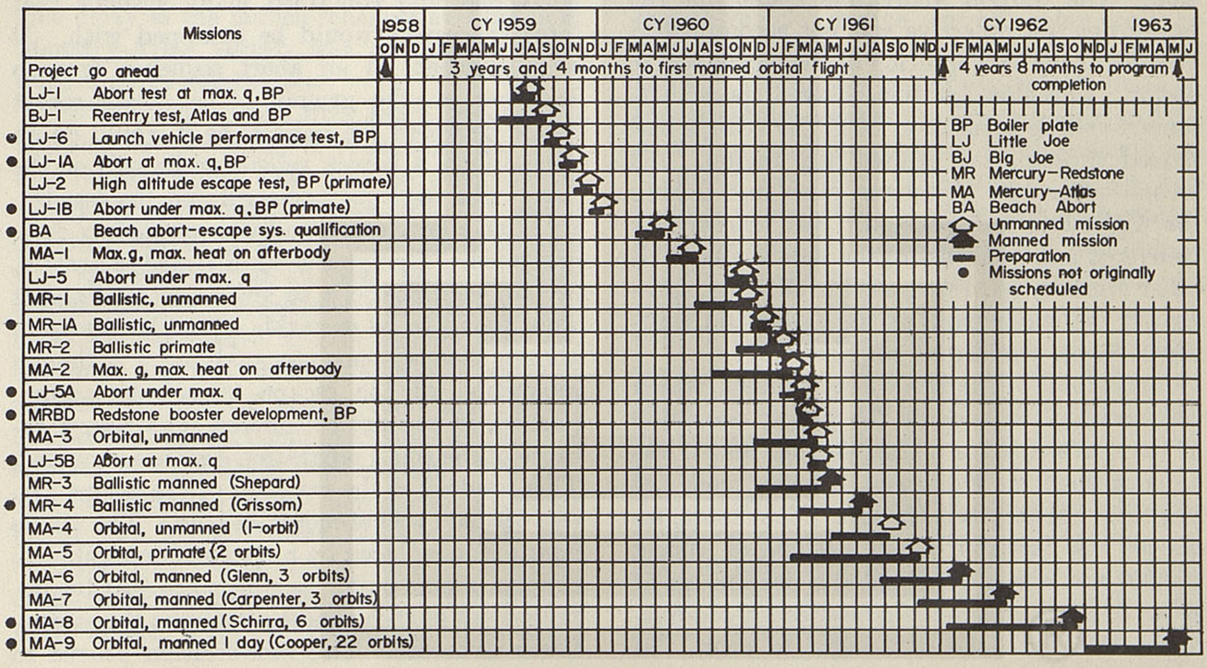


FIGURE 1–3.—Actual flight schedule.

nautical miles short of the intended landing point. Even so, the recovery team retrieved the spacecraft about 7 hours after landing.

Data from instrumentation and results of inspection of the spacecraft showed that the heat-protection method planned for the production spacecraft was satisfactory for a normal reentry from the planned orbit. On the basis of these results, the backup Big Joe mission was cancelled.

Little Joe 6.—The Little Joe 6 mission was successfully accomplished on October 4, 1959, from the Wallops Station launch site and demonstrated a qualification of the launch vehicle by successfully flying with staged propulsion on a trajectory which gave structural and aerodynamic loads in excess of those expected to be encountered on the other planned Little Joe missions. In addition, a method devised for correcting the launcher settings for wind effects, the performance of the booster command thrust termination system, and the launch operation were checked out satisfactorily. Two minor modifications were made to the Little Joe vehicle as a result of this flight to protect the second-stage rocket motor and the launch vehicle base from heat radiated from the thrusting motors.

Little Joe 1A.—Little Joe 1A was launched on November 4, 1959, from the Wallops Station launch site, as a repeat of the Little Joe 1 mission. The inflight abort was made, but the first-order test objective was not accomplished because of the slow ignition of the escape rocket motor. This slow ignition delayed spacecraft-launch-vehicle separation until the vehicle had passed through the desired test region. All second-order test objectives were met during the flight and the spacecraft was successfully recovered and returned to the launch site. All other Mercury hardware used in this test, principally the major parts of the escape and landing systems, performed satisfactorily.

Little Joe 2.—The Little Joe 2 mission, which was intended to validate the proper operation of the spacecraft for a high altitude abort, was accomplished on December 4, 1959, from the Wallops Station launch site. The abort sequence was initiated at an altitude of almost 100,000 feet and approximated a possible set of abort conditions that could be encountered during a Mercury-Atlas exit flight to orbit. In addition to the first-order objectives, the spacecraft reentry dynamics behavior without a control system was found to be satisfactory. The spacecraft dynamic stability on descent through the atmosphere was found to be as expected. Additional information was obtained on the operation of the Mercury parachute, the Mercury spacecraft flotation characteristics, and the operational requirements of spacecraft recovery by surface vessels. A monkey was a passenger on this mission; both the monkey and the spacecraft were recovered in satisfactory condition at the end of the mission.

Little Joe 1B.—The Little Joe 1B mission was successfully accomplished on January 21, 1960, from the Wallops Station launch site. This mission had been added to the flight schedule because of the failures of Little Joe 1 and Little Joe 1A to meet the test objectives. On this mission, all test objectives were successfully met, with the accomplishment of an abort at the conditions described for Little Joe 1A. This spacecraft also had a monkey as a passenger. Both the monkey and the spacecraft were recovered satisfactorily at the end of the mission.

Beach Abort 1.—Mission Beach Abort 1 (BA–1) was accomplished on May 9, 1960, from the Wallops Station launch site and marked the first time that a production spacecraft underwent a major qualification flight test. Production spacecraft 1 was a reasonably complete spacecraft and contained many systems that later spacecraft would be equipped with. It was launched on an abort sequence from a launcher on the ground. The escape-rocket motor provided the impulse as it would on an escape from a launch vehicle while still on the pad. The test was successful and the feasibility of an abort from a pad was adequately demonstrated. Though the mission was successful, certain modifications to spacecraft equipment were found to be desirable after the performance of these systems was analyzed. Although separation of the escape tower was accomplished, it was not considered satisfactory because of the small separation distance provided. This resulted in the redesign of the escape-system jettison rocket-motor nozzles. The single nozzle was replaced by a tri-nozzle assembly to prevent rocket-motor performance loss by impingement of the exhaust plumes on the escape-tower structure. This modification proved to be satisfactory and was retained for the remainder of the Mercury program. Another anomaly was the poor performance of the spacecraft telemetry transmitters. Investigation showed that the cause of this poor performance was a reversal of the cabling of the transmitter systems; thus, for the first time in the program, inadvertent cross connection of connectors had been deleted.

Mercury-Atlas 1.—The Mercury-Atlas 1 (MA–1) vehicle was launched from the Cape Canaveral test site on July 29, 1960. The primary purpose of the MA–1 flight was to test the structural integrity of a production Mercury spacecraft and its heat-protection elements during reentry from an exit abort condition that would provide the maximum heating rate on the afterbody of the spacecraft. The spacecraft involved was production item 4 and was equipped with only those systems which were necessary for the mission. An escape system was not provided for this spacecraft. The mission failed about 60 seconds after lift-off. The spacecraft and launch vehicle impacted in the water east of the launch complex. Because of this failure, an intensive investigation into the probable causes was undertaken. As a result of this investigation modifications were made to the interface area between the launch vehicle and the spacecraft to increase the structural stiffness. This inflight failure and subsequent intensive investigation resulted in a considerable delay in the launch schedule and the next Mercury-Atlas launch was not accomplished until almost 7 months later.

Little Joe 5.—The Little Joe 5 vehicle was launched on November 8, 1960, from the Wallops Station launch site. The test was intended to qualify a production spacecraft. It was a complete specification spacecraft at that time with the following exceptions: the landing-bag system was not incorporated; the attitude stabilization and control system was not fully operational, but was installed and used water to simulate the control system fuel; and certain components of the communications system not essential to the mission were omitted. The mission failed during flight when the escape-rocket motor was ignited before the spacecraft was released from the launch vehicle. The spacecraft remained attached to the launch vehicle until impact and was destroyed. The exact cause of the failure could not be determined because of the condition of the spacecraft components when recovered from the ocean floor and because of the lack of detailed flight measurements. The results of the analyses attributed the failure to components of the sequential system, but the cause could not be isolated. The sequential systems of spacecraft 2 and 6 were modified to preclude the possibility of a single erroneous signal igniting the escape-rocket motor.

Mercury-Redstone 1 and 1A.—The Mercury-Redstone 1 (MR–1), which was to provide qualification of a nearly complete production spacecraft number 2, in flight with a Mercury-Redstone launch vehicle, was attempted on November 21, 1960, at the Cape Canaveral launch site. The mission was not successful. At lift-off, the launch-vehicle engine was shut down and the launch vehicle settled back on the launcher after vertical motion of only a few inches. The spacecraft also received the shutdown signal and its systems reacted accordingly. The escape-rocket system was jettisoned and the entire spacecraft landing system operated as it had been designed. Analyses of the cause of malfunction showed the problem to have been caused by failure of two ground umbilicals to separate from the launch vehicle in the proper sequence. In the wrong sequence, one umbilical provided an electrical path from launch-vehicle power through blockhouse ground and the launch-vehicle engine cut-off relay coil to launch-vehicle ground that initiated the cut-off signal. Except for loss of expendable items on the spacecraft, such as the escape system and the parachutes and the peroxide, the spacecraft was in flight condition. The launch vehicle was slightly damaged in the aft section by recontact with the launcher The spacecraft and launch vehicle were demated. The launch vehicle was replaced by another Mercury-Redstone launch vehicle, and the spacecraft was again prepared for its mission. Modifications included a long ground strap that was placed between the launch vehicle and the launcher to maintain electrical ground until umbilicals had been separated. The refurbished spacecraft and new Mercury-Redstone launch vehicle were launched successfully as mission MR–1A on December 19, 1960. At this time, all test objectives were met. All major spacecraft systems performed well throughout the flight. The launch-vehicle performance was normal except for a higher than nominal cut-off velocity. The only effects of this anomaly were to increase the range, maximum altitude, and maximum acceleration during reentry. The spacecraft was picked up by a helicopter 15 minutes after landing and was delivered back to the launch site on the morning after the launch.

Mercury–Redstone 2.—The MR–2 mission was accomplished on January 31, 1961, from the Cape Canaveral test site with a chimpanzee as a passenger. Production spacecraft 5 was used. The mission was successful and the majority of the test objectives were met. Analyses of launch-vehicle data obtained during the flight revealed that launch-vehicle propellant depletion occurred before the velocity cut-off system was armed and before the thrust chamber abort switch was disarmed. This combination of events resulted in an abort signal being transmitted to the spacecraft from the launch vehicle. The spacecraft reacted correctly to the abort signal and an abort sequence was properly made. The greater than normal launch-vehicle velocity combined with the velocity increment obtained unexpectedly from the escape-rocket motor produced a flight path that resulted in a landing point about 110 nautical miles farther downrange than the planned landing point. This extra range, of course, was the prime factor in the 2 hours and 56 minutes that it took to locate and recover the spacecraft. The chimpanzee was recovered in good condition, even though the flight had been more severe than planned. By the time the spacecraft was recovered, it had nearly filled with water. Some small holes had been punctured in the lower pressure bulkhead at landing. Also, the heat-shield retaining system was fatigued by the action of the water and resulted in loss of the heat shield. Another anomaly that occurred during the flight was the opening of the spacecraft cabin inflow valve during ascent, which prevented the environmental control system from maintaining pressure at the design level. Because the pressure dropped below the design level, the emergency environmental system was exercised, and it performed satisfactorily. From the experiences of this flight, a number of modifications were made to the spacecraft systems to avoid recurrence of the malfunctioning items. These modifications included the following:

(1) An additional fiber glass bulkhead was installed between the heat shield and the large pressure bulkhead to protect the bulkhead during landing, and items in the large pressure bulkhead area that could be driven "dagger-like" through the larger pressure bulkhead during the landing were removed or reoriented.

(2) The heat-shield retention system was improved with the addition of a number of cables and cable-retention devices. The modified heat-shield retention system was proved to be capable of retaining the heat shield to the spacecraft in rough seas for periods of up to 10 hours.

(3) Tolerances of the inflow valve detent system were changed to assure positive retention during periods of vibration.

Mercury–Atlas 2.—The Mercury–Atlas 2 vehicle was launched from the Cape Canaveral test site on February 21, 1961, to accomplish the objectives of the MA–1 mission. The space vehicle for this flight consisted of the sixth production spacecraft and Atlas launch vehicle No. 67–D. Several structural changes made in the spacecraft launch-vehicle interface area as a result of the failure of the preceding Mercury–Atlas missions were as follows:

(1) The adapter was stiffened.

(2) The clearance between the spacecraft retropackage and the launch-vehicle lox tank dome was increased.

(3) An 8-inch-wide stainless-steel band was fitted circumferentially around the upper end of the launch-vehicle lox tank.

(4) The lox-valve support structure was changed so that the valve was not attached to the adapter.

(5) Special instrumentation was installed in the spacecraft launch-vehicle interface area to measure loads, vibrations, and pressures.

The major test objective of the MA–2 mission was to demonstrate the integrity of the spacecraft structure, ablation shield, and afterbody shingles for the most severe reentry from the standpoint of load factor and afterbody temperature. The flight closely matched the desired trajectory, and the desired temperature and loading measurements were obtained. The spacecraft landed in the planned landing area and was recovered and placed aboard a recovery ship approximately 55 minutes after it was launched. A preliminary evaluation of measured data and a detailed inspection of the recovered spacecraft indicated that all test objectives were satisfied and that the spacecraft structure and heat-protection elements were in excellent condition.

Little Joe 5A.—The Little Joe 5A mission was accomplished on March 18, 1961, from the Wallops Station launch site. This was an added mission, as a result of the failure of the Little Joe 5. For the Little Joe 5A mission, production spacecraft 14 and the sixth Little Joe launch vehicle to be flown were used. The spacecraft was a basic Mercury configuration with only those systems installed that were required for the mission. As during the Little Joe 5 mission early ignition of the escape-rocket motor occurred. The mission was unsuccessful. However, unlike the Little Joe 5 mission, a backup spacecraft separation system was initiated by ground command and successfully separated the spacecraft from the launch vehicle and released the tower. Because of the severe flight conditions existing at the time of parachute arming, both main and reserve parachutes were deployed simultaneously. They filled and enabled the spacecraft to make a safe landing. All other active systems operated properly except that the cabin pressure-relief valve failed to maintain the spacecraft cabin pressure because of a piece of safety wire found lodged in the seat. The spacecraft was recovered and returned to the launch area in good condition. Analysis of data from the spacecraft proved that the early ignition of the escape rocket motor was caused by structural deformation in the spacecraft-adapter interface area. This early ignition permitted separation sensing switches to falsely sense movement and give the signal for the remainder of the sequence. The corrections applied were to reduce air loading in the area by better fairing of the clamp-ring cover, by increasing the stiffness of the switch mounting and reference structures, and rerouting the electrical signals from these switches through a permissive network.

Mercury–Redstone–Booster Development.—The Mercury–Redstone–Booster Development (MR–BD) mission was made on March 24, 1961, from the Cape Canaveral launch site, with a Mercury-Redstone launch vehicle and the refurbished and ballasted Little Joe 1A research-and-development spacecraft. This flight was made as the result of the analyses of the performance of the launch vehicles on the Mercury-Redstone 1A and Mercury-Redstone 2 flights, which showed that there were some launch-vehicle problems that required correction and requalification. Most of these problems had to do with the overspeed performance that was attained during those missions. The flight was successful and analyses of the launch-vehicle data indicated that the launch-vehicle corrections were entirely satisfactory. No recovery of the spacecraft was attempted since it was used only as a payload of the proper size, shape, and weight, and no provisions were made to separate it from the launch vehicle during the mission.

Mercury–Atlas 3.—The Mercury–Atlas 3 (MA–3) mission was accomplished on April 25, 1961, from the Cape Canaveral test site. The planned flight, which was intended to orbit an unmanned production spacecraft once around the earth, was terminated about 40 seconds after lift-off by range-safety action when the launch vehicle failed to roll and pitch over into the flight azimuth. The spacecraft was aborted successfully as the result of the command signal and was quickly recovered. The spacecraft came through the abort maneuver with only minor damages. The performance of all spacecraft systems was generally satisfactory throughout the short flight. The spacecraft used on this mission was the eighth production unit. The launch vehicle, Atlas 100–D, had increased skin thickness in the forward end of the lox tank and had the abort sensing and implementation system installed for closed-loop operation. Analysis of records indicated that there was an electrical fault in the launch vehicle autopilot. Subsequent action resulted in closer examination of electrical components and connections.

Little Joe 5B.—The Little Joe 5B vehicle was launched on April 28, 1961, from the Wallops Station launch site. The vehicle was composed of Mercury production spacecraft 14A and the seventh Little Joe launch vehicle to be flown. The spacecraft, which had previously been used for the Little Joe 5A mission, had been refurbished with only those systems installed that were required for the mission. There was no landing bag and certain other nonessential systems were missing. It was the first spacecraft to be flight-tested with modified spacecraft-adapter clamp-ring limit-switch mountings and fairings. Also, the sequential system was modified to prevent the limit switches on the spacecraft-launch-vehicle clamp ring or the spacecraft-escape-tower clamp ring from closing any circuits which would ignite the escape rocket until the band separation bolts were fired. These changes in and around the spacecraft-launch-vehicle interface and in the sequential system were made as the result of the problems encountered in missions Little Joe 5 and Little Joe 5A. Because of a severe change in flight path as the result of the delayed ignition of one of the two main launch-vehicle rocket motors, the test was made at substantially more severe flight conditions than planned. The abort was planned to be initiated at a dynamic pressure of 990 lb/sq ft; instead the dynamic pressure had attained a value of about 1,920 lb/sq ft when the abort was initiated. However, the spacecraft escape system worked as planned and this test successfully demonstrated the structural integrity of the Mercury spacecraft. The spacecraft landed in the ocean after about 5 minutes of flight and was recovered and returned to the launch site in less than 30 minutes after launch. Analyses of the flight data and inspection of the spacecraft after the mission showed the spacecraft to be in good condition. An anomaly that showed up was the failure of two of the small spacecraft umbilicals to eject. Evidence indicated that these umbilicals failed to eject because of interference with the clamp-ring fairing after its release. This condition was corrected by changing the manner in which the fairing was supported on subsequent spacecraft. All test objectives were considered to have been met.

Mercury-Redstone 3.—The Mercury-Redstone 3 (MR–3) mission, the first manned space flight by the United States, was successfully accomplished on May 5, 1961, from the Cape Canaveral launch site. Astronaut Alan B. Shepard was the pilot. The space vehicle was composed of production spacecraft 7 and a Mercury-Redstone launch vehicle, which was essentially identical to the one used for the MR–BD launch-vehicle qualification mission. Analyses of the results of the mission showed that Astronaut Shepard satisfactorily performed his assigned tasks during all phases of the flight. Likewise, launch vehicle and spacecraft systems performed as planned. The spacecraft achieved an altitude of about 101 nautical miles and was in weightless flight for slightly over 5 minutes. Postflight examination of Astronaut Shepard and inspection of the spacecraft showed both to be in excellent condition. A helicopter pickup was made of the spacecraft after the pilot had made his egress from the side hatch of the spacecraft and had been hoisted aboard the helicopter. The pilot and the spacecraft were landed aboard an aircraft carrier 11 minutes after spacecraft landing, and the spacecraft was brought back to the launching site the morning after the flight.

Mercury-Redstone 4.—The Mercury-Redstone 4 (MR–4) flight was successfully made on July 21, 1961, from the Cape Canaveral launch site. Astronaut Virgil I. Grissom was the pilot. The space vehicle was made up of the 11th production spacecraft and a Mercury-Redstone launch vehicle essentially identical to the one used for MR–3 mission. The spacecraft on this mission was somewhat different from spacecraft 7, in that, for the first time, a manned spacecraft had a large top window, a side hatch to be opened by an explosive charge, and a modified instrument panel. The spacecraft achieved a maximum altitude of about 103 nautical miles, with a period of weightlessness of about 5 minutes. The flight was successful. After landing, premature and unexplained actuation of the spacecraft explosive side hatch resulted in an emergency situation in which the spacecraft was lost but the pilot was rescued from the surface of the water. Analyses of the data from the flight and debriefing by the astronaut indicated that, in general, the spacecraft systems performed as planned, except for the action of the spacecraft hatch. An intensive investigation of the hatch actuation resulted in a change in operational procedures. No fault was found in the explosive device.

Mercury-Atlas 4.—The Mercury-Atlas 4 (MA–4) vehicle was launched on September 13, 1961, from the Cape Canaveral launch site; it was a repeat of the MA–3 test and became the first Mercury spacecraft to be successfully inserted into orbit, returned, and recovered. Further objectives of this flight were to evaluate the

Mercury network and recovery operations concerned with orbital flight. The space vehicle for this flight was made up of Mercury–Atlas launch vehicle 88–D, with the same modifications as the launch vehicle used on the MA–3 mission, and the spacecraft which was used on the MA–3 mission. The spacecraft had been refurbished and designated 8A for this mission. This was a very complete spacecraft which included a man-simulator onboard to provide a load on the environment control system during orbital flight. Other differences between this spacecraft and spacecraft flown on subsequent missions were:

(1) The landing bag was not installed

(2) The spacecraft had small viewing windows rather than the large overhead window used on later spacecraft

(3) The spacecraft entrance hatch did not have the explosive-opening feature

(4) The instrument panel had a slightly different arrangement.

The launch vehicle provided the desired orbital path with a perigee of 85.9 nautical miles and an apogee of 123.3 nautical miles. The planned retromaneuver over the coast of California resulted in a landing in the Atlantic Ocean approximately 160 nautical miles east of Bermuda in the primary landing area. The spacecraft was recovered in excellent condition 1 hour and 22 minutes after landing. The mission achieved the desired objectives, even though certain anomalies showed up in systems behavior during the mission. None of the anomalies had serious consequence. The anomalies and action taken are as follows:

(1) A spacecraft inverter failed during the powered phases of flight. The cause was determined to be a vibration-sensitive component and found to be preventable by more precise and exacting acceptance tests.

(2) Some anomalies in the spacecraft scanner signals were detected during the mission. Steps were taken to modify the system to make it less sensitive to the effects of cold cloud layers.

(3) A leak developed in the spacecraft oxygen-supply system during the exit phase of the flight. The leak was small, and sufficient oxygen was available for the mission. Postflight analyses determined that the leak was caused by failure in a pressure reducer. The fault was corrected for subsequent missions.

(4) Some thrusters in the spacecraft automatic attitude control system had either reduced output or no output during the latter part of the orbit. Postflight analyses indicated that possibly the trouble was contamination of the metering orifices in some thruster assemblies.

Mercury–Atlas 5.—The Mercury–Atlas 5 (MA–5) mission was successfully made on November 29, 1961, from the Cape Canaveral launch site. A chimpanzee was the passenger on this flight. The mission was planned for three orbital passes and was to be the last qualification flight of the Mercury spacecraft and launch vehicle prior to a manned mission. The orbit was about as planned with perigee at 86.5 nautical miles and apogee at 128.0 nautical miles. Further objectives of this flight were to evaluate the Mercury network and recovery operations. In general, the spacecraft, launch-vehicle, and network systems functioned well during the mission until midway through the second pass when abnormal performance of the spacecraft attitude control system was detected and verified. This malfunction precluded the probably successful completion of the third pass because of the high rate of control fuel consumption. Accordingly, a retrofire command was transmitted to the spacecraft which resulted in it landing in the selected area at the end of the second pass. Recovery was completed 1 hour and 15 minutes after landing. The chimpanzee performed his assigned tasks without experiencing any deleterious effects during the mission and was recovered in excellent condition.

The primary anomaly during the mission was the control-system trouble which gave rise to increased fuel consumption by the attitude control system and which precipitated the abort of the mission at the end of the second orbital pass. The trouble was found to be a stopped-up metering orifice in one of the low-roll thrusters. Corrective action applied to subsequent missions included closer examinations for contamination in this system.

The spacecraft used for this mission was production spacecraft 9; and since it was the last qualification vehicle prior to the first manned orbital flight, it was intentionally made as nearly like the spacecraft for the manned mission as possible. This spacecraft included the large viewing window over the astronaut's head position, the landing bag, a positive lock on the emergency-oxygen rate handle, an explosive-release type hatch, new provisions for cooling the inverters, and rate gyros modified to insure satisfactory operation in the vacuum condition. The launch vehicle, Atlas 93–D, was much like those launch vehicles used on the previous two Mercury–Atlas missions; however, some additional modifications were included on this vehicle. These modifications included a new lightweight telemetry system and a redundant path for the sustainer engine cut-off signal.

Mercury–Atlas 6.—Mercury–Atlas 6 (MA–6), the first manned orbital space flight made from the United States, was successfully made on February 20, 1962, from the Cape Canaveral test site. Astronaut John H. Glenn, Jr., was the pilot. The flight was planned for three orbital passes to evaluate the performance of the manned spacecraft systems and to evaluate the effects of space flight on the astronaut and to obtain the astronaut's evaluation of the operational suitability of his spacecraft and supporting systems. All mission objectives for this flight were accomplished. The astronaut's performance during all phases of the mission was excellent, and no deleterious effects of weightlessness were noted. In general, the spacecraft, launch vehicle, and network system functioned well during the mission. The main anomaly in spacecraft operation was the loss of thrust of two of the 1-pound thrusters which required the astronaut to control the spacecraft for a large part of the mission manually. The orbit was approximately as planned, with perigee at 86.9 nautical miles and apogee at 140.9 nautical miles. During the second and third passes, a false indication from a sensor indicated that the spacecraft heat shield might be unlocked. This indication caused considerable concern and real-time analyses resulted in the recommendation that the expended retropackage be retained on the spacecraft during reentry at the end of the third pass to hold the heat shield in place in the event it was unlatched. The presence of the retropackage during reentry had no detrimental effect on the motions of the spacecraft. Network operation, including telemetry reception, radar tracking, communications, command control, and computing, were excellent and permitted effective flight control during the mission. The spacecraft for this mission was production unit number 13 which was essentially the same as spacecraft 9 used in the MA–5 mission except for those differences required to accommodate the pilot, such as the couch, a personal equipment container, filters for the window, and some minor instrumentation and equipment modifications. The launch vehicle was Atlas 109–D. It differed from the MA–5 launch vehicle in only one major respect. For this launch vehicle, the insulation and its retaining bulkhead between the lox and fuel tank dome was removed when it was discovered that fuel had leaked into this insulation prior to launch. The spacecraft landed in the planned recovery area, close to one of the recovery ships. The spacecraft, with the astronaut inside, was recovered approximately 17 minutes after landing. The astronaut was in excellent shape.

Action to prevent recurrence of the anomalies encountered during the MA–6 mission included relocation of metering orifices and a change in screen material in the attitude control system thruster assemblies. Improved specifications, tighter quality control, and more conservative switch rigging and wiring procedures were applied to the sensors that indicated heat-shield release.

Mercury–Atlas 7.—The Mercury–Atlas 7 (MA–7) vehicle was launched on May 24, 1962, from the Cape Canaveral launch site. Astronaut M. Scott Carpenter was the pilot for this mission. The mission was planned for three orbital passes and was a continuation of the program to acquire additional operational experience and information for manned orbital space flight. All objectives of the mission were achieved. The spacecraft used for this flight was production unit number 18 which was very similar to the spacecraft 13 used on the MA–6 flight. Some of the more significant features and modifications applied to this spacecraft include: the SOFAR bomb and radar chaff were deleted, the earth-path and oxygen partial pressure indicators were deleted, the instrument observer camera was removed, provisions for a number of experiments and evaluation were added, a more complete temperature survey system was added, the astronaut's suit circuit constant-bleed orifice was deleted, the landing-bag limit (heat-shield release) switches were rewired to prevent erroneous telemetry signals should one switch malfunction.

The launch vehicle, the Atlas 107–D, was similar to the previous Atlas launch vehicle except for a few minor changes, the major one of which was that for this mission, the fuel tank insulation bulkhead was retained. Launch-vehicle performance was satisfactory. A perigee of 86.8 nautical miles and an apogee of 145 nautical miles were the orbital parameters. During most of the flight, the spacecraft-system operation was satisfactory until, late in the third pass, the pilot noted that the spacecraft true attitude and indicated attitude in pitch were in disagreement. Because this control system problem was detected just before retrofire, no corrective action was possible and the astronaut was forced to provide manual attitude control, using the window and horizon as the attitude reference, for the retrofire maneuver. Retrofire occurred about 3 seconds late, and the optimum spacecraft attitudes were not maintained during retrofire. As a result, the spacecraft landed several hundred miles downrange of the planned landing point. Because of this, recovery of the astronaut was not accomplished until about 3 hours after landing. The spacecraft was retrieved later by a destroyer after about 6 hours in the water. Exact cause of the control system malfunction was not determined because the scanner circuitry suspected of causing the anomaly was lost when the antenna section was jettisoned during the landing phase. Changes in checkout procedures used in launch preparations were incorporated to prevent recurrence of this type of problem.

Mercury–Atlas 8.—The Mercury–Atlas 8 (MA–8) vehicle was launched from the Cape Canaveral launch site on October 3, 1962; Astronaut Walter M. Schirra, Jr., was the pilot. The MA–8 mission was planned for six orbital passes in order to acquire additional operational experience and human and systems performance information for extended manned orbital space flight. The objectives of the mission were successfully accomplished. The orbital parameters were as follows: perigee, 86.9 nautical miles; and apogee, 152.8 nautical miles. The space vehicle for this mission consisted of production spacecraft 16 and Atlas launch vehicle 113–D. The spacecraft was basically the same as spacecraft 18 utilized on the previous mission; however, a number of changes were made in the configuration to increase reliability, to save weight, to provide for experiments, and to conduct systems evaluations. The launch vehicle also had some changes as compared with the previous Mercury–Atlas launch vehicle. These changes include the following: the fuel tank insulation bulkhead was removed at the factory to be similar to the launch vehicle for the MA–6 mission, the two booster engine thrust chambers had baffled ejectors installed for improved combustion characteristics, and no holddown delay was programed between engine start and beginning of release sequence.

The pilot performed numerous experiments, observations, and systems evaluations during his mission. For the first time, extended periods of drifting flight were accomplished. Pilot adherence to the flight plan was excellent. Basic spacecraft systems, launch-vehicle systems, and ground-network systems performed well with only a few minor anomalies. The landing was made in the Pacific Ocean within sight of the primary recovery ship, and the spacecraft and pilot were recovered in about 40 minutes.

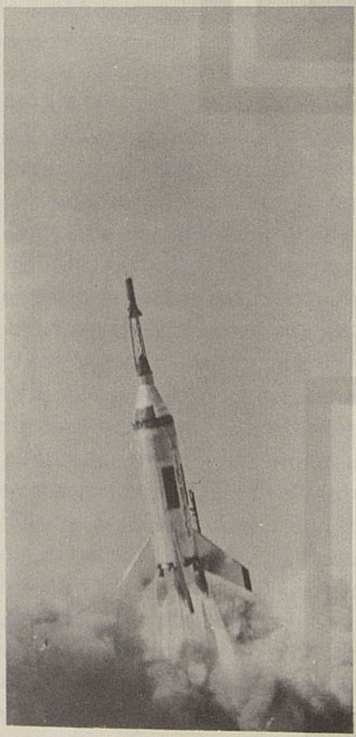
Mercury–Atlas 9.—The Mercury–Atlas 9 (MA–9) mission utilizing production spacecraft 20 and Atlas launch vehicle 130–D, was successfully accomplished on May 15 and 16, 1963, with Astronaut L. Gordon Cooper as the pilot. It was launched from the Cape Canaveral test site for a planned 22 orbital-pass mission. Launch-vehicle performance was excellent and a near perfect orbit was attained. The orbital parameters were as follows: perigee, 87.2 nautical miles; apogee, 144.2 nautical miles. For the first 18 orbital passes, the spacecraft systems performed as expected, and the pilot was able to adhere to the flight plan and perform his activities as planned. Up to that time, anomalies were limited to small nuisance-type problems. Beginning with the 19th orbital pass, the spacecraft systems problems began with actuation of the 0.05g warning light. Investigation of the occurrence of this warning light indicated that the automatic control system had become latched into the mode required for the reentry phase. Later, the alternating-current power supply for the control system failed to operate. These failures were analyzed by the pilot and the ground crew in real time and it was determined that the pilot would have to make a manual retrofire and reentry. He performed these meneuvers with close precision and landed a short distance from the prime recovery ship in the Pacific. The pilot and the spacecraft were recovered and hoisted aboard the carrier only 40 minutes after landing. More detailed results of this mission are contained in other papers in this document.

Lift-off photographs of the three types of Mercury space vehicles are shown in figure 1–4.

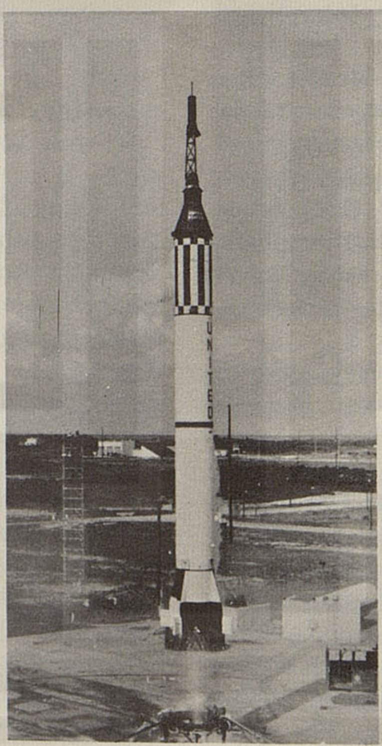
PERFORMANCE

An examination of the history of the major flight tests, presented in figure 1–3, will show that the basic objectives of the Mercury Project were achieved 3⅓ years after official project approval, with the completion of Astronaut John Glenn's successful orbital flight on February 20, 1962. Subsequently, Astronaut Carpenter completed a similar mission. Then, Astronauts Schirra and Cooper completed orbital missions of increased duration to provide additional information about man's performance capabilities and functional characteristics in the space environment. In addition, increasing numbers of special experiments, observations, and evaluations performed during these missions by the pilots as their capabilities were utilized have provided our scientific and technical communities with much new information. It is emphasized that goals beyond those originally established were achieved in a period of 4⅔ years after the beginning of the project with complete pilot safety and without change to the basic concepts that were used to establish the feasibility of the Mercury Project.

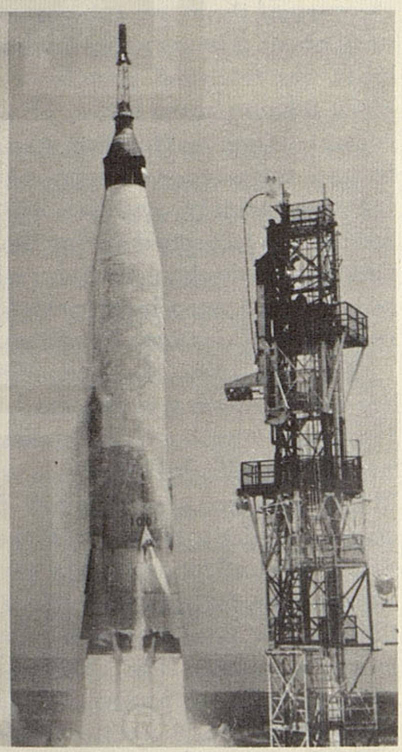
In early 1959, immediately after project go-ahead, the first manned orbital flight was scheduled to occur as early as April 1960, or 22 months before the event actually took place (see fig. 1–5). This difference was caused by an accumulation of events which included delays in production spacecraft deliveries, difficulties experienced in the preparations for flight, and by the effects of the problem areas that were detected during the development and early qualification flight tests. The primary problem areas included those which were associated with the spacecraft–launch-vehicle struc-



Little Joe



Redstone



Atlas

FIGURE 1–4.—Lift-off photograph of the three types of Mercury space vehicles.

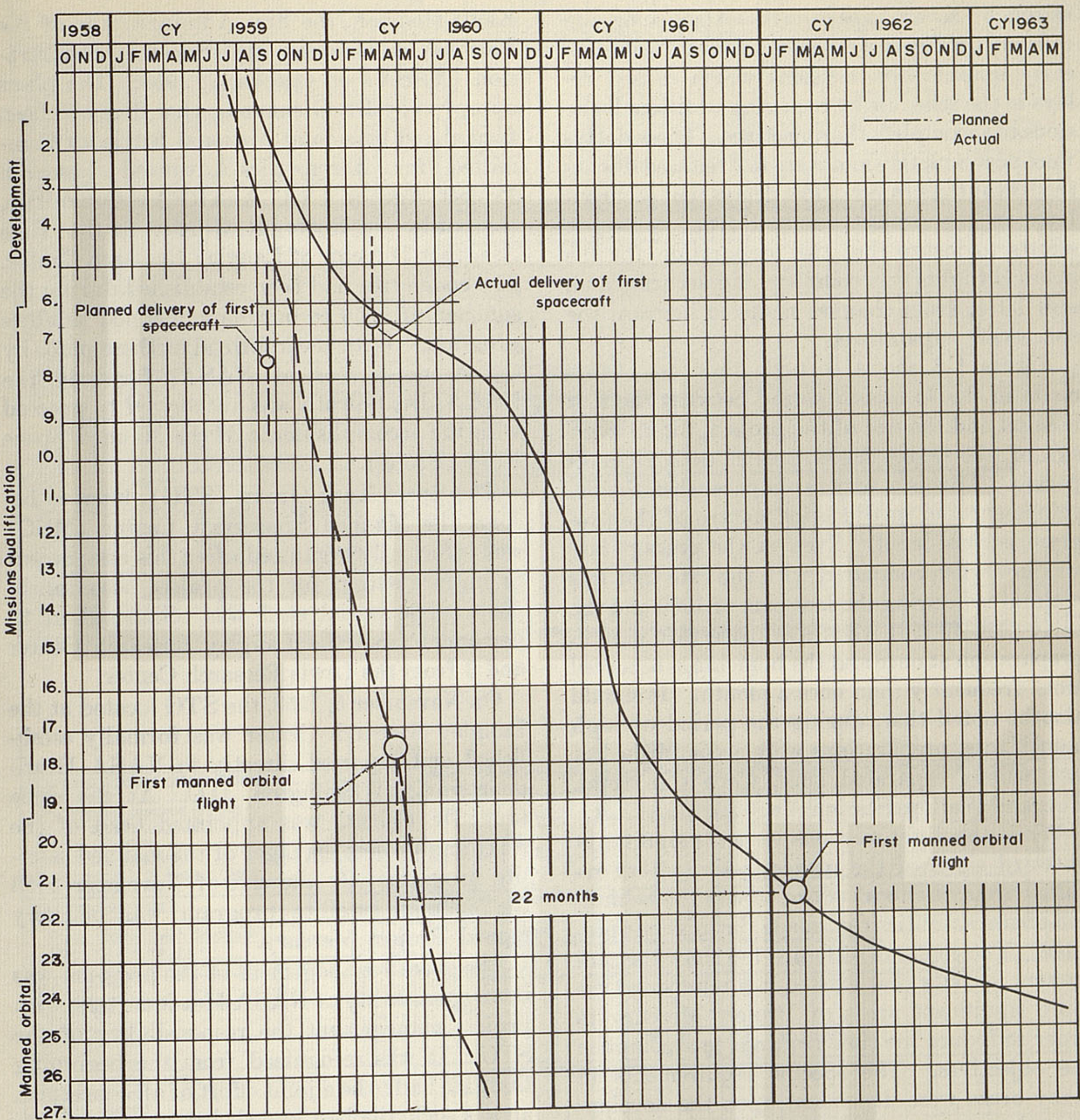


FIGURE 1–5.—Comparison of planned and actual flight schedules.

tural interface on the MA–1 mission, spacecraft sequential-system sensors on Little Joe missions 5 and 5A, launch-vehicle umbilical-release sequence on the MR–1 mission, launch-vehicle propulsion system on MR–2, and launch-vehicle control system on MA–3.

The applicability of these statements can be illustrated by reference line representations of the planned and actual schedules that are compared in figure 1–5. This comparison shows that the flight-test program was intiated about 1 month late. Missions through the development phase and those missions accomplished through most of the qualification phase were accomplished at about the planned rates. The major deviations occurred in 1960 when production spacecraft deliveries were later and when launch preparation took longer than planned. The planned schedule allowed for about a 4-week prelaunch preparation period at the launch site. Actual preparation time averaged about six times the estimated amount. Some of the additional required preparation time was compensated for by concurrent preparation of several spacecraft. Also, some significant problems were encountered during the early qualification missions which caused delays in the schedule by requiring additional missions to accomplish the objectives. These delays were accumulative and were not reduced during the life of the project. The delays that occurred later in the project resulted from deliberate efforts to insure that the preparation for the manned flights was complete and accurate and, still later, from changes made to increase the spacecraft capabilities.

Figure 1–3 shows that 25 flight tests were made in the 45-month period between the first mission and the end of the project, for an average of about one flight test in each 2-month period. This is a very rapid pace when the development and qualification nature of the program is considered. Even so, the average rate was low when compared with the rate that was maintained during the last part of 1960 and the early part of 1961 when five spacecraft were in preparation at once and the launchings occurred more frequently than once a month. It should also be noted that, during the period of high launch rate, preparations were accomplished at two widely separated sites, Cape Canaveral, Fla., and Wallops Station, Va.

While the flight missions were the significant outward signs of the project activity that resulted from the total effort, it was the behind-the-scenes activities that made the missions possible. The contents of figure 1–6 show the concurrent activity that existed in a number of the more significant areas of Project Mercury in order to reduce the time required to accomplish the objectives. The specific requirements in many areas were dependent upon the development being accomplished in the other areas. Thus, there was a continual iteration process carried on which resulted in a gradual refinement of requirements and completion of the work.

Management

Modes of Operation

Development of the management structure and operating mode to direct this complex and rapidly moving project began concurrently with the approval of the plans for a program of research and development leading to manned space flight which were presented to Dr. T. Keith Glennan, the first Administrator of the National Aeronautics and Space Administration (NASA) on October 7, 1958. The plans approved by Dr. Glennan on that date had been formulated by a joint National Advisory Committee for Aeronautics-Advanced Research Project Agency (NACA–ARPA) Committee, chaired by Dr. Robert R. Gilruth, at that time Assistant Director of Langley Research Center. The committee had been established during the summer of 1958 to outline a manned satellite program. With the approval of these plans by the Administrator of NASA, formerly the NACA, Dr. Gilruth was authorized to proceed with the accomplishment of the Manned Space Flight Project.

The Space Task Group (STG), later to become the Manned Spacecraft Center (MSC) was informally organized after this assignment to initiate action for the project accomplishment. The initial staff was comprised of 35 personnel from the Langley Research Center and 3 from the Lewis Research Center.

On November 5, 1958, the STG located at the Langley Research Center was formally established and reported directly to NASA Headquarters in Washington, D.C. At the same time, Dr. Gilruth was appointed head of the STG and project manager of the manned satellite program. By the end of November 1958 the manned satellite program was officially named Project Mercury.

The overall management of the program was the responsibility of NASA Headquarters, with project management the responsibility of the STG. It was recognized from the beginning that this had to be a joint effort of all concerned, and as such, the best knowledge and experience as related to all phases of the program and the cooperation of all personnel was required if success was to be achieved. It was also recognized that it was an extremely complex program that would probably involve more elements of government and industry than any development program before undertaken. Because of this complexity and involvement of so many elements, management was faced with an extremely challenging task of establishing an overall operating plan that would best fit the program and permit accomplishment of all objectives at the earliest possible date. To achieve

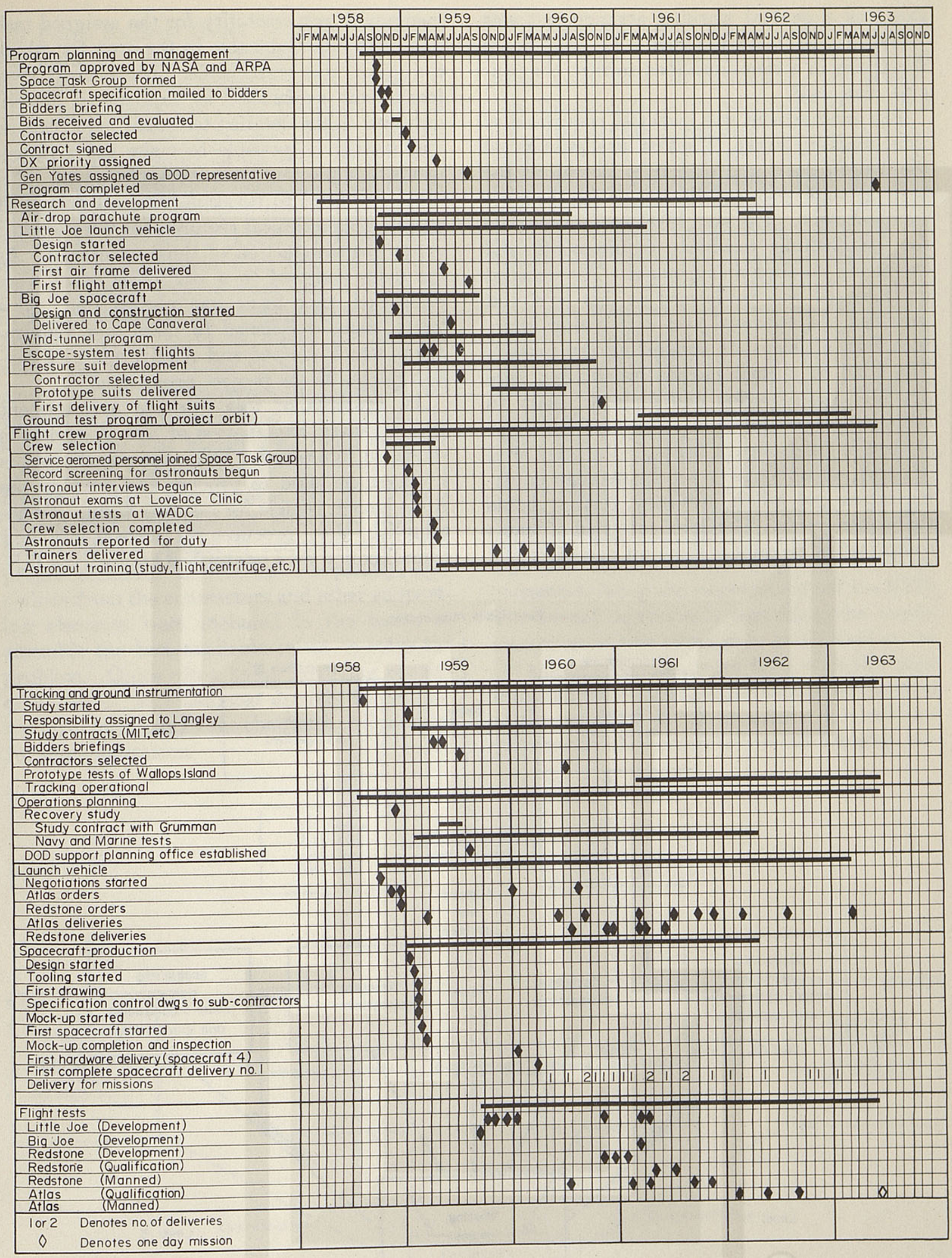


FIGURE 1–6.—Areas of primary activity during Project Mercury.

this task a general working arrangement was established as shown in figure 1–7. This figure illustrates in a very simplified format, the general plan used.

The arrangement was basically comprised of three working levels. The first level established the overall goals and objectives as well as the basic ground rules and the means for their accomplishment. The next level was responsible for establishing technical requirements and exercising detailed management. The detailed management was performed at this level and provided the approval and authorizing interface with all elements supporting the project. The bond of mutual purpose established here provided the direction and force necessary to carry the project forward. This same bond was evident in the groups or teams, in the third level of effort, set up to carry out the detailed implementation and, where necessary, further define the requirements. This level consisted of teams comprised of personnel from all necessary elements with responsibility for the assigned task and most knowledgeable in the area for which the group was responsible. These third level teams were established as required to investigate and define detailed technical requirements and insofar as possible to make the arrangements to implement their accomplishment. The team continued to function until all details of a particular technical requirement were worked out to the satisfaction of those concerned. As the tasks assigned to a particular team were completed, that team was phased out. New teams were established to meet new requirements which evolved and requirements of various phases as the project progressed.

An example of this working arrangement with a general explanation of how it worked is shown in figure 1–8. This example shows the arrangement used to procure and develop the Atlas launch vehicle for manned flight. To accomplish this, procurement agreements and overall policy were established between the U.S.

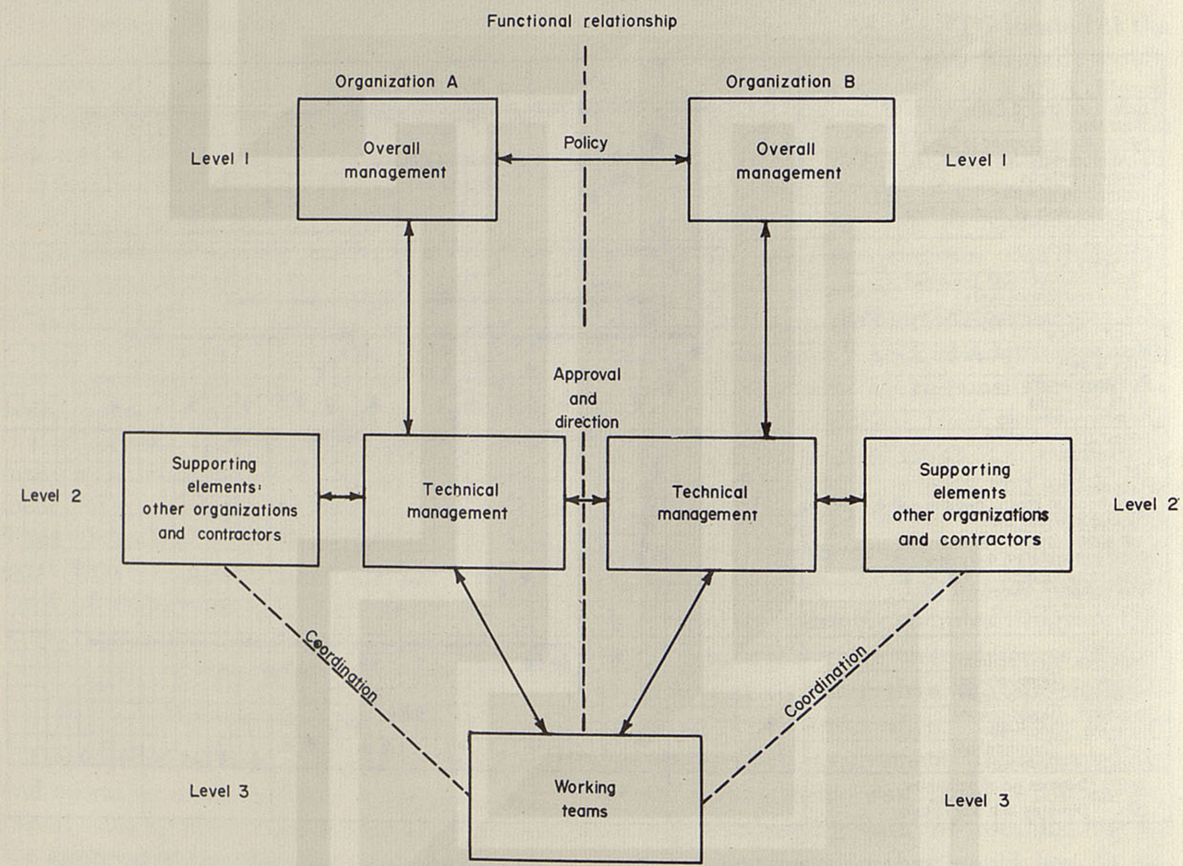


FIGURE 1–7.—Typical management arrangement.

Air Force Ballistic Missile Division of the Department of Defense and the NASA Headquarters. Working within the framework of these agreements the Atlas Weapons Systems Command of the U.S. Air Force and the NASA STG formulated the basic technical requirements necessary to adapt the Atlas for use in the program. Working teams consisting of specialists from the STG and the Atlas Weapons Systems Command were established to define the detail requirements and initiate the necessary action for their implementation. This implementation could be direct for cases in which the team had the authority or the recommendation for implementation could be forwarded to the necessary level of authority. In any case, the next higher level could alter the decisions of the lower level if developments required. This arrangement also provided a "closed-loop" management structure, thus assuring positive means of communication and proper technical directions. Frequently, specialists from the contractors and other supporting elements were included in the teams to assemble the best available talent to solve the problem. Quite often, tasks involving considerable effort were assigned directly to individual team members by the chairman of the group for implementation.

The same general arrangement was employed between NASA elements in accomplishing major tasks, such as establishing the Worldwide Tracking Network, as illustrated in figure 1–9. In addition to the many overall arrangements that had to be made in establishing the Worldwide Tracking Network, such as agreements with foreign governments, working through the State Department, regarding the location and operation of ground stations in their territory, the task of providing the hardware and facilities that made up the ground stations represented a major task that was primarily the responsibility of the STG and the Langley Research Center. This example covers the means by which the basic technical requirements and hardware needs of the ground stations were accomplished through the combined efforts of the STG and Langley. The Langley Research Center was responsible for the procurement and establishment of the network, with the basic flight monitoring and control requirements being the responsibility of the STG. The overall agreements regarding the implementation of this effort were established at the Director-Project Manager level with the basic technical requirements being defined at the level of the cognizant divisions. After the basic requirements were presented to the Langley Re-

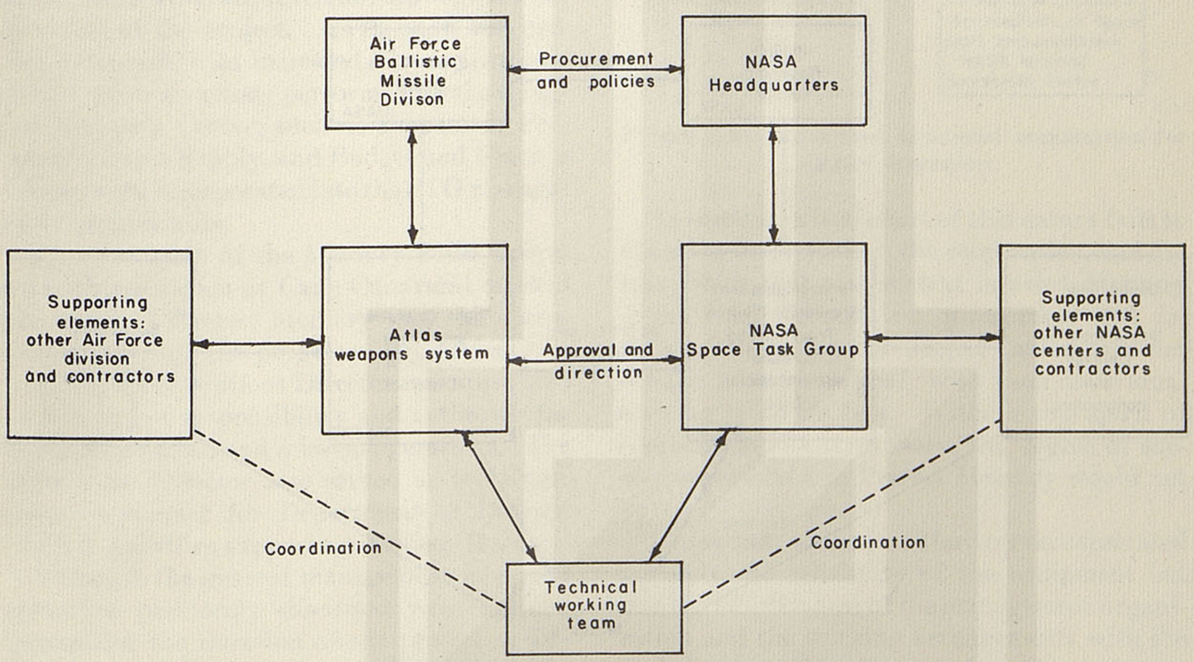


FIGURE 1–8.—Management arrangement used to procure, develop, and prepare the Atlas launch vehicle for manned flight.

search Center, teams were established to discuss and resolve the detail technical requirements of the network. For example, a team was assigned the task of establishing the communications and tracking requirements and resolving the type of equipment to be used on the spacecraft and the detail design characteristics of this equipment. They then had to determine if suitable receiving equipment for the ground stations was available or if it had to be developed. This involved coordinating overall requirements given to both the Langley Research Center's ground station contractors and the STG's spacecraft contractor to determine if the desired requirement could be achieved and if not, to determine an acceptable means of achieving the desired results. This points out only one detail area that this kind of group had to resolve; other areas such as location of the ground stations, frequencies of transmission, bandwidths, spacecraft antenna radiation patterns, and so on presented the same type of problems that had to be resolved. These efforts evolved into the Mercury Worldwide Tracking Network, the operation of which was the responsibility of the Goddard Space Flight Center (GSFC). Similar arrangements existed between the many elements necessary to develop the network and implement its operation.

To illustrate further this type working arrangement the identifications on figure 1–7 could be changed to represent those of the STG and the spacecraft contractor, McDonnell Aircraft Corporation (MAC). In this instance it was recognized by both parties that normal contractual procedures alone were insufficient to achieve the desired results within the scheduled time frame. Direct communication regarding technical requirements between the specialists of STG and MAC had to be the rule rather than the exception. Management agreements on the upper levels provided the framework whereby this could be accomplished and provided the management decisions for project direction. Frequently, the teams determined a course of action and proceeded without further delay, with verification documentation following through regular channels. The "closed-loop" built into the working arrangement provided the assurance that contractual and program requirements were met in all cases. Regular management reviews of hardware status and task achievement kept management abreast of the problem areas and afforded the opportunity for timely direction of effort to many specific problem areas. This mode of operation enhanced the rapidity with which a design change could be implemented or a course of action altered. This contributed to the timely conclusion of a project.

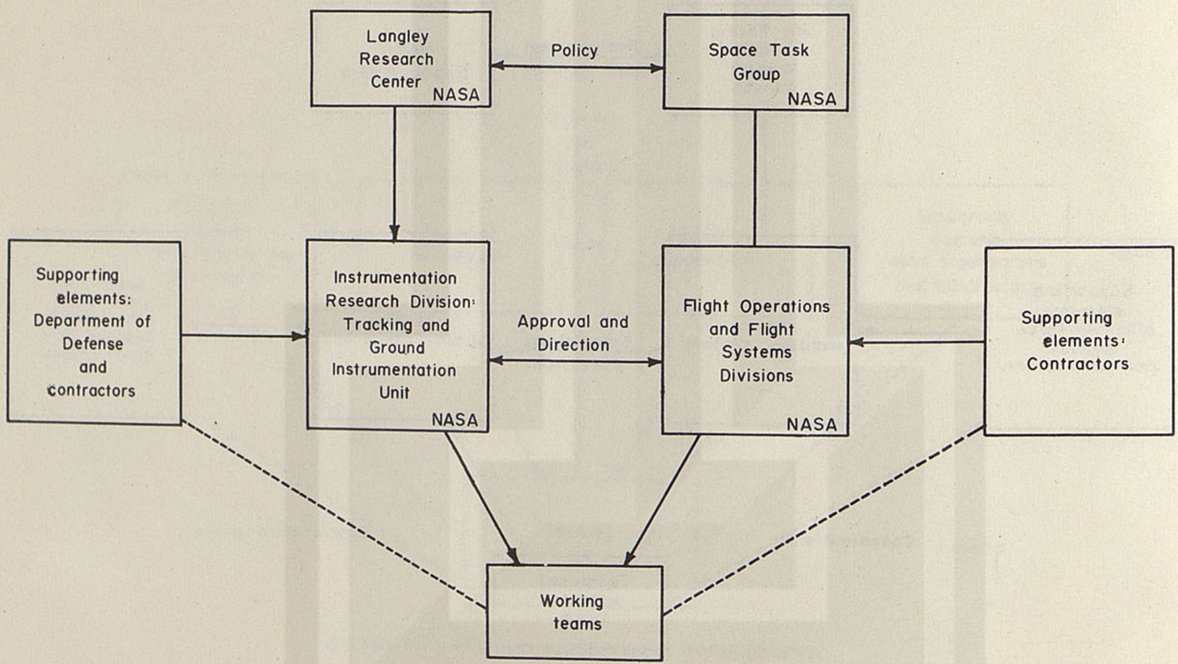


FIGURE 1–9.—Management arrangement used to establish the ground tracking organization.

The foregoing discussion is primarily concerned with the management techniques that existed with the external organizations, but the same type of procedure was commonly used within the organizational structure of the STG. As firm definition of the program emerged and final spacecraft design details were formalized, it became necessary to centralize the coordinating effort within the STG. To accomplish this, centralized review meetings were conducted on a regular basis to correlate all elements of the effort and ascertain that unified approaches and directions were maintained. These meetings were attended by cognizant personnel from within the STG and by personnel from other activities when required. The primary function of these meetings was to obtain the best inputs available for the technical management of the project and to control the engineering and design and thereby the configuration of the spacecraft. Information channeled into these meetings was dispersed directly to the responsible individuals within the STG, with assignments being made directly to the cognizant organization when action was required. Technical direction required as a result of action initiated at the coordination meetings, after thorough review as to need, cost, and effect on schedule, was issued to the applicable contractors. Meetings of this type provided fast response and accurate direction throughout the duration of the project. As the staff and project responsibilities increased, the support administrative functions performed by the Langley Research Center, such as Personnel, Procurement and Supply, and Budget and Finance Offices, were incorporated into the STG management organization.

The formation of the Mercury Field Operations Organization at Cape Canaveral marked the entry of Project Mercury into the operational phase of the program. In conjunction with this an Operations Director was appointed with complete responsibility and authority for flight preparation and mission operations. The Operations Director also served as the single point of contact for Department of Defense (DOD) activities supporting Project Mercury.

Although the general management modes of operation previously discussed were applied throughout the duration of the project, a different type functional organization was established for the specific purpose of conducting a space-flight mission. The organization covering the flight operations phase of the project was a line organization with elements from the government and contractor organizations involved in the operation reporting directly to the Operations Director. Figure 1–10 illustrates the manner in which these elements merged to form this functional line organization.

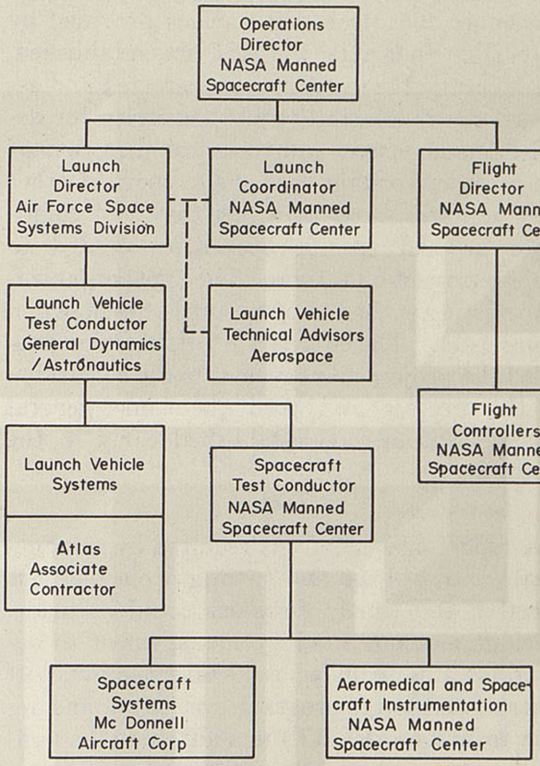


FIGURE 1–10.—Integrated functional organization for launch operations.

An organizational chart of this nature fails to show the unified effort, the cooperation, and the team work that was evident in every Mercury flight. All elements of government and industry supporting the project pulled together toward a common goal, with each individual striving to do his best. Without this spirit of cooperation and team work, the degree of success experienced in Project Mercury would not have been possible.

The success of Project Mercury demonstrated not only the reliability of the equipment but also the effectiveness of the management organization and the working arrangements with the various supporting elements throughout government and industry. Efforts to assure that Project Mercury would meet its objectives evolved in the high level agreements that resulted in clear lines of authority and responsibility for technical direction.

With the increasing national effort in the field of space exploration, additional manned space projects were assigned to the STG. Because of the increased emphasis and scope of the manned spaceflight effort, the MSC was established in November 1961 from the nucleus provided by the STG. Soon after the MSC was established, the Mercury Project Office was created and assigned the responsibility and authority for detailed management and technical direction of the project, working with the support of other MSC units in areas in which they had cognizance or had specific specialties needed to achieve project objectives. The MSC organization existing at the end of the project is shown in figure 1–11. The Mercury Project Office provided the project management to the conclusion of the project and used the same general management method established early in the program.

Tools

A reporting system was required by management to control the fast-moving project so that effective and timely decisions could be made. Various methods used by management to accomplish this included reports, schedules, cost control, and later, program evaluation and review technique (PERT) in addition to the technical reviews previously mentioned.

Many types of technical reports were prepared for management in order to keep it abreast of progress and problems. These reports were concise and factual status reports issued daily, weekly, monthly, and quarterly to highlight progress or lack of progress without conjecture. Obviously, close to the launch date, the daily reports became the most important. Another valuable report was the one prepared after the completion of each mission. These were prepared expeditiously to present analyses of the performance of all the systems involved in the mission, from the lowest elements through operational recovery techniques. The results of these analyses were used immediately after a mission to form the basis for corrective action that often influenced the hardware on the very next mission. These results were issued in formal report formats that contained detailed descriptions of the mission and equipment, performance analyses, result of investigations of anomalies, and much of the data. The reporting effort became greater as the complexity and duration of the missions increased, and larger reports and longer preparation times resulted. However, in most cases, the reports were printed for distribution within 30 days after the mission. The report of the MA–9 mission, for example, contained more than 1,000 pages of information.

Innumerable documents were generated covering all aspects of the program during the life of Project Mercury so that management as well as the individual elements could have overall knowledge of project details and progress. These documents were prepared by all elements participating in the program and included such general types as drawings, familiarization manuals, specifications, operational procedures, test procedures, qualification status, test results, mission results, reports on knowledge gained and status reports of all kinds. It is estimated that at least 30 formal documents, excluding drawings, engineering change orders, and so forth, were issued during the course of the project. A partial listing of the types of documentation used during the program is included in appendix A.

Overall schedule control was accomplished by the use of a Master Working Schedule which indicated major milestones, such as spacecraft deliveries and checkout periods, launch-vehicle deliveries and checkout times, launch-complex cleanup and conversion, and tracking network status. Detailed bar-chart schedules were maintained in areas of direct concern, such as individual spacecraft at the manufacturer's plant, launch preparation of the spacecraft and launch vehicle at the launch site, astronaut training, and the major test programs.

To control cost, management constantly monitored commitments, obligations, and expenditures through the normal accounting techniques. During the later phases of the program, the project office maintained cost control charts on which approved programed funds were shown, as well as obligations for a given time period. From these charts, management could tell at a glance the amount of remaining unobligated funds for any given area.

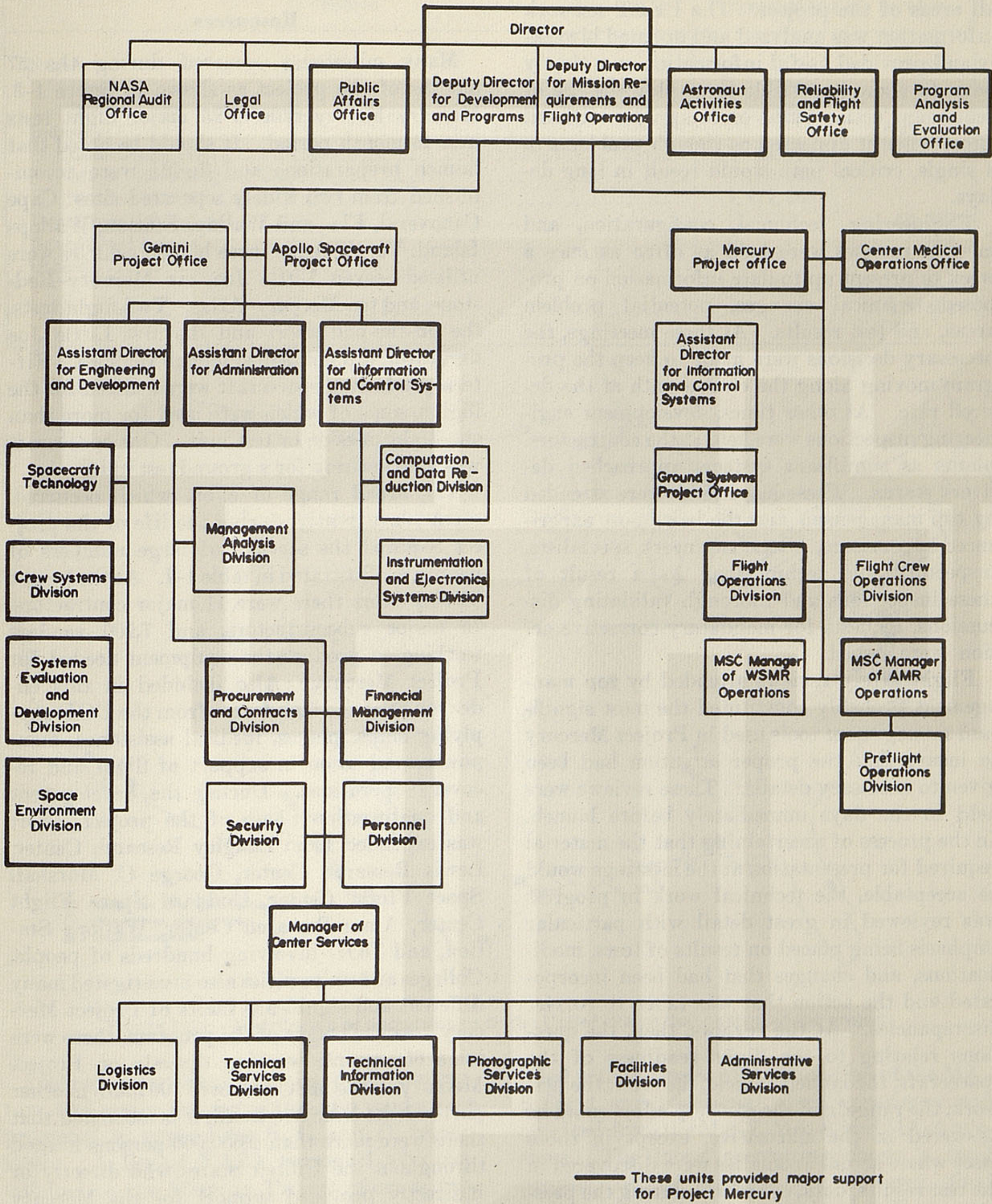


FIGURE 1-11.—Organization existing at end of Project Mercury.

In the last year and a half, the Manned Spacecraft Center applied the PERT system to cover all areas of the project. The PERT network information was analyzed and updated biweekly and provided useful information on a timely basis to make it possible to employ the use of redundant action paths or to apply additional effort when it appeared as though problems in a single, critical path would result in long delays.

Engineering, technical, configuration, and mission reviews were held as often as once a week to present up-to-date information on proposed technical changes, potential problem areas, and test results. At these meetings, the necessary decisions were made to keep the program moving along the chosen path at the desired rate. At other times, development engineering inspections were held at the contractors' plants as significant systems approached delivery status. These inspections were attended by top management and the best, most experienced supervisors, pilots, engineers, specialists, inspectors, and technicians. As a result of these inspections and thorough validating discussions, requests for mandatory corrective action were issued.

Flight safety reviews attended by top management probably constituted the most significant management tools used in Project Mercury to insure that the proper attention had been given to necessary details. These reviews were held in the days immediately before launch. In the process of ascertaining that the material required for presentation at the meetings would be acceptable, the technical work in progress was reviewed in great detail with particular emphasis being placed on results of tests, modifications, and changes that had been incorporated and the action that was taken to correct discrepancies. At the reviews, then, the questions relating to the flight readiness of the spacecraft, the launch vehicle, the crew, the network, the range, and the recovery effort could be answered in the affirmative, except in those cases where actual anomalies were discovered in the test results, data, or records during the presentation. Of course, these anomalies were then completely corrected or resolved, because no Mercury launchings were ever made in the face of known troubles or unresolved doubts of any magnitude that could affect mission success or mission safety.

Resources

Many milestones occurred during the 57 months of the project as shown in figure 1–3. Mercury history reflects 25 major flight tests in a 45-month period. It should be noted that launch preparations and flights were accomplished from two widely separated sites: Cape Canaveral, Fla., and Wallops Station, Wallops Island, Va. Twenty-three launch vehicles were utilized—seven Little Joe, six Mercury–Redstone, and ten Mercury–Atlas. Two flight tests, the off-the-pad abort and the first Little Joe flight test, did not utilize launch vehicles. Fifteen production spacecraft were utilized for the flights, some of which were used for more than one flight mission or test unit. One spacecraft was used entirely for a ground test unit.

The broad range of effort which occurred, often concurrently, during the life of the project required the services of large numbers of people, as illustrated in table 1–I. At the height of this effort there were 11 major contractors, 75 major subcontractors, and 7,200 vendors working to produce the equipment needed for Project Mercury. Also included in this endeavor were the task forces from the DOD supplying ships, planes, medical assistance, manpower, and so on in support of flight and recovery operations. During the development and qualification phase of the project, effort was expended from Langley Research Center, Lewis Research Center, George C. Marshall Space Flight Center, Goddard Space Flight Center, Ames Research Center, Wallops Station, and DOD involving hundreds of people. Colleges and universities also investigated many different and significant facets of Project Mercury. At the height of the program, there were some 650 people working directly on Project Mercury in the MSC and over 700 more in other parts of the NASA. In all, it is estimated that there were more than 2,000,000 persons located throughout the United States who directly or indirectly provided support for the Mercury Program. The general locations of the major contractors, universities, NASA centers and other government agencies are illustrated in figure 1–12.

Table 1-I.—Peak Manpower Support

Source	Approximate peak numbers	
NASA:		1,360
Direct	650	
Research and development	710	
Industry:		2,000,000
Contractors (11)	33,000	
Major subcontractors (75)	150,000	
Vendors (7,200)	1,817,000	
Department of Defense		18,000
Educational groups		168
Others		1,000
Total		2,020,528

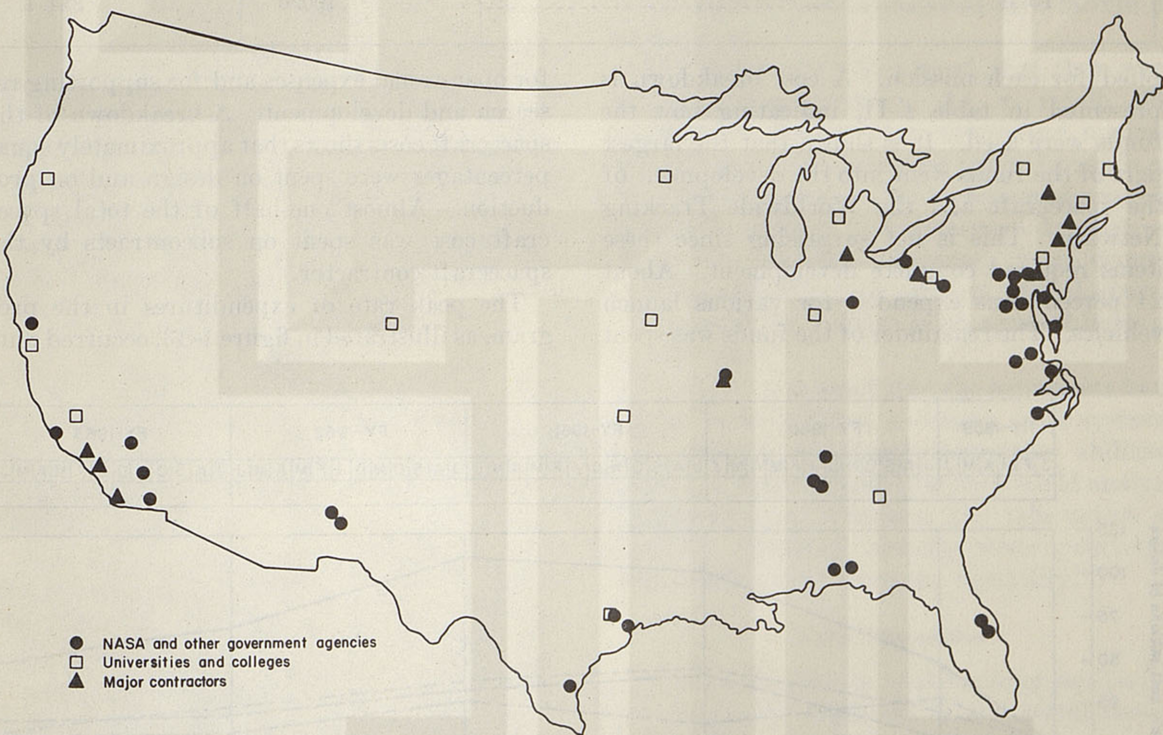


Figure 1-12.—Distribution of organizations in the United States that supported the project.

Lists of government agencies, prime contractors, and major subcontractors and vendors are presented in appendixes B, C, and D, respectively. A list of NASA personnel who contributed to the Mercury Project effort is presented in appendix E.

The total cost of the Mercury Program as published in the Congressional Committee Record in January 1960 was estimated to be $344,500,000. The basic objectives were fulfilled with the successful completion of the MA-6 flight and additional space experience was obtained from the MA-7, MA-8, and MA-9 missions. The latest accounting shows a total project cost of $384,131,000; however, final auditing has not been completed. These cost figures include the cost of the Mercury tracking network which will be used for manned space programs for years to come, and the cost of the operational and recovery support sup-

Table 1–II.—Cost Breakdown

Breakdown	Percent of total	Cost in millions of dollars
Spacecraft:	37.6	144.6
Design	8.6	33.2
Production	5.6	21.7
Test and flight preparation	4.2	15.9
Subcontract	16.2	62.2
Qualification	3.0	11.6
	37.6	144.6
Network	32.4	124.6
Launch vehicles	23.7	90.9
Operations	4.3	16.4
Supporting development	2.0	7.6
Total	100.0	384.1

plied for each mission. A cost breakdown is presented in table 1–II, indicating how the funds were used. It is shown that the largest part of the funds went into the development of the spacecraft and the Worldwide Tracking Network. This is not surprising since these items required complete development. About 24 percent was expended for various launch vehicles. The remainder of the funds was spent for operational expenses and for supporting research and development. A breakdown of the spacecraft costs shows that approximately equal percentages were spent on design and on production. Almost one-half of the total spacecraft cost was spent on subcontracts by the spacecraft contractor.

The peak rate of expenditures in the program, as illustrated in figure 1–13, occurred dur-

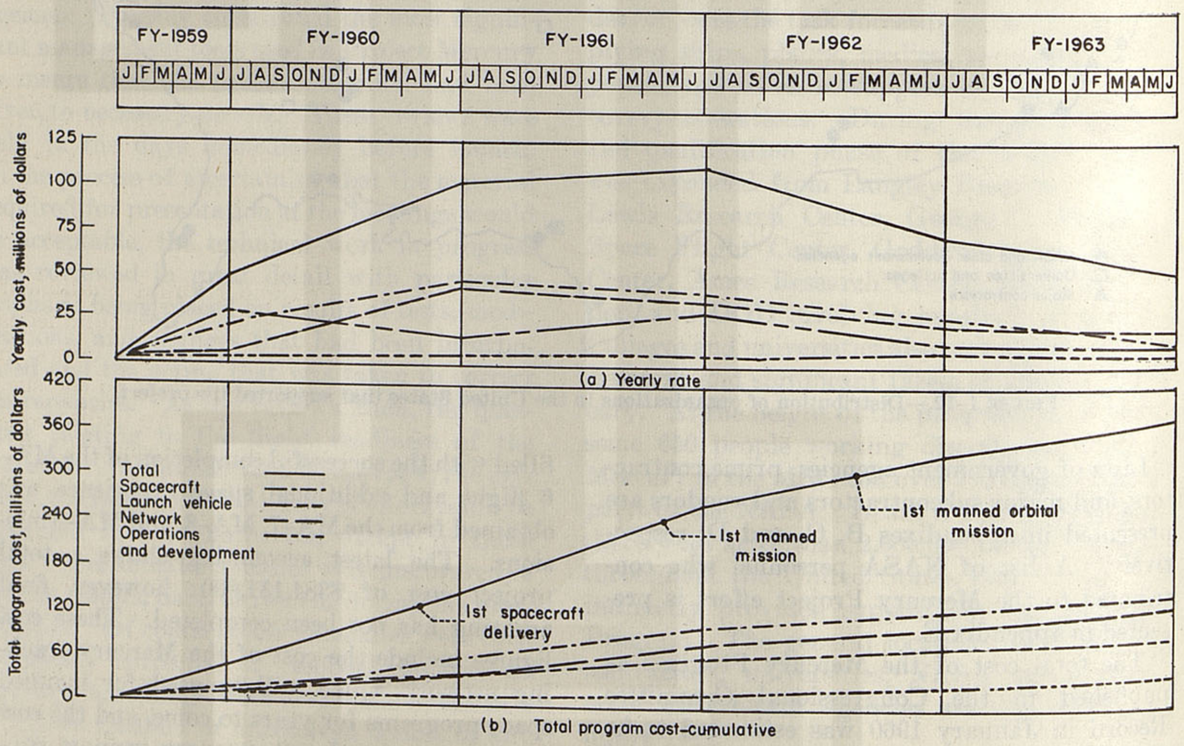


FIGURE 1–13.—Rate of expenditures and accumulated cost.

ing the fiscal year of 1961 and can be attributed to several factors. During this period, more than half of the total production spacecraft were delivered and more major flight missions were accomplished than in any other comparable time period. Launch activities were supported both at Wallops Station, Va., and at Cape Canaveral, Fla. Funds were being spent on the Worldwide Tracking Network for the coming orbital missions. The Redstone phase of flight program was nearing completion and the Atlas phase was approaching a peak. Also, much astronaut training was accomplished and the first manned ballistic flight was completed during this period.

Technical Experience

The major results obtained and the significant philosophies and techniques developed during the course of the project are grouped for discussion in the following areas: physiological and psychological responses of man in the space environment, flight and ground crew preparational procedures, and techniques and philosophy for launch preparation.

Responses of Man

The manned Mercury flights produced considerable information on human response and general physiological condition. Some of the most significant results may be summarized as follows:

(1) Results of repeated preflight and postflight physical examinations have detected no permanent changes related to the space-flight experience, although Astronauts Schirra and Cooper temporarily showed indications of orthostatic hypotension after their missions.

(2) There have been no alarming deviations from the normal, and the astronauts have proved to be exceedingly capable of making vital decisions affecting flight safety, taking prompt accurate action to correct systems deficiencies, accomplishing spacecraft control, and completing all expected pilot functions.

(3) The weightless state for the time periods of up to 34 hours has shown no cause for concern. Food and water have been consumed and the astronaut has slept. No abnormal body sensations and functions have been reported by the astronauts. The health of all of the astronauts has been good and remains so.

Not only has it been found that man can function normally in space, at least up to a maximum of 34 hours, but it has been found that he can be depended upon to operate the spacecraft and its systems whenever it is desired that he do so. On the MA–6 and MA–7 missions, the astronauts overcame severe automatic control system difficulties by manually controlling their spacecraft for retrofire and reentry. Also, on the MA–9 mission, the performance of the astronaut demonstrated that man is a valuable spacecraft system because of his judgment, his ability to interpret facts, and his ability to take corrective action in the event of malfunctions which would have otherwise resulted in a failure of the mission.

The astronauts also proved that they were qualified experimenters. As a result, the weight allocated in each succeeding manned orbital space flight increased from 11 pounds on MA–6 to 62 pounds on MA–9 for equipment not related to mission requirements. In each of these missions, the astronauts have demonstrated their ability to perform special experiments and to be a scientific observer of items of opportunity.

It can be concluded that the astronauts have proved to be qualified, necessary space systems, with flexible, wide-band-observation abilities, and have demonstrated that they could analyze situations, make decisions, and take action to back up spacecraft systems when provisions were made to give them the capability.

Crew Preparation

Studies, simulators, and training equipment for preparing flight crews and simultaneous participation of flight and ground crews in simulated missions were important to the success of the mission. This training is discussed in detail in later papers of this document. Before the final round of training and simulation began, it was found necessary to formulate and freeze a well-defined, detailed flight plan. This must be done far enough in advance of the mission to give the pilot sufficient time to train to the particular plan with the ground network teams who will support him during the mission. It has also been found to be important to avoid filling every available moment of the flight with a planned crew or ground-station activity. Time must be available to the flight crew to manage the spacecraft systems and to investigate anomalies or malfunctions in the system and to observe and measure the unexpected. Time must be provided to allow the pilot to consider thoughtfully his reactions to the space environment and its effects upon him. He must have time to eat and drink and to obtain sufficient rest. Training in simulator devices has proved to be a valuable tool for preparing a man for space flight. Well in advance of his flight, the pilot must have detailed training in the basic systems and procedures for the mission. In addition to preparing the pilot for normal and emergency flight duties, the training must also prepare him to conduct successfully the special experiments assigned to his mission. For certain of these tasks, the pilot becomes a laboratory experimenter and must be suitably trained. So far, many different training modes have been used to good advantage. These modes include lectures by specialists, discussions with the associated scientists, familiarization sessions with the specialized flight equipment before the flight, and parallel study in the field of the experiment. During the project, the special training given the astronauts produced trained experimenters for each mission.

Launch Preparation

In the process of hardware checkout during launch preparations, it has been found essential to have detailed written test and validation procedures, procedures that are validated and followed to the most minute detail during the preliminary systems checkout and, again, during later and final systems and integrated systems checkouts. It is necessary for the procedures to be so written that even small anomalies become readily apparent to those persons involved in the checkout. These persons must be so trained and indoctrinated that they are always watchful for anomalies which would be direct or indirect indications that the hardware may be approaching failure. Checkouts are not completed at the end of the detailed procedures, for it has been found that the data accumulated during a checkout procedure may reveal, upon detailed analyses, further symptoms that all is not well within a system. Finally, the Mercury personnel have developed and adhered to a philosophy that is believed to be a basic reason for Mercury's operational success. This philosophy is that Mercury launchings will not take place in the face of known troubles or in the face of unresolved doubts of any magnitude that could possibly affect mission success or flight safety. It is believed that adherence to this philosophy is of utmost importance to success of any manned space flight program.

Areas for Improvement

A list of those general technical areas that appeared to be either the source of, or a major contributing factor to the problems that repeatedly cost the project time and money would include design requirements, qualification practices, definition of standards, tests and validation procedures, and configuration management. The conditions and effects described in these areas are not unique to this project, but represent those that generally exist in the aerospace field. Therefore, improvements in these areas would be beneficial in reducing the number of discrepancies that may potentially cause schedule delays and rising costs. Discussion of these areas will reveal that in most trouble areas careful and continuing attention to detail and quality assurance program were not as effective in the aerospace industry as necesary. It is believed that the need for improvements has become clear and that the changes for the space flight era are beginning to be made.

Design Requirements

Requirements and philosophies applied during the detail design phase have a profound and lasting effect on the overall performance of a project; therefore, some of the more significant shortcomings observed in the design phase are emphasized. Adequate design margins must be established and they must be adequate. An example where inadequate margins were detrimental is the weight-sensitive landing system. Experience with aircraft and spacecraft designs shows that weight continues to increase with time. In Mercury, this increase was significant; and although the rate tended to decrease with time, it was present throughout the duration of the project. The orbital weight of the spacecraft increased at an average rate about 5 pounds (0.2 percent) per week during 1959 and 1960; thereafter the increase averaged less than 2 pounds per week, even after a strong weight-control program had been initiated. The overall weight increase caused an extensive requalification of the landing system because the original design did not have sufficient growth margin. During the initial design phase careful consideration should be given to the use of redundancy. There are different forms of redundancy and the correct form must be chosen for the particular application to prevent degrading the overall reliability of the system. Because of the hazards of space flight and the lack of provisions for repairing or replacing equipment in flight, it was imperative in Mercury spacecraft that all critical functions have redundant modes. The redundancy was made less automatic, as man demonstrated the capability of applying the redundant function or providing the redundancy himself.

In the design of a spacecraft, consideration must be given to accessibility of components and assemblies. More than 3,000 equipment removals were made during the launch preparations on an early spacecraft; at least 1,000 removals were performed during preparations of the other production spacecraft. The majority of these removals occurred to permit access to a failed part. It is important that the design be such that a minimum number of other components have to be disturbed when it is necessary to replace or revalidate a component.

Since man first began making things, particularly with machines that could produce identical copies, he has found himself in the position where interchangeability is a combination of a blessing and a trap. Time and time again airplanes, automobiles, and other types of systems have had troubles and faults, because things that could be connected wrong have been connected wrong, regardless of printed instructions, colors, or common sense. Therefore, it is imperative that electrical connectors, mechanical components, and pneumatic and liquid connectors be so designed that they cannot physically be assembled in the wrong orientation or in the improper order. Experience shows clearly that this requirement cannot be overemphasized. Mismated or misconnected parts continued throughout the project to ruin components, give false indications of trouble, and result in improper functions that can cause test failure during the life of the project.

In the design of equipment for specific applications, consideration must be made for the shelf-life periods, including a margin for delays and extensions to the schedule. Occasionally in Mercury, these periods were not adequate and some equipment had to be replaced because the lifetime limit had been exceeded while still in storage.

Still another and often overlooked consideration is compatibility of materials. This may be related to the materials themselves, to the environment, or, in the case of manned vehicles, to the sensitivity of the man. In any event, care must be taken to see that only those materials properly approved for use in the vehicle are actually used. Time and money were expended in Mercury to rectify cases where improper materials were found in the systems because someone had failed to follow the approved materials list.

Qualification Practices

Complete and appropriate qualification of components, assemblies, subsystems, and systems is essential for reliable performance of space equipment. In the design of the Mercury spacecraft, allowances were made for the unknown environment of the planned manned space-flight missions, by conservatism in design, by redundancy of equipment in systems, and, most important, by component qualification testing through ranges of environmental conditions that were believed to exceed the real conditions. The exact conditions that the components and equipment would be subjected to during Mercury space flights, of course, was unknown prior to the time of the flights. Therefore, care was taken in selecting the qualification conditions because underqualification could result in inflight failures, and drastic overqualification could cause unnecessary delays and high costs in the program. The selected qualification conditions proved to represent the actual environment conditions very well. Some modifications to the specifications were made as the project progressed to make allowances for specific environments, such as local heating in equipment areas and system-induced electrical "glitches." Complete coverage of conditions is important, but not sufficient if the qualification is not also appropriate. During the MA–9 mission, equipment faults occurred late in the mission which resulted in the failure of the automatic control system and required Astronaut Cooper to make his retromaneuver and reentry manually. These faults, which occurred in the electrical circuitry interfaces of the automatic control system, were caused by the accumulation of moisture. The components that suffered these faults had passed the Mercury humidity and moisture qualification tests; however, detail investigation revealed that one inappropriate step had occurred. The qualification procedures were set up so that the equipment was functionally validated before the test; however, during exposure to humid air and moisture, it was not functionally operated because it was not convenient to do so in the test facility. While it was being prepared for the posttest validation, it was given an opportunity to do some drying. The obvious fault was that the equipment was not required to operate during the entire course of the test. Of course, the weightless condition could not be simulated in these or any other ground tests and it is quite likely that this omission also played a role in this flight failure.

To be complete, qualification test requirements must be selected to cover all possible normal and contingent conditions and to allow for the integrated efforts that show up when a complete system is operated.

One way the qualification of a complete system has been accomplished in the project is through the use of full-scale, simulated environment tests. A spacecraft was completely outfitted with flight equipment and instrumented and tested under environmental conditions to reproduce as closely as possible the normal and abnormal, but possible, flight conditions. From these tests, it was possible to determine the effects of modifications and to demonstrate the performance of the integrated system. Almost 1,000 hours of this type of testing was accomplished, compared with less than 60 hours of actual space flight during the entire project.

Definition of Standards

It has become very apparent that certain standards that have been used for years in the aircraft industry must be revised and tightened to make them satisfactory for application to aerospace equipment. Among these are shop practices; for example, those practices used in preparing electrical wiring must be reevaluated to assure that each step is accomplished in a manner that meets high-quality standards. Insulation stripping, soldering, crimping or welding, and cleaning processes must be accomplished without degrading the materials and in such a way that the quality of the work can be verified. Requirements must be made more rigorous and must be thoroughly understood by the people performing the operations, by their supervisors, and by the inspectors to insure continuing high quality work.

Some space equipment is designed to close tolerances which make it very sensitive to contamination in any form; therefore, it is imperative that steps be taken to assure that proper and consistent cleanliness standards are set up throughout the manufacturing, assembly, validation, and checkout phases. A number of these cleanliness standards exist at the present time. However, what is considered clean by one standard may be dirty when compared with "clean" by a similar appearing standard. Steps are now being taken in the industry to formulate logical and consistent standards and it is necessary to implement and to enforce these standards as soon as possible to prevent recurrence of the continual difficulty caused in this project by contamination that ruined metering orifices, check valves, pressure regulators, relief valves, reducers, compressors, and other mechanical equipment, as well as electrical and electronic equipment.

Test and Validation Procedures

Checkout, test, and verification procedures must be compatible with one another and with procedures serving the same function on similar equipment at different test sites. Numerous cases of anomalies, or suspected malfunctions, and failed equipment have been traced to improper or incompatible test procedures and test mediums or equipment. Also, it was found that careful attention to test techniques is essential; otherwise equipment can be damaged because connections are made improperly or dirt can be introduced into the equipment by the test equipment. It has been found that test techniques must be tightened, verified, analyzed, and written in detail to lessen the chance for inadvertent steps to ruin the operation or give false assurance.

Configuration Control

During the course of the project, considerable effort was expended by NASA and its contractors in maintaining an accurate definition of system configuration so that configuration management could be properly maintained. Much of this was manual effort that could not respond as rapidly to changes and interrogations as desired. At least 12 major documents, some of which were updated continually, some periodically, and some for each mission, were used to present the necessary information which was summarized for the desired definition. Component identification, which is essential to component traceability, also was often a tedious, time-consuming, and inaccurate process. To provide for adequate configuration control, it is important that vital information of systems, subsystems, and components be gathered at a central point. Then, provisions must be made to view this information from appropriate levels and directions so that accurate and responsive configuration management can be accomplished. Eventual incorporation of such a system on a national scale would provide a retrievable file to insure maximum use of technical experience and to lessen the chance of repeated errors.

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2. PROJECT SUPPORT FROM THE NASA

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Summary

This paper outlines the contributions that were made to the Mercury Project by NASA organizations other than the Manned Spacecraft Center. These contributions began several years before the Mercury Project had official status through the basic research of the National Advisory Committee for Aeronautics which showed such a project to be feasible. The assistance provided by these organizations contributed directly to the timely development of the Mercury spacecraft and its systems, two of the three launch vehicles used in the Mercury program, and the Mercury Tracking Network.

Introduction

The efforts that were recently ended with the successful completion of the Mercury program did not begin with the initiation of the Mercury Project in late 1958 but, in reality, began several years before that date. The research conducted in the wind tunnels and other facilities of the National Advisory Committee for Aeronautics (NACA) in a decade preceding the Mercury Project established the concepts that eventually led to the Mercury Project. None of these original concepts needed to be changed during the Mercury program.

It is well known that the NACA provided its personnel and its facilities as a nucleus for the new agency when the NASA was established in October 1958. Almost immediately, a small group of scientists and engineers was organized at the Langley Research Center in Virginia to formulate plans for the Mercury Project. Many of this group were personnel of the Langley and Lewis Research Centers who had contributed to the original concepts of a man-in-space project in the preceding years. This organization became the Space Task Group (STG) and quickly began growing in size and capability. While the Space Task Group, and later the Manned Spacecraft Center (MSC), provided the direction and management of the Mercury Project, many thousands of scientists, engineers, technicians, and administrators throughout the NASA organization provided vital support for the Mercury Project. Without this support, Mercury could not have accomplished its goals within the time and costs that were realized.

It is appropriate to recognize that Langley Research Center is mentioned most frequently throughout this paper. The close association between Mercury and Langley is attributed to the fact that many of the original Space Task Group were personnel from the Langley Research Center and to the equally important fact that the STG and the MSC were physically located within the Langley Research Center for over 3½ years.

In addition to the formal technical support discussed in the following sections, administrative support was provided in the fields such as procurement, personnel, and security, by Langley in the initial phase of STG. The Launch Operations Center provided similar administrative support to the Mercury Field Office at Cape Canaveral.

Spacecraft Development

After a contract was awarded for the Mercury spacecraft, some 16 months passed before the contractor delivered the first production spacecraft. In order that full-scale tests could be conducted in the meanwhile, a large number of research and development spacecraft were constructed by NASA. These test articles were largely made of steel plate and, hence, have been called "boilerplates."

The boilerplates, which were made cheaply and quickly, resembled the Mercury spacecraft only in external configuration, in weight, and in center-of-gravity location. They were used primarily to obtain data on the performance of Mercury rocket motors and parachutes, and to obtain aerodynamic and thermal data needed for the design of the Mercury spacecraft.

In September, 1959, one of these boilerplates was flown through a ballistic flight by using the first Mercury-Atlas launch vehicle. This test, called Big Joe, was flown to gather thermodynamic data during reentry. This boilerplate was constructed in phases by both the Langley and the Lewis Research Centers. The Langley Research Center also provided the parachute landing system for the boilerplate and the Lewis Research Center designed and furnished the instrumentation and telemetry system. This successful flight test, in which the Langley and Lewis Centers played so large a part, provided valuable design data for the Mercury spacecraft.

The Langley Research Center also designed and constructed a series of boilerplates which were used in the Little Joe series of flights flown at Wallops Station, Va., in 1959 and 1960. The Little Joe tests were flown to prove the concepts of the launch escape system for inflight aborts at critical conditions and to evaluate the performance of this system.

Similar boilerplates were used in the Mercury program in drop tests for parachute-system qualification and as astronaut egress trainers until a Mercury spacecraft became available for this purpose. Much of the environmental qualification of equipment carried on all these boilerplates was conducted at Langley.

The many wind tunnels of the Langley, Lewis, and Ames Research Centers were used to perform tests early in the Mercury program to define the configuration of the Mercury spacecraft. Some 28 different wind-tunnel facilities conducted 103 separate investigations and accumulated over 5,300 hours of tunnel time by the end of 1960. These tests measured static and dynamic stability, pressure distributions, and heat-transfer data through subsonic, transonic, and supersonic speed regimes. Certain tests were made for vibration and flutter characteristics, and others to determine the correct size of the drogue parachute for stabilization. The Mercury escape and reentry configurations were tested alone and in combination with all of the launch vehicles in the Mercury program. Additional tests were made at Langley on alternate escape configurations, on the structural characteristic of the Mercury shingles, and on Mercury heat-shield materials. Langley also assisted in the data reduction and analysis of tests run outside of NASA, such as the buffet study made in a wind-tunnel at the Air Force Arnold Engineering Development Center.

Tests were conducted at Wallops Station, Va., early in the program to evaluate the escape system planned for the Mercury spacecraft. These tests used both boilerplate and production spacecraft with the production escape and landing systems. The first such tests were "off-the-pad" aborts. These tests were followed by inflight aborts from the Little Joe launch vehicle. Wallops supported these tests with radar tracking, optical tracking, photography, telemetry reception, data playback, and radio command functions. This support was in addition to providing normal launch and range-clearance support and shop and office facilities.

During the development of the propulsion systems for the Mercury spacecraft, special tests were conducted in a high-altitude wind tunnel at the Lewis Research Center to evaluate the performance of the escape rocket and retrorocket motors. The popgun effect of firing the posigrade rocket motors into the Mercury-Atlas adapter cavity between the spacecraft and the launch vehicle was measured. In addition, the effect of the escape rocket exhaust on the Mercury spacecraft window was evaluated.

Lewis also conducted developmental tests on the hydrogen peroxide reaction control system and on the manual proportional control system in the altitude chamber.

The Langley Research Center conducted a series of tests on the solenoid valves for the reaction control system thrusters. These tests were conducted in altitude chambers to determine the effect of vacuum on the valve. The results of the tests established that a vacuum did not affect the operation of a valve even when it was not operated for 24 hours. A method of evaluating the movement of the solenoid valve's seat by measuring the electric current flow (signature) was developed for these tests. This method of measuring the valve's signature was later used for selecting valves that were acceptable for flight.

The development of the spacecraft landing system required an extensive series of tests which began at Langley Research Center in 1958. In the early development of the main parachute, drops were made at West Point and Wallops Island, Va., and at Pope Air Force Base, N.C. Langley supported these tests with personnel, aircraft, test vehicles, instrumentation, and tracking equipment. Later tests were made at the NASA Flight Research Center at Edwards Air Force Base, Calif., to develop the Mercury drogue parachute. For these tests, the Flight Research Center provided personnel, test vehicles, and all other facilities needed to accomplish the program. The development of the landing-impact skirt required the assistance of NASA facilities at Langley Research Center and Wallops Station.

In the development of the Mercury heat protection system, the Langley Research Center made numerous structural tests at elevated temperatures on samples of the ablation heat shield, the René 41 conical shingles, and the beryllium recovery-section shingles.

When a formal program was established by Manned Spacecraft Center to conduct special inflight experiments on Mercury flights which were not directly related to the mission objectives, other NASA organizations proposed and furnished many of the experiments that were performed. On all the manned orbital flights, the Goddard Space Flight Center and the NASA Headquarters Office of Space Sciences sponsored experiments related to astronomy and earth and space science in general. These organizations also provided assistance in the evaluation of all proposed experiments. Goddard provided special filters and other optional equipment used in making some of these space-science observations.

The flashing-beacon experiment flown on the MA–9 flight was designed, constructed, and qualified by the Langley Research Center. Langley also provided the balloon-drag experiments flown on MA–7 and MA–9. The Lewis Research Center proposed and furnished the zero-gravity experiment carried on the MA–7 spacecraft. On the MA–8 flight, a number of ablation materials were bonded to the recovery-section shingles to evaluate them for heat-protection on future spacecraft. Langley not only furnished two of these materials, but conducted many tests on samples of the coated shingles to assure a good bond and no degradation of the safety aspects of the MA–8 mission.

Launch-Vehicle Development

The NASA centers were involved in the procurement and operation of two of the three launch vehicles used in the Mercury program—the Little Joe and the Redstone. The Little Joe was conceived early in 1958 by the same group at Langley that formulated the man-in-space program. This launch vehicle performed much of the qualification of the Mercury spacecraft at approximately one-sixth the cost of an Atlas. Shortly after the official start of Project Mercury, the Space Task Group requested Langley to accept the responsibility for the procurement of six flight vehicles and one test article. Accepting this responsibility, Langley performed the basic design of the vehicle, wrote the specification, evaluated contractors' proposals, and awarded and monitored the contract for detail design, construction, and testing. After delivery of the Little Joe vehicles, Langley provided personnel for the assembly, checkout, and launch of these vehicles at Wallops Station, Va. A command destruct system was also designed and provided by Langley for the first four Little Joe vehicles. In addition, Langley designed and constructed the spacecraft-launch-vehicle adapters for all Little Joe flights.

The Marshall Space Flight Center was instrumental in implementing the Mercury-Redstone program. Marshall's task was the provision of a launch vehicle for manned flight that had previously been used only for unmanned payloads of considerably lighter weight. Technical groups were formed to conduct studies and perform reliability and structural tests. As a result of these studies, a number of modifications were made in the Redstone launch vehicle to make it acceptable for manned flight. Major modifications, made largely at Marshall, were made in some subsystems, and an Abort Sensing and Implementation System (ASIS) was designed for and integrated into the launch vehicle. Other work done at Marshall included compatibility testing of the spacecraft-launch-vehicle combination and static firing of each launch vehicle prior to delivery to Cape Canaveral. The resulting launch-vehicle reliability was a milestone in the Mercury program that contributed to the reduced requirement for only five Redstone flights instead of the eight originally programed.

Prelaunch checkout and launch operations for the Mercury-Redstone missions were conducted by the NASA Launch Operations Center at Cape Canaveral which was formerly the Launch Operations Division of the Marshall Space Flight Center. The Launch Operations Center now provides much support to the Manned Spacecraft Center at Cape Canaveral in many technical and administrative areas and in the provision of facilities.

Mercury Network Development

Of considerable importance in the successful accomplishment of the Mercury missions was, of course, the worldwide Mercury Tracking and Communications Network. The responsibility for the development of this network was given to the Langley Research Center. A group formed at Langley in early 1959 wrote the specifications for the network and awarded a contract for its design and construction in July 1959. After the contract award, this Langley group continuously monitored and contributed to the design and development of the network facilities. The nerve center of the Mercury network is the automatic, high-speed computing equipment located at and operated by the Goddard Space Flight Center. Langley's responsibility for the network ended with the acceptance of the facilities by the government. Thereafter, the maintenance and operation of the Mercury network became the total responsibility of the Goddard Space Flight Center.

I

SPACE-VEHICLE DEVELOPMENT

3. SPACECRAFT SYSTEMS DEVELOPMENT AND PERFORMANCE

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Summary

Project Mercury began in 1958 with some basic systems research and a number of feasibility studies to determine if a spacecraft could be built which would sustain man in orbital space and return him safely to earth. Although it was recognized that some system development would be required, many of the spacecraft systems could be synthesized from existing hardware. A top priority was placed on the spacecraft production from the contract award in 1959, and 3 years later Astronaut John H. Glenn, Jr., completed three orbital passes about the earth. In this time span, design, development, and qualification of the spacecraft and its systems were accomplished nearly concurrently. The ground and flight-test programs, which included hundreds of wind-tunnel tests and parachute drops from aircraft, provided an opportunity to develop flight systems and acquire operational experience as the program progressed. Though a continuing attention to engineering detail by technical specialists and management personnel throughout the project, the spacecraft and its systems were qualified for suborbital flight in approximately 2 years from the spacecraft contract award date. Many lessons have been learned which were not only applied to Mercury systems development, but which have been applied in more advanced space projects. Interesting conclusions regarding system performance can be derived by reviewing all of the flight results. The spacecraft control system was a source of considerable trouble during the project. However, when inflight failures of this type occurred, it was the backup capability of the pilot which made possible the successful completion of the mission. In fact, the pilot's ability to control accurately the spacecraft attitude was instrumental in three of the four manned orbital flights in completing the mission successfully when a malfunction was present in the automatic system. One of these control-system malfunctions, an electrical anomaly during Astronaut Cooper's mission and the only one of major significance in the spacecraft throughout the entire 34-hour flight, was successfully circumvented by the pilot's manual control during the critical retrofire and reentry maneuvers.

Introduction

The initial goal of Project Mercury was to place a man into orbit successfully and return him safely to earth, and this objective was fulfilled in February 1962 by the flight of Astronaut John H. Glenn, Jr. This objective was confirmed 3 months later by the flight of Astronaut M. Scott Carpenter. The final two missions in Mercury constituted a continuation of a program to acquire new knowledge and operational experience in manned orbital space flight. The ninth Mercury-Atlas mission (MA–9) was planned for up to 22 orbital passes and was the concluding flight in the United States' first manned space program. The primary objectives of the MA–9 mission were to evaluate the effects on the astronaut of approximately 1 day in orbital flight, to verify that man can function as a primary operating system of the spacecraft; and to evaluate the combined performance of the astronaut and the spacecraft, which was specifically modified for the 1-day mission.

The MA–9 spacecraft, *Faith 7*, used by Astronaut Cooper in successfully performing the fourth United States manned orbital mission was basically similar to those used for previous orbital flights. The major exceptions were system modifications prompted by the extended nature of the mission, and these changes will be discussed in later paragraphs. It is important to note, however, that since the original design of the Mercury spacecraft all major system concepts have remained essentially unaltered. Although some design and early development were conducted prior to the official award of the prime contract, the Mercury spacecraft was developed, qualified, and met its original objective of manned orbital flight 3 years after the spacecraft contract award in 1959. In this brief span of time, many lessons have been learned and much experience has been gained in the design, development, and operation of manned orbital flight systems. In this paper, the intent is to describe brie.ly the original design philosophy, discuss the system development and qualification experiences, and present a summary of the experiences relating to systems performance.

Design Philosophy

In the initial design of the Mercury spacecraft, two guidelines were firmly established: (1) to use existing technology and off-the-shelf equipment wherever practical and (2) to follow the simplest and most reliable approach to system design. These guidelines were administered to provide for the most expedient realization of program objectives. The original Mercury concept also included a number of mandatory design requirements which were imposed on the spacecraft contractor:

(1) The spacecraft must be fitted with a reliable launch-escape system which would rapidly separate the spacecraft with its crew from the launch vehicle in case of an imminent disaster.

(2) The mode of reentry into the earth's atmosphere would be by drag braking only.

(3) The spacecraft must carry a retrorocket system capable of providing the necessary impulse to bring the vehicle out of orbit.

(4) The spacecraft design should place prime emphasis on the water-landing approach.

(5) The pilot must be given the capability of manually controlling spacecraft attitude.

In many design areas, there existed no previous experience in reliable system operation which could be applied to the Mercury concept, and new development programs had to be initiated. In addition, there was no information pertaining to man's capability to operate under space environmental conditions, particularly weightlessness; therefore, all of the spacecraft systems which relate to crew recovery from orbit had to be designed for automatic operation and many had to include redundancy. It has since been learned that man is not only a contributory element but a necessary part of the spacecraft. It is important to note that because of the pilot's demonstrated ability to function as a primary operating system of the spacecraft, some of the redundant elements were not required and were deleted.

The spacecraft systems (fig. 3–1) include the heat protection, mechanical and pyrotechnic spacecraft control, communications, instrumentation, life support, and electrical and sequential systems. The mechanical and pyrotechnic system group comprises the separation devices, the rocket motors, the landing system, and the internal spacecraft structure. These systems have been described in previous literature (refs. 1 to 10); therefore, detailed descriptions are not included in this paper.

The design requirements stated earlier involved certain implications for these systems. The launch-escape system was found to be most practical if it incorporated a solid rocket motor to propel the spacecraft rapidly away from the launch vehicle during an abort in the atmosphere. This type of system needed to provide a high level of thrust for a brief time period should be easily handled in the field and should require a minimum of servicing. The tower arrangement could be readily assembled to the spacecraft and jettisoned during powered flight once it no longer was required for abort.

An important design feature of the Mercury spacecraft was the favorable manner in which the astronaut was exposed to flight accelerations. For all major g-loads, which occur during powered flight, launch-escape motor thrusting, posigrade motor thrusting, retrograde motor thrusting, reentry, parachute deployment and touchdown, the pilot experienced acceleration in the most favorable manner, one that forces him into the couch (fig. 3–2).

The mode of reentry was specified to be drag braking only because of simplicity. This concept implied that the configuration should be a blunt body with high drag properties having a slender afterbody, primary because of heating considerations. Thus the bell shaped Mercury configuration was evolved, and the heat-protection system was devised to accommodate this shape. Originally, a beryllium thermal shield employing the heat-sink principle was specified. The specification was later changed to provide a more efficient ablation-type heat shield, which was used on all Mercury–Atlas orbital missions. Because the heat flux was expected to be considerably less on the afterbody than at the heat shield, a combination of insulation and thin shingles constructed of an alloy to withstand high temperature was calculated to be sufficient in maintaining the temperature of the pressure vessel at a safe level. The exterior finish of the spacecraft body was intentionally made a dull black because of its high emissivity and, therefore, favorable thermal radiation properties.

Again, because of their reliability and ease of handling and servicing, solid propellants were chosen for the retrorocket system. For even greater reliability, however, a system of three solid rocket motors, any two of which would effect a safe reentry, was chosen. These three rocket motors, together with three additional rockets to effect spacecraft-launch-vehicle separation, were assembled in a jettisonable package to permit a clean reentry configuration.

For the period during and after touchdown, the spacecraft had to meet two basic requirements. These requirements were: (a) the structure had not only to retain its integrity such that it would be habitable after landing and (b) the touchdown decelerations had to be reduced to an acceptable level. Touchdown deceleration was primarily limited by the human tolerance to acceleration; and, because of the blunt shape of the spacecraft, touchdown decelerations of as high as 50g could have resulted even for a water landing. Therefore, a landing-shock attenuation system was designed which consisted of a fiberglass fabric bag with holes in it and was attached between the spacecraft structure and the ablation shield. Prior to landing, the ablation shield would be deployed and the shield weight would extend the bag, which would fill with air and provide a cushion against the landing shock. The landing bag arrangement adequately attenuated the landing deceleration loads to approximately 15g.

In addition to the automatic and rate control modes of the attitude control system, two manual control modes, one electrical and the other mechanical, were provided the astronaut. This control-mode arrangement had the feature that,

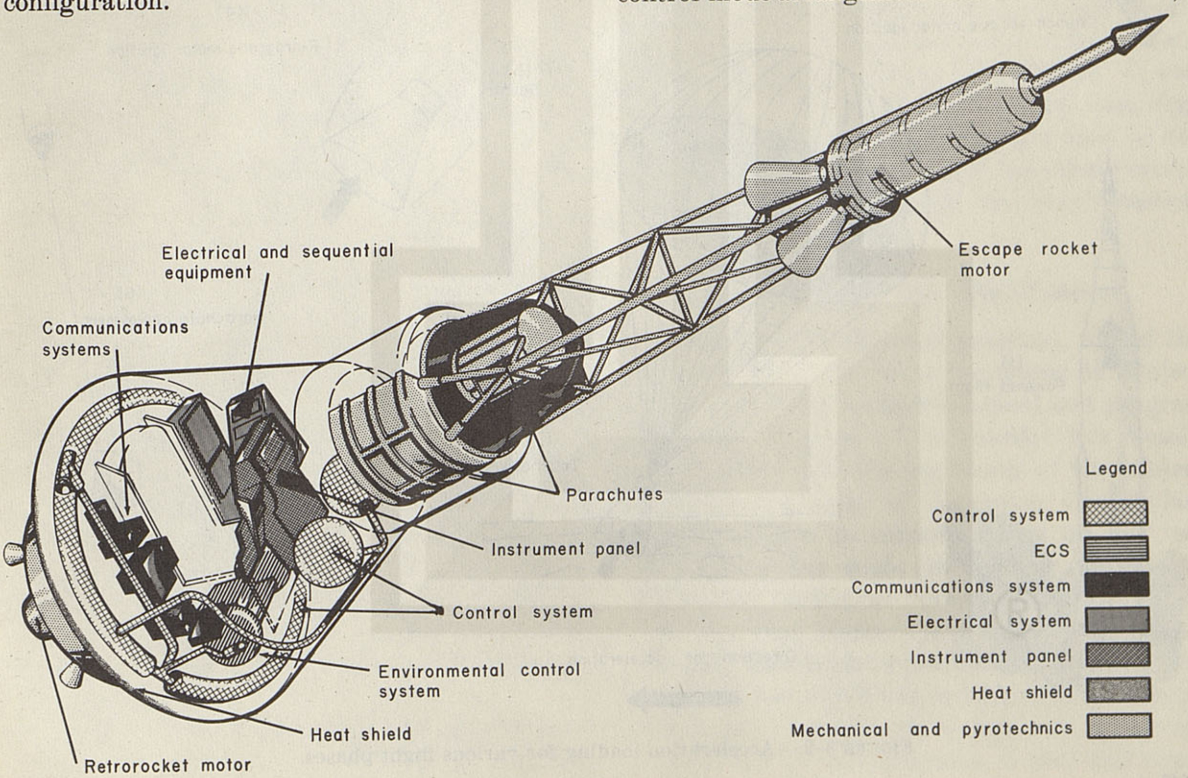

FIGURE 3–1.—Spacecraft interior arrangement.

in the event of a spacecraft power failure, the direct-linkage mechanical mode would still be available for control. The two manual control modes were each supplied control-system fuel from separate tanks for additional reliability. Although the thrust units were designed to provide an impulse sufficient for the majority of spacecraft maneuvers, these redundant manual control modes could be used simultaneously, if desired, in critical situations, such as retrofire and reentry, where rapid response to undesirable attitude rates might become necessary.

A monopropellant reaction control system using hydrogen peroxide as the fuel was chosen for the spacecraft control system to provide the simplest system design and installation. Furthermore, similar systems had already been developed for use on other space vehicles. A flexible bladder under pressure provided a positive means of fuel expulsion.

Many challenging design problems were encountered in the remaining spacecraft systems because of the new operating environment. As a result of the need to provide flight-control support on the ground, the requirement for multiple redundancy and high reliability in the communications system was evident. Although part of the instrumentation system was not required for flight safety and mission success, certain parameters, such as those which indicate the psysiological well-being of the crew and the proper operation of critical spacecraft systems, were necessary for effective flight control and monitoring. The remainder of the instrumentation data was acquired to complement the flight-control parameters for use in postflight

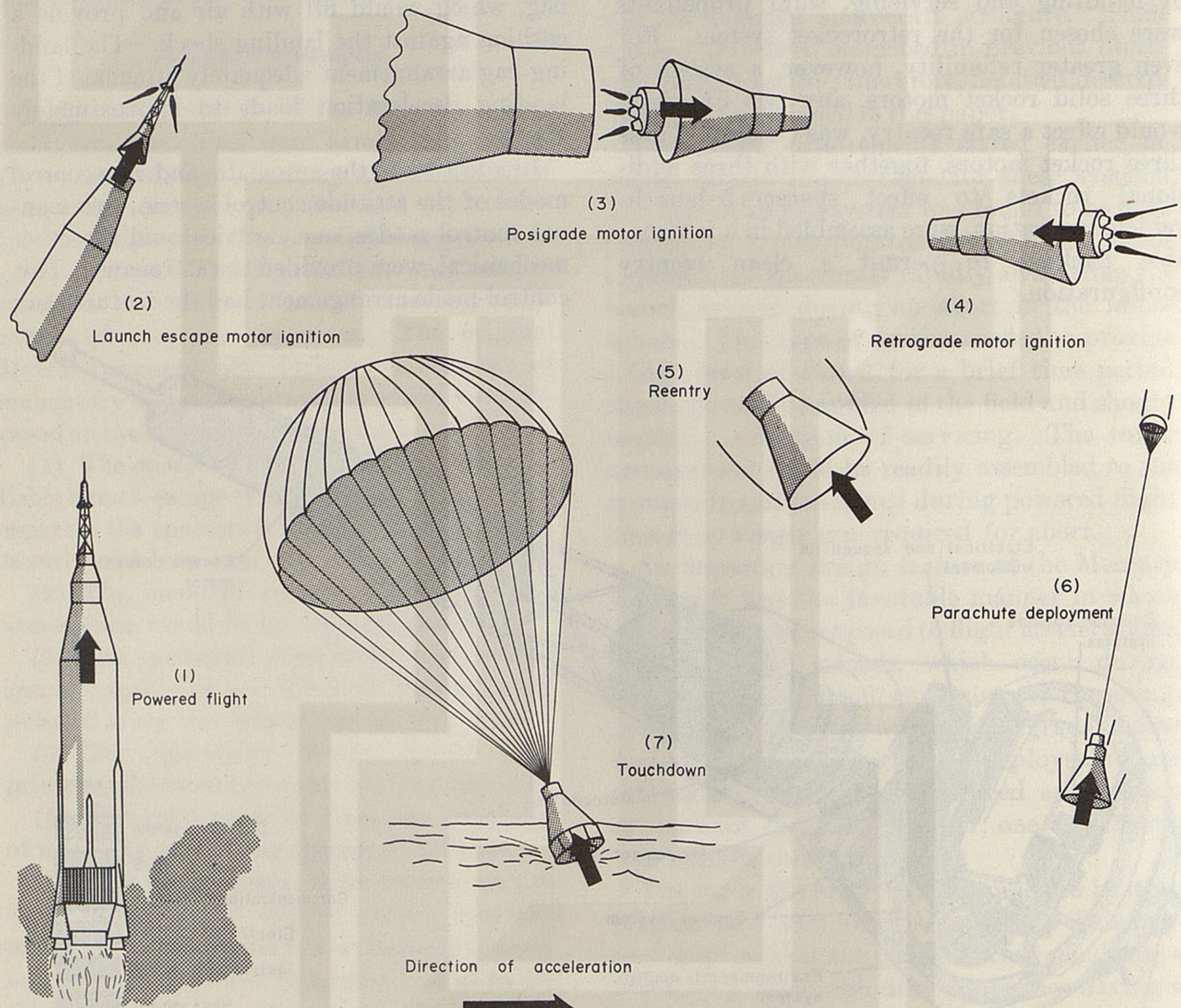

FIGURE 3-2.—Acceleration loading for various flight phases.

analyses of system performance. New design areas were opened up in the fields of gas partial pressure measurement and of bioinstrumentation, such as long term attachment of human sensor leads. The life-support-system design considerations involved a development task, since it was concerned with the sustenance of the astronaut and his protection from the hard vacuum of space, as well as from the widely varying temperature conditions associated with an orbital-flight profile. This system also was required to provide for the management of the cooling and drinking water in the spacecraft, the food to be consumed by the pilot, and his normal liquid wastes, again in the weightless environment. Although pressure suits and cooling equipment had been used in high-performance aircraft, only part of this experience could be directly applied to the design of the Mercury environmental control system because of weightless flight and more demanding performance requirements. In the electrical and sequential design area, the application of previous design work and use of off-the-shelf components was made. But the very nature of the mission and the requirement for reliability, automation, and system redundancy imposed a degree of complexity somewhat greater than any previous manned flight system. This increased electronic complexity, in turn, made it more difficult to insure interface compatability, eliminate stray voltages (back-door circuits), and minimize system sensitivity to current transients.

As an example of the consequences of stray voltages, the Little Joe–1 mission, the first launch attempt using a full-scale Mercury spacecraft, is cited. This test, conducted at Wallops Island, Va., was in the final moments of countdown when, during a spacecraft battery charging operation, a stray voltage initiated the launch escape sequence. The spacecraft was separated by the escape motor from the launch vehicle, and the drogue parachute was properly deployed. Because the battery had been only partially charged, sufficient current was not available to deploy the main parachute, and the spacecraft was destroyed upon landing. This back-door circuit was subsequently located and eliminated.

Because of work conducted immediately prior to and in the early period following contract award, the system-design phase of the project proceeded at a rapid pace. Wind-tunnel research, studies by prospective subcontractors and vendors, the joint participation of key NASA and other government installations, and early design studies by the eventual prime contractor all helped to facilitate the design effort and make possible the early availability of test hardware.

Based on the total Mercury experience, one of the underlying principles during the initial design period should be an emphasis placed on "designing for operation." For example, one of the lessons learned was that the instrumentation system should be designed with mission flexibility as a guide, such that, in the later phases of the program, new instrumentation requirements can be handled with a minimum of complication. In still another area, it was learned that component accessibility can be extremely important where schedule demands become critical. Certain time-critical systems and short-life components must be easily accessible in order to minimize the degree of disturbance to other systems and the time required to replace these types of units. Because of the weight and volume constraints, this concept could not faithfully be applied in the design evolution of the Mercury spacecraft, and significant penalties have been experienced whenever items needed to be removed under a tight schedule. It was learned in Mercury that all systems requiring manual operation by the astronaut must be designed with the limitations of the cabin volume (see fig. 3–3), suit mobility, and weightlessness in mind.

Development and Qualification

As in any development program, one of the original ground rules at the outset of Project Mercury was to conduct a logical and progressive test program. This concept was closely maintained from the beginning of the project through the flight of Astronaut Cooper last May. Success in certain phases of this test progression has made possible the elimination of certain backup or follow-on flights. Since the time that Mercury was initially conceived, literally thousands of individual tests have been conducted in which test articles were used in all forms from components to full-scale spacecraft and under all combinations of real and simulated operating conditions. For example, during the 1-year period from November 1959, about 10 months after the prime contract was awarded, to November 1960, some 270 hours were spent in testing the environmental control system in the altitude chamber, with a man wearing a pressure suit in the chamber to load the circuit more realistically. Early in 1961, further tests were conducted, often using astronauts, in the centrifuge to qualify the environmental system under acceleration loads.

For convenience, the spacecraft-system testing can be grouped into ground tests and flight tests of special test articles. The ground tests, in turn, can be categorized into areas of research, design, development, qualification, acceptance, and checkout. The discussion of development flight tests, which will be restricted to those using other than production spacecraft, consists of research studies, development tests, and qualification programs. The performance of the production spacecraft will be discussed in a later section of this paper. It is interesting to note that because of the rapid pace dictated by the high priority of the program, many of the individual test programs were conducted concurrently. This technique involved some risk, since, had a major problem developed, the expense in both time and money could have been considerable. The following paragraphs relate the more salient lessons learned during the formal Mercury development and qualification test program.

Ground Testing

The research tests included those which attempted to verify design theories or sought new methods for accomplishing a given design task, whether it was a structural assembly, a heat-protection system, or improved methods of instrumenting the spacecraft and its crew. Hundreds of tests of this type, particularly those conducted in the wind tunnel, were car-

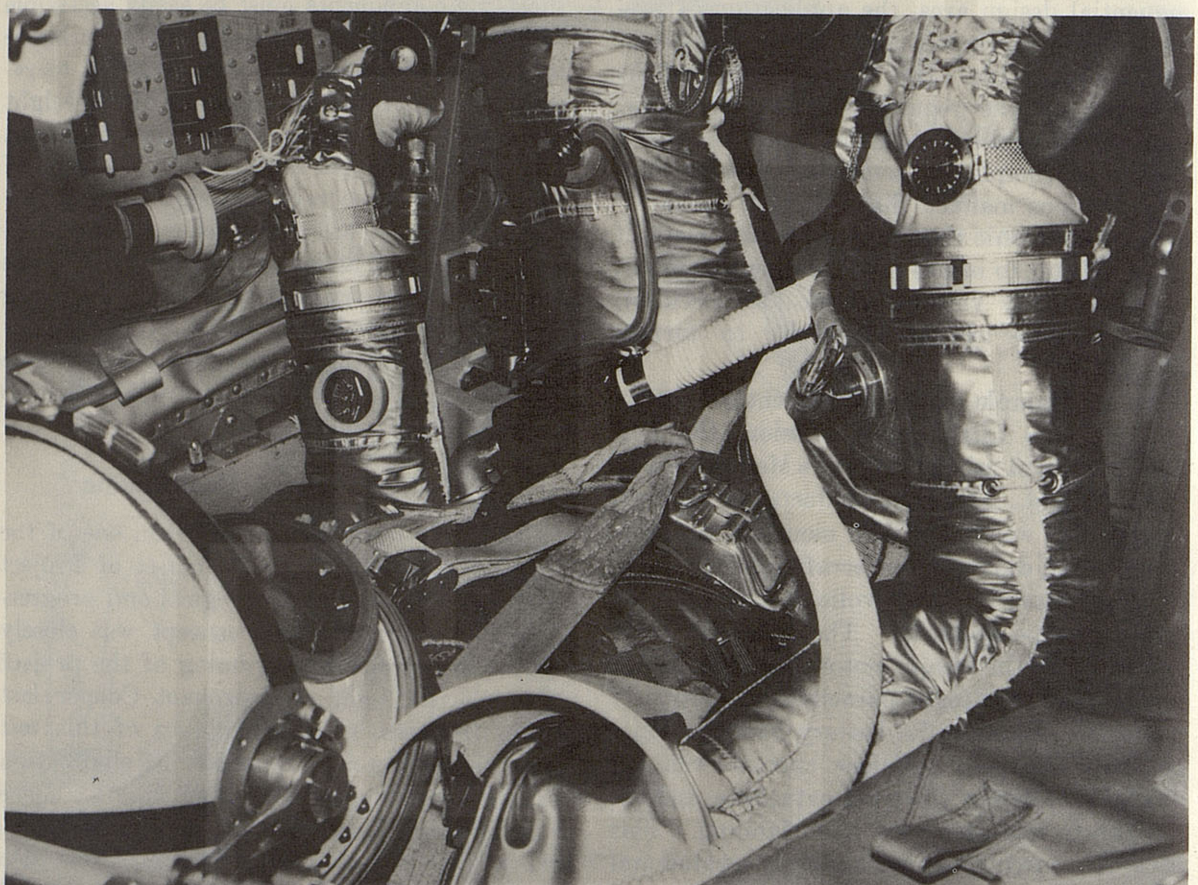

FIGURE 3–3.—Photograph of spacecraft interior.

ried out in the early phases of the Mercury effort at many of the NASA centers and at the contractor's plant. These tests will always be required when a new flight spectrum in a relatively unknown operational environment is penetrated, as it was in Mercury. It was tests of this kind which established the basic Mercury configuration, a shape which has already been projected into more advanced manned space programs.

The design testing, exemplified by the breadboard layouts in the case of electrical and sequential circuitry, was conducted jointly by the NASA and the contractor. This effort made possible the proof testing concurrent with initial design studies. Many thousands of tests were conducted, such as those in the design of the spacecraft-control-system thrust chambers, once the initial concept had been established.

When the basic design concept had evolved to a working hardware item, development testing served to expose this concept in the laboratory to the many combinations of operational and environmental conditions expected in space. Development testing was naturally hampered by the fact that weightlessness, a prime example, could not be adequately simulated on the ground; and this very deficiency resulted in an ineffective design for the water separation device of the environmental control system. The development of Mercury systems was a continuing program through the final mission and was aimed at mission flexibility, even after the spacecraft had been basically qualified for manned orbital operation. It was during the development testing that facets of the design which pertain to all aspects of its use were most evident, including the design-for-operation standards. It is in this testing area that engineering mock-ups have proved to be extremely valuable. In the case of the landing system, drop tests of boilerplate spacecraft were made to develop the landing-system deployment sequence and operation. Tests were made in the altitude chamber to verify that systems could operate for their required life cycle under realistic conditions. In essence, the development-test phase provided a means of validating the design concept and proving its intended reliability features.

Qualification testing conducted on the ground can further impose realistic operational conditions on a test article in various combinations for the specific purpose of verifying its reliable operation for inclusion as a final flight article. That is, there can and should be more than one type of qualification program for a given component, subsystem, or system, but these programs should become progressively more demanding on the capability of the hardware. In this testing area, adherence to prescribed test criteria must be rigorously enforced. The various combinations of qualification tests can be grouped into environmental tests, load tests, and performance tests with each of these groups having a specific purpose. Sometimes, the test conditions are not realistic enough or are not sufficiently demanding to reveal system weaknesses. During Mercury, for some of the subsystems, it was not until the actual unmanned flights that a system could be fully qualified for manned operation. For example, the launch-escape tower was subjected to all expected environmental conditions, an exhaustive series of load tests, and the operational situations associated with the launch-escape-system performance tests. Yet in the actual qualification flight program the heating loads on the truss structure of the tower were found to be more critical than had been calculated. Ground qualification is relatively inexpensive compared with full-scale flight qualification, and any system discrepancies which can be revealed in this phase will yield rewards in terms of time and expenditures later on. For example, during an early qualification test, it was found that the original igniters in the retrorocket motors would sometimes fail and blow out through the rocket nozzle before the main propellant grain had been ignited. New igniters, actually miniature solid rockets, were substituted for the original igniters. Had this system characteristic been overlooked through the manned orbital flights, the consequences could have been catastrophic. For flight-acceptance tests on units scheduled to be installed in flight vehicles, however, it was found that care should be taken not to over-test the article to the point at which its useful lifetime is approached or exceeded. During qualification testing, one must be assured that the unit being tested is not a "handmade" article and that, later on, a similar production version will not fail because it does not have the same ability to withstand the testing environment. Of course, a critical requirement for the qualification program is that the test conditions imposed on the hardware exceed those expected to be present in the design environment in order to provide a safe margin for manufacturing deviations and unanticipated design weaknesses. It was found in Mercury that no single qualification criterion necessarily applies to all systems, and local operating conditions must be evaluated specifically for each system to insure that they are adequately accounted for in the qualification test environment.

It was learned in Mercury that, whenever a significant design change is to be incorporated into the spacecraft, a new hardware qualification program should be initiated to requalify major systems. Approximately 1,000 hours of test time were accumulated on a full-scale spacecraft in a program called "Project Orbit" which was conducted in a vacuum-thermal facility to insure that, during the orbital flight program, systems would maintain their previously demonstrated performance. As an example, when the spacecraft thruster assemblies were modified as discussed in this paper, the modified assemblies were tested in a vacuum chamber as part of the Project Orbit testing. These tests, using hydrogen peroxide, were made to determine if exposure to combined temperatures and low pressures for the expected duration of the mission would have adverse effects on the operation of the thruster assemblies. It was found to be most effective if actual operating conditions and procedures, including preflight checkout tests, could be realistically simulated in order to expose hardware to a complete operating cycle. Since system qualification and operating reliability are so closely related, the reader is referred to the paper entitled "Reliability and Flight Safety" for additional details.

Finally, the acceptance and checkout tests which are conducted by using actual flight hardware involve the same recommendations previously mentioned, those of avoiding overtesting, realistic operational test conditions, and thoroughness. It was learned in Mercury that, if tests of this type are conducted at multiple stations across the country by separate groups, the test procedures must be consistent if the test results are to be comparable. It is essential to repeat a system checkout if the system has been disturbed for any reason, such as the removal of another system where a definite interface exists. The acceptance and checkout aspect of ground testing is more thoroughly discussed in the paper entitled "Spacecraft Preflight Preparation."

Flight Testing

This brief discussion of the development flight phase of Mercury will be limited to those flights where specially configured test vehicle (boilerplate spacecraft) were employed. Because the experiences gained by flights of production spacecraft are of more operational significance, they will be presented in the next section, Systems Performance. The flight-test program began with a number of tests in which spacecraft models were flown by using small multistage rockets. These tests provided preliminary data on the aerodynamic properties of the chosen external configuration. Almost concurrently with these flights, tests of the parachute systems were staged in which boilerplate spacecraft were dropped from cargo aircraft. These "drop tests" were initiated as an important step in the early design and development of the landing system. Specifically, the drogue parachute was developed by utilizing a weighted pod, which was dropped from an aircraft at high altitude. Other early flight tests included off-the-pad, or beach, aborts to develop the launch-escape system. In 1959, a reentry flight was conducted in which a specially designed and instrumented spacecraft and an Atlas launch vehicle were used to provide aerodynamic-heating data in the real flight spectrum. This flight, termed "Big Joe," was the first test in Mercury in which the Atlas was used. It was as a result of the data derived during this flight that the shingles initially on the spacecraft cylindrical section were replaced with somewhat thicker shingles made of beryllium to provide for more effective heat protection. The final series of early flight tests used the solid-propellant Little Joe vehicle (shown in fig. 3–4) to test the launch-escape system concept at critical inflight abort condition. Although most of the early flight tests were of a developmental nature, their missions served to qualify critical flight systems for later, more demanding flight tests. The intermediate series of aircraft drop tests, for instance, was completed to qualify the parachute and landing-shock attenuation systems. During this test phase in Mercury, valuable system improvements were incorporated at a minimum of cost and time.

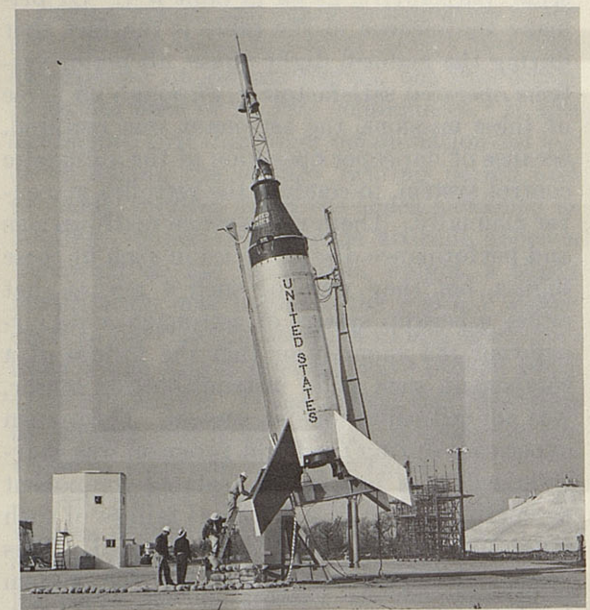

FIGURE 3-4.—Mercury Little-Joe launch-vehicle configuration.

Weight Growth

A critical problem which was present throughout the Mercury program was that of weight growth. This problem, which seems to be characteristic of any development program where high performance and reliability are required, almost defies the steps taken to control weight. Figure 3-5 depicts the weight chronology of the spacecraft's orbital configuration. The maximum growth in weight was approximately 10 pounds per week in the very early phases of the program, but this figure was reduced to less than 2 pounds per week, or approximately ½ percent, at the final stage of the program. The launch weight of Astronaut Cooper's spacecraft, *Faith 7*, was some 700 pounds greater than the original design weight, despite repeated design reviews and other continuing weight controls. The lesson here is that proper planning must account for the inevitable weight growth in the design and development of high-performance spacecraft, since the consequences of not planning for it are either a degradation of the performance goals or exceeding the capability of the launch vehicle with its attendant delays.

Attention to Detail

One of the most significant lessons learned from the Mercury program was the need for a careful and continuing attention to quality and engineering detail in all phases of the program. The spacecraft is made up of many individual systems and components to form a complex entity, and only through a close monitoring of the design and development of each piece of hardware and its relationship to all other associated components is it possible to recognize and correct problems rapidly before a costly failure occurs. Many performance discrepancies could not be anticipated because of the lack of experience or the inability to simulate adequately realistic conditions in the early test program. Later tests, however, were established to reveal these anomalies with a minimum of cost and delay. Although somewhat limited by the lack of experience, attention to detail during the design phase resulted in the incorporation of sys-

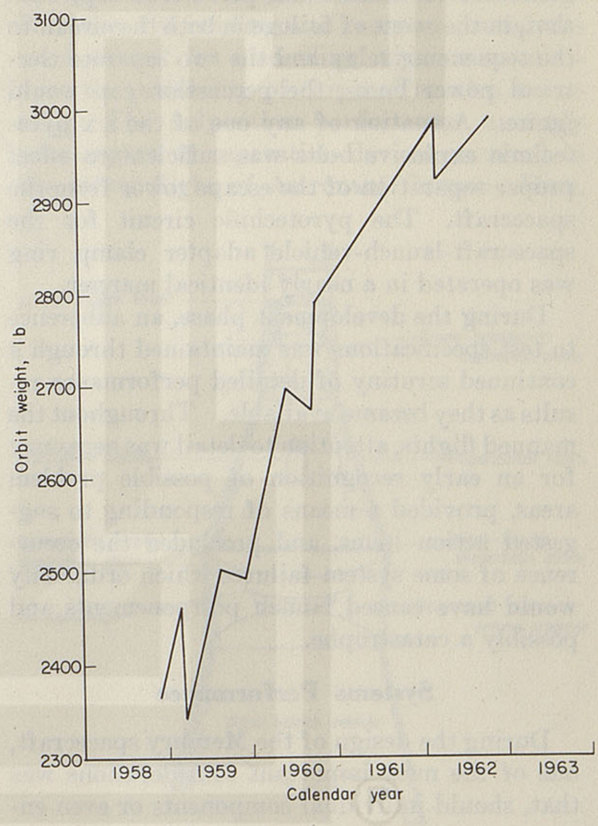

FIGURE 3-5.—Weight chronology for Mercury specification spacecraft.

tem redundancy, where a direct relationship to mission success existed.

As a prime example of the attention given to the incorporation of redundancy in the detailed design of critical spacecraft components, the actuation system of the launch-escape-tower clamp ring was backed up in nearly every component because of the serious consequences that would have resulted from a failure of the escape tower to jettison. In this system, the clamp ring is assembled at three points on its periphery, with each point being held by a dual explosive unit. Five of these six pyrotechnic units were ignited by an electrical squib, whereas the sixth was actuated by a percussion cap. Each of the electrical units incorporated a dual bridgewire. The automatic sequence was designed to send electrical signals from one power source to six of the bridgewires, with another but independent electrical supply for the remaining four bridgewires. Should the automatic relay fail, the astronaut was provided with a manual pull-ring which would energize the same jettison relay and also operate a gas generator to initiate the percussion cap, such that, in the event of failure in both the circuit to the sequencing relay and the two separate electrical power buses, the percussion cap would ignite. Actuation of any one of the six pyrotechnic explosive bolts was sufficient to effect proper separation of the escape tower from the spacecraft. The pyrotechnic circuit for the spacecraft-launch-vehicle adapter clamp ring was operated in a nearly identical manner.

During the development phase, an adherence to test specifications was maintained through a continued scrutiny of detailed performance results as they became available. Throughout the manned flights, attention to detail was necessary for an early recognition of possible problem areas, provided a means of responding to suggested action items, and precluded the occurrence of some system failures which ordinarily would have caused launch postponements and possibly a catastrophe.

Systems Performance

During the design of the Mercury spacecraft, one of the most important considerations was that, should individual components or even entire systems fail, some means would exist either to complete the mission safely or to conduct a successful mission abort so that crew safety would be maintained. A summary of the flight-program objectives and results for the full-scale spacecraft is given in table 3–I. Of primary significance in the table is the fact that during the manned flight phase, all major systems operated satisfactorily, although on three of these missions, the astronaut was required, because of improper operation of the automatic control system, to conduct the retrofire maneuver manually. There were system malfunctions and performance discrepancies in each of these flights, but they were of such a nature that either a backup system or astronaut could circumvent the anomaly or that the failure of a component, such as an instrumentation sensor, was not critical to mission success. The system experience during the flight program was characterized by a number of isolated component anomalies, rather than a critical failure of such magnitude that a catastrophe resulted. This system development, accounting for system malfunctions and performance discrepancies, the action taken to correct them, and the steps required to increase system capability for the extended flight of Astronaut Cooper, is discussed in the following paragraphs. Since system anomalies are discussed specifically as they pertain to the continuing development of the major spacecraft systems, references 5, 6, 8, and 10 should be consulted for a more detailed performance discussion. Although random failures and system deficiencies are mentioned briefly herein, the greater emphasis is placed on system performance as it relates to design experience and the lessons which can be derived from actual operation of the systems in the space environment. Throughout the flight program, with the exception of the MA–9 mission, no changes were required specifically to accommodate a longer flight duration. The modifications made to the *Faith 7* (MA–9) spacecraft including those incorporated to make possible the extended flight period are summarized in table 3–II. Each major spacecraft system will be discussed separately, as in previous reports on the individual manned flights (refs. 5, 8, and 10).

Heat Protection System

The heat protection system performed satisfactorily throughout the entire program and essentially as designed.

Some cracking and slight delamination of the ablation heat shield following reentry have been experienced on certain flights, but this occurrence has been of no real consequence. It was established that this minor delamination did not occur during the reentry heating period and probably resulted from the shock sustained at landing. Since the flotation attitude depends somewhat on the heat-shield weight, a slight modification was made to the *Faith 7* spacecraft to provide for retention of any small portions which might possibly have broken away after touchdown. It has always been desirable to achieve the most upright position in the water to facilitate astronaut egress.

Temperature measurements were made at various depths in the ablation shields for the orbital flights, and the maximum values experienced during reentry are summarized in figure 3–6 for each flight. The measurements showed good agreement with predicted values and were satisfactory.

Mechanical and Pyrotechnic Systems

The mechanical and pyrotechnic systems consist of the separation devices, the landing system, the rocket motors, and the internal spacecraft structure. Each of the systems in this group is discussed separately.

There have been only minor problems with the separation devices. The primary separation planes (shown in fig. 3–7) are those between the launch-escape tower and the spacecraft cylindrical section, between the spacecraft and the launch vehicle, at the heat shield, and at the spacecraft hatch. In three of the earlier unmanned qualification flights, some difficulty was experienced in separating the spacecraft-adapter umbilicals, but postflight examinations showed that the pyrotechnic charges ignited satisfactorily. Further investigation revealed, however, that aerodynamic loads during clamp-ring separation had caused the clamp-ring segments to damage the umbilicals. A minor redesign of the clamp-ring cover which protects these separation devices eliminated the problem. In the Mercury-Redstone 4 (MR–4) mission, the explosively actuated side hatch, incorporated for the first time for this flight, was prematurely released. The astronaut egressed rapidly through the open hatch, and the spacecraft subsequently took on sea water and sank before recovery could be effected. A postflight investigation involving a thorough analysis and exhaustive testing was conducted, but the cause of the malfunction has never been established. However, the landing and recovery procedures were altered for succeeding missions to minimize the possibility of this malfunction recurring. The only other performance anomaly with regard to separation devices occurred in the recent flight of Astronaut Cooper. Here,

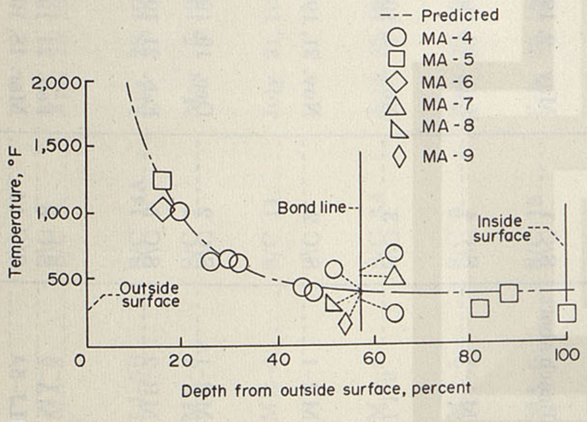

FIGURE 3–6.—Ablation shield maximum temperatures.

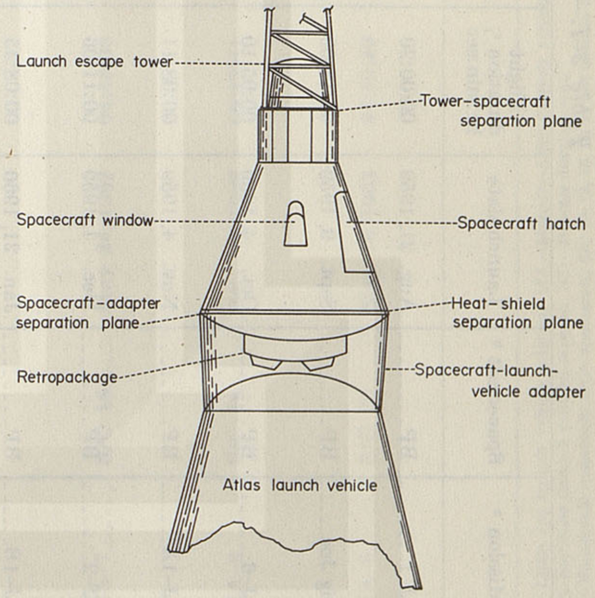

FIGURE 3–7.—Major spacecraft separation planes.

Table 3-I.—Mercury Flight Program Summary

Mission [a]	Spacecraft [b]	Launch date	Flight duration [c], hr:min:sec	Occupant	Basic test objectives [d]	Summary of results [e]
LJ-1	BP	Aug. 21, 1959	00:00:20		Max. dynamic pressure abort; evaluate launch escape and recovery systems.	Object. not met; inadvertent abort initiated during countdown.
Big Joe	BP	Sept. 9, 1959	00:13:00		Ballistic flight; evaluate heat-protection concept, aerodynamic shape, and recovery system.	Successful.
LJ-6	BP	Oct. 4, 1959	00:05:10		Ballistic flight; qualify launch-vehicle structure; evaluate command system.	Successful.
LJ-1A	BP	Nov. 4, 1959	00:08:11		Max. dynamic pressure abort; same as LJ-1.	Primary object. not met; escape motor ignition was late during reduced pressure region.
LJ-2	BP	Dec. 4, 1959	00:11:06	Rhesus monkey	High-altitude abort; evaluate launch, abort, and reentry dynamics on S/C; recovery.	Successful.
LJ-1B	BP	Jan. 21, 1960	00:08:35	Rhesus monkey	Max. dynamic pressure abort; same as LJ-1A; evaluate launch and abort.	Successful.
Beach abort	S/C 1	May 9, 1960	00:01:16		Off-the-pad abort; qualify structure and launch escape system for simulated pad abort.	Successful; expended rocket motor and tower not separated as quickly as expected.
MA-1	S/C 4	July 29, 1960	00:03:18		Ballistic flight; S/C-launch-vehicle compatibility; thermal loads in critical abort.	Object. not met; mission failed at about 60 sec after lift-off; S/C not recovered.
LJ-5	S/C 3	Nov. 8, 1960	00:02:22		Max. dynamic pressure abort; qualify launch escape system and structure.	Object. not met; S/C did not separate from launch vehicle.
MR-1	S/C 2	Nov. 21, 1960	00:00:00	Simulated man	Suborbital flight; qualify S/C-launch-vehicle compatibility, posigrades, ASCS.	Test object. not met; launch vehicle shutdown at lift-off; S/C landing system correctly deployed.
MR-1A	S/C 2	Dec. 19, 1960	00:15:45	Simulated man	Suborbital flight; same as MR-1	Successful; cutoff overspeed caused overshoot.
MR-2	S/C 5	Feb. 21, 1960	00:16:39	Chimpanzee	Suborbital flight; qualify ECS, landing bag.	Successful; launch vehicle failed to shutdown until fuel depletion, S/C overshot by 130 miles.
MA-2	S/C 6	Feb. 21, 1961	00:17:56		Ballistic flight; same as MA-1	Successful.
LJ-5A	S/C 14	Mar. 18, 1961	00:23:48		Max. dynamic pressure abort; same as LJ-5.	Object. not met; escape rocket ignited early; S/C recovered intact.

MR–BD	BP	Mar. 24, 1961	00:08:23	----------	Suborbital flight; evaluate modifications to correct MR–1 and MR–2 malfunctions.	Successful.
MA–3	S/C 8	Apr. 25, 1961	00:07:19	Simulated man	One-pass orbital flight; evaluate all S/C systems, network, recovery forces.	Object. not met; launch vehicle failed to follow roll program; S/C escape system operated.
LJ–5B	S/C 14A	Apr. 28, 1961	00:05:25	----------	Max. dynamic pressure abort; same as LJ–5 and LJ–5A.	Successful.
MR–3	S/C 7	May 5, 1961	00:15:22	Alan B. Shepard	Suborbital flight; familiarize man with space flight; evaluate response and S/C control.	Successful; first American astronaut in space.
MR–4	S/C 11	July 21, 1961	00:15:37	Virgil I. Grissom.	Suborbital flight; same as MR–3	Successful; premature hatch release caused S/C to take on water and sink; astronaut recovered.
MA–4	S/C 8A	Sept. 13, 1961	01:49:20	Simulated man	One-pass orbital flight; same as MA–3.	Successful; open circuit in control system caused S/C to land 75 miles uprange; S/C recovered.
MA–5	S/C 9	Nov. 29, 1961	03:20:59	Chimpanzee	Three-pass orbital flight; qualify all systems, network, for orbital flight recovery.	Successful; control system malfunction terminated flight after two passes.
MA–6	S/C 13	Feb. 20, 1962	04:55:23	John H. Glenn, Jr.	Three-pass orbital flight; evaluate effects on and performance of astronaut in space; astronaut's evaluation of S/C and support.	Successful; first American to orbit earth; control system malfunction required manual retrofire and reentry; erroneous T/M signal, retropack retained through reentry; S/C landed 40 miles uprange.
MA–7	S/C 18	May 24, 1962	04:56:05	M. Scott Carpenter.	Three-pass orbital flight; same as MA–6; evaluate S/C modifications and network.	Successful; horizon scanner circuit malfunction required manual retrofire; yaw error caused S/C to land 250 miles downrange, recovery in 3 hr.
MA–8	S/C 16	Oct. 3, 1962	09:13:11	Walter M. Schirra, Jr.	Six-pass orbital flight; same as MA–6 and MA–7 except for extended duration.	Successful; partially blocked ECS coolant valve delayed stabilizing suit temperature until 2nd pass; S/C landed 4½ miles from primary recovery ship.
MA–9	S/C 20	May 15, 1963	34:19:49	L. Gordon Cooper, Jr.	Twenty-two pass orbital flight; evaluate effects on man of up to 1 day in space; verify man as primary S/C system.	Successful; short circuit late in flight disabled ASCS, inverters, prompted manual retrofire and reentry; S/C landed 4½ miles from ship.

[a] LJ—Little Joe launch vehicle mission; MA—Mercury-Atlas (launch vehicle) mission; MR—Mercury-Redstone (launch vehicle) mission; BD—Booster development.

[b] BP—Boilerplate spacecraft; S/C—spacecraft; S/C 10, 12, 15, 17, 19 not used in flight program.

[c] Duration measured from lift-off to landing.

[d] ASCS—automatic stabilization and control systems; ECS—environmental control system.

[e] Object.—objectives of flight; prop.—propellant; T/M—telemetry.

Table 3-II.—Summary of Modifications to MA-9 Spacecraft

System	Modification	Justification
Spacecraft control system.	1. Removed rate control system (RSCS) 2. Added 15-pound-capacity fuel tank 3. Installed modified 1- and 6-pound thrust chambers 4. Installed interconnect valve	1. Not necessary; reduced weight by 12 lb 2. Additional control capability [a] 3. Improved reliability and operating characteristics 4. Improved control-fuel management
Communications systems.	1. Removed backup UHF voice transmitter 2. Installed slow-scan television unit	1. Primary unit reliable, reduced weight by 3 lb 2. Inflight evaluation of TV for ground monitoring of astronaut and instruments
Instrumentation system.	1. Deleted backup telemetry transmitter 2. Changed recorder speed from 1⅞ ips to $^{15}/_{16}$ ips and programed 3. Deleted periscope 4. Deleted low-level commutator	1. Primary unit reliable, reduced weight by 2 lb 2. Greater flight coverage necessary without changing recorder or reel size 3. Reduce weight by 76 lb; unnecessary for attitude reference 4. Served its purpose on previous flights
Life support systems	1. Added 4 lb of breathing oxygen 2. Installed parallel suit-coolant control valve 3. Added inline condensate trap 4. Added urine and condensate transfer systems with manual operation 5. Added 9 lb of cooling water 6. Added 4.5 lb of drinking water 7. Added 0.8 lb of CO^2 adsorber	1. Necessary for extended mission 2. Added reliability in case of partial valve blockage as experienced in MA-8 3. Existing condensate system believed ineffective 4. Increase urine and condensate storage capability because of extended mission 5. Increase cooling capability because of mission 6. Necessary for increased mission duration 7. Necessary for increased mission duration
Electrical and sequential systems.	1. Replaced two 1,500 watt-hour batteries with two 3,000 watt-hour units 2. Replaced two of three inverters	1. Necessary for extended flight duration 2. Improved thermal and operating properties

[a]Tank intentionally serviced to only 10 lb. of fuel.

four of the five umbilicals, two between the spacecraft and the adapter and three between the spacecraft and the retropackage (fig. 3–8) failed to separate in a normal manner. Later analysis revealed that each of the malfunctioned disconnects (see fig. 3–9), which normally contained a dual charge came from a special test lot which did not contain the main charge of explosive powder. Somehow, this lot had been improperly marked as intended for flight hardware. The umbilical which separated normally contained the intended amount of explosive and came from a properly identified lot. The four umbilicals which failed to separate pyrotechnically were released through actuation of a backup mechanical device. This experience points up the necessity for close control of flight articles and a means for establishing that the hardware intended for flight satisfies prescribed specifications.

The landing system, which includes the main, reserve, and drogue-stabilization parachutes and the landing-shock attenuation system (landing bag), has never failed in flight during the production-spacecraft flight program. In the second Mercury-Redstone mission, the heat shield was lost after landing because the metal retaining straps and landing-bag material to which the shield was attached failed as a result of wave action and strengthening of existing straps for later spacecraft eliminated this problem. The only other anomalies in the operation of the landing system were concerned with the altitude of parachute deployment, and these anomalies are discussed in the Electrical and Sequential Systems section. The successful performance of the landing system, particularly the parachutes, can be attributed to a thorough test program involving some 80 air drops of full-scale spacecraft.

Figure 3-8.—Spacecraft photograph displaying retro-rocket umbilicals.

The rocket motors include the launch-escape motor, the retrorockets, the posigrade rockets, and the launch-escape-tower jettison motor. All of the rocket motors used solid propellants, and their nominal thrust values are indicated in table 3-III. Each of these rocket systems has operated satisfactorily throughout the Mercury flight program. It was found early in the program that the launch-escape tower did not separate rapidly enough from the spacecraft after an off-the-pad abort test because of thrust impingement on the tower; therefore, the tower-jettison rocket-nozzle configuration was subsequently changed from a one- to a three-nozzle arrangement. Because of reliable launch-vehicle operation, the launch-escape system was never needed for an atmospheric abort during the manned flight program, and the large escape motor successfully ignited each time when the system was normally jettisoned. An abort, however, occurred during the unmanned MA-3 mission, and the system operated satisfactorily.

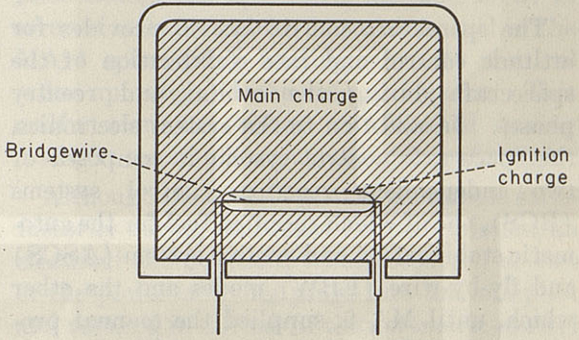

Figure 3-9.—Schematic diagram of explosive umbilical disconnects.

Table 3-III.—Nominal Rocket Motor Characteristics

Rocket motor	Number of motors	Nominal thrust each, lb	Approximate burning time each, sec
Escape	1	52, 000	1
Tower jettison	1	800	1. 5
Posigrade	3	400	1
Retrograde	3	1, 000	10

The internal spacecraft structure has been compromised only once during a mission critical situation, a record which is essentially proved by the fact that water, following an ocean landing, had never entered the spacecraft in appreciable amounts, except in one instance, because of a structural failure. In the MR-2 mission following landing recontact of the heat shield with the large pressure bulkhead caused puncturing that resulted in a sizable leakage rate.

The spacecraft was recovered, however, within a safe period. During postflight inspections of all manned spacecraft, some evidence of recontact by the heat shield upon landing has been present, but this damage to the large pressure bulkhead has been slight. The integrity of the spacecraft's load-carrying structure was especially proven during the Little Joe flight program. In one of these flights, the late ignition of one of the Little Joe rocket motors caused the trajectory to be considerably flattened, and as a result the spacecraft was exposed to loading conditions approximately twice those expected for a normal flight.

Spacecraft Control System

The spacecraft control system provides for attitude control and rate stabilization of the spacecraft during the orbital and reentry phases. In addition to the system electronics, the spacecraft control system is composed of two independent reaction control systems (RCS), one of which supplied fuel for the automatic stabilization and control system (ASCS) and fly-by-wire (FBW) modes and the other which, until MA–9, supplied the manual proportional (MP) and the rate stabilization and control system (RSCS) modes. The RSCS unit was installed in the MR–4 and subsequent flights as a backup to one of the secondary modes of the ASCS, that of auxiliary damping. This unit was removed as unnecessary for the MA–9 flight, with major deciding factors being its high fuel-consumption characteristics and weight. The FBW and MP modes were available for direct manual control by the astronaut, initially as backups to the ASCS and in the final two orbital flights as modes of equal priority. Although the control system has operated adequately in all of the manned flights, largely because of the ability of the pilot to exercise precise attitude control manually, this system has exhibited failures of one type or another in nearly every flight. The one exception was the six-pass mission of Astronaut Schirra, in which the system operated correctly.

The single most prevalent malfunction in the control system during the early manned flight program was the intermittent failure of the small 1-pound thrust-chamber assemblies (thrusters). In addition, during a manned suborbital flight (MR–3) a 6-pound thruster also failed to produce thrust when required. During the MA–5 flight, the mission duration was terminated early because of a failure in the thrust chamber assembly. During the flight of Astronaut Glenn, intermittent failures of the 1-pound pitch and yaw thrusters would have caused a similar early termination of the mission had the pilot not been present to exercise his manual control option. Immediately following the first inflight thruster failures, a complete analysis was begun to determine the exact cause of the system discrepancy. In the postflight inspections for the MR–3, MA–5, and MA–6 spacecraft, small particles were discovered at critical points in the thrust chamber assembly, and for the MA–5 mission a large metal deposit which partially blocked the thruster orifice was found. Although thruster malfunctions were experienced during the MA–4 flight, the postflight inspection did not reveal any thruster valve contamination. The exact mechanism for transporting these particles, some of which were found to be broken pieces from the stainless-steel dutch-weave screens which distributed the flow, to upstream points is still unknown. Three steps were taken for the MA–7 mission to correct this anomaly, one being the replacement of the dutch-weave screens with a combination of a stainless-steel fuel distribution plate and platinum screens, another being the reduction of the bore and size of the heat barrier, and the third being the relocation of the fuel-metering orifice to the upstream side of the solenoid valve (ref. 8). While these changes constituted the MA–7 modification, a more refined design change was being developed and qualified in the Project Orbit altitude chamber tests. This configuration, compared in figure 3–10 with previous 1-pound thruster configurations, involved both the 1- and 6-pound thrusters and was installed in the MA–9 spacecraft. No thruster failures of this type occurred on either the MA–7, MA–8, or MA–9 flights after the modifications had been successively incorporated.

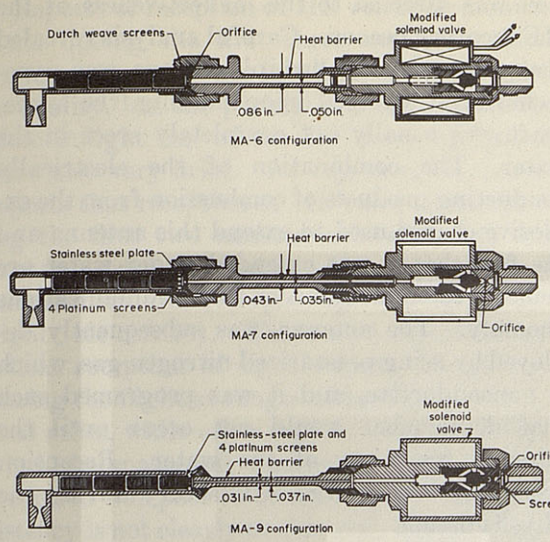

FIGURE 3-10.—Comparison of 1-pound thrust-chamber configurations.

The horizon scanners, which were used to provide an external reference for the attitude gyros, were a source of difficulty in the earlier orbital flights. In the MR-4 flight after tower jettisoning, the scanner was observed to be generating unexpected ignore signals, the cause of which was later traced to the impingement and heating effects caused by the ignition of the launch-escape rocket. A modification to the horizon-scanner cover eliminated this problem.

In the MA-4 flight, both scanners exhibited output variations which could not be correlated with attitude changes, and this anomaly was subsequently found to have been partially caused by "cold-cloud effects"; in addition, a shorted capacitor in the scanner circuit contributed to the attitude discrepancy. Since the scanner unit had been designed without accurately taking into account the effect of high-altitude cloud formations in the view field, a temporary modification of altering the bias levels was made for the MA-5 flight, but this change did not completely eliminate the problem. Further system refinement involving signal clipping for the earth portion of the view resulted in a successful modification for the first manned orbital flight. Since that time, only isolated occurrences of "cold-cloud effects" have been observed. During the MA-7 flight, a horizon-scanner circuit failure (see ref. 8) of another type occurred, but because the antenna canister was normally jettisoned prior to landing, it was impossible to conduct a postflight inspection of the hardware and determine the cause of the failure. This malfunction, which occurred in the pitch scanner, is believed to have been random in nature within the scanner circuitry.

The only remaining control system problem of any consequence during the full-scale flight program was the existence of an open circuit in the pitch-rate gyro input to the amplifier-calibrator (Amp-Cal), or autopilot, during the MA-4 mission. The Amp-Cal is the electronic unit which generates automatic control system logic for the various ASCS operating modes. The partial loss of gyro information to the autopilot caused the spacecraft attitude to be in error at retrofire, which in turn resulted in the MA-4 spacecraft's landing some 75 nautical miles up range of the intended point. This malfunction was either not detected during preflight tests or it occurred during the flight.

Although the control system performed satisfactorily during Astronaut Cooper's mission, an electrical short circuit, which occurred at two of the power-carrying plugs into the autopilot and resulted in the loss of the automatic control mode during the final few orbital passes. However, because this malfunction occurred at this specific interface and is primarily of an electrical nature, it is discussed in a later paragraph under Electrical and Sequential Systems. Because of the loss of the automatic control mode during the retrofire and reentry flight maneuvers, the astronaut conducted these maneuvers by using both manual modes available to him.

The only other major modifications to the control system for the 1-day mission of Astronaut Cooper were the addition of a 15-pound-capacity fuel tank, which is shown in figure 3-11, and the incorporation of the interconnect valve between the two RCS systems for better fuel utilization, in an emergency, and for more effective fuel jettisoning.

Communications Systems

The original design configuration of the communications systems proved to have been the most conservative of all of the major systems. These systems—the voice transceivers, the radar beacons, the location aids, and the command receivers—operated satisfactorily throughout the flight program. Because of the excellent performance of these systems, some of their backup units were deleted, including one of the two command receivers and decoders and the high-frequency (HF) recovery transceiver for the MA–8 and MA–9 flights and the ultra-high frequency (UHF) backup voice transceiver for the MA–9 flight. One of the two UHF telemetry transmitters, which were part of the instrumentation system, was also deleted as unnecessary for the MA–9 mission. A slow-scan television system, shown in figure 3–12, was included for evaluation aboard the *Faith 7* spacecraft, but the quality and usefulness of its transmissions were not satisfactory.

In the initial two manned orbital flights, it was noted that signals were not being received from the HF recovery transmitter, but because of the circumstances at the time of recovery and the uncertainty of HF reception in the landing area, it could not be established that an anomaly existed. However, when this discrepancy still existed on the MA–8 mission, attention was directed to the ineffectiveness of the HF recovery beacon. Careful analysis revealed that when the HF "whip" antenna was pyrotechnically deployed upon landing, the spacecraft was usually not completely erect in the water. The combination of the electrically conducting products of combustion from the explosive charge used to extend this antenna and the fact that it was extended under water are believed to be the cause of this communications anomaly. The antenna was subsequently deployed by using pressurized nitrogen gas, which is nonconductive, and it was programed such that deployment would not occur until the antenna was clear of the water. Reception from this beacon was satisfactory during the MA–9 mission.

For the MA–8 flight, a pair of more sensitive microphones was installed in the pilot's helmet, and the increased sensitivity apparently caused the background noise from the launch vehicle to trigger the voice-operated relay in the air-ground circuit. For the MA–9 mission, these microphones were modified to reduce background noise sensitivity such that this triggering action ceased.

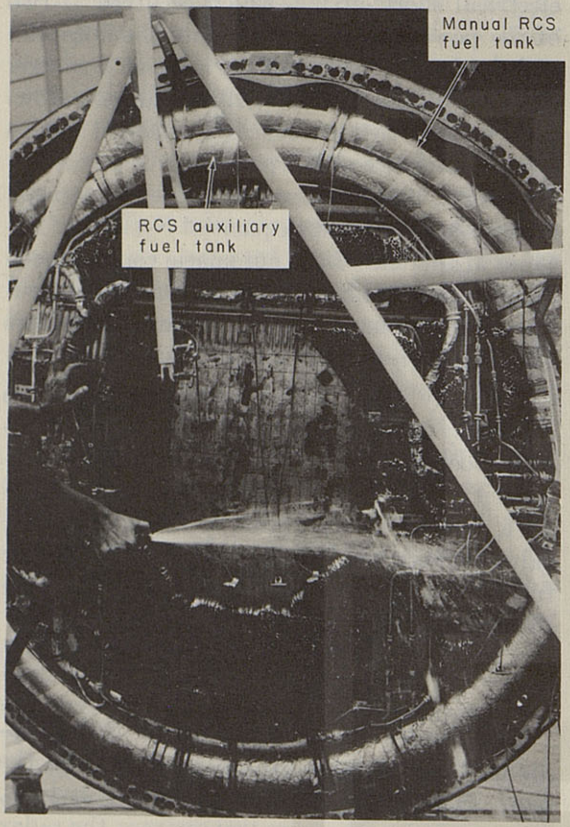

FIGURE 3–11.—Auxiliary reaction control system fuel tank.

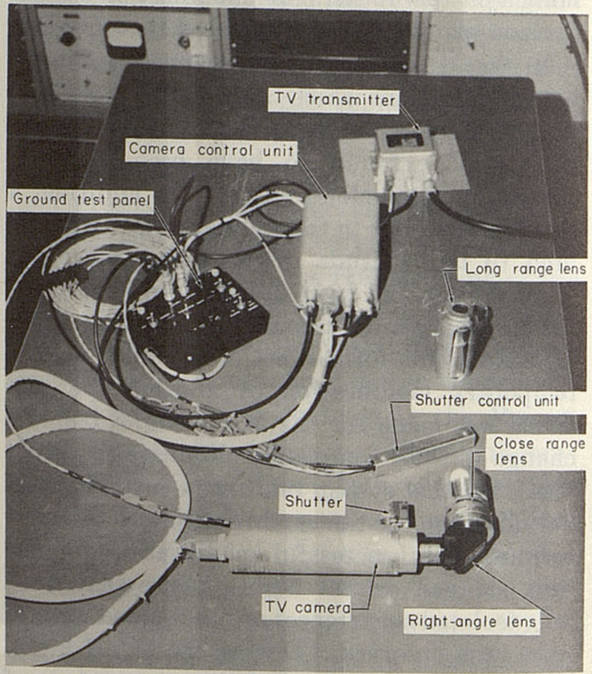

FIGURE 3–12.—Television system evaluated during MA–9.

Reports of reception of HF voice communications during the first three manned orbital flights were somewhat inconsistent with regard to quality, but the periods allowed for a complete inflight test of the HF voice equipment were also very brief. At any rate, because of reports that reception of HF voice signals during the first two manned orbital flights was unsatisfactory, a special HF antenna was installed on the retropackage for the MA–8 flight (see ref. 10). There were reports of excellent reception of signals from this antenna during the flight at ranges exceeding 2,000 nautical miles, while other reports stated that even when the spacecraft was nearly overhead, the reception was poor to unreadable. This inconsistency is not clearly understood, but the effects of spacecraft attitude at the time of transmission, the atmospheric propagation characteristics at the time of contact, and the status of operational ground equipment remain as unknown variables. A more closely controlled test of this special dipole antenna was conducted during the MA–9 flight, and it was fully successful. Although HF voice transmissions were heard during MA–8, the results of MA–9 were more consistent and indicated reliable operation. It might be mentioned that both the pilots and ground-control personnel preferred the UHF voice equipment to the HF system, particularly since none of the missions were such that nearly continuous communications were required. The UHF communications, of course, are limited to essentially line-of-sight ranges, but have signal-to-noise characteristics superior to those of HF in flight. However, the MA–9 astronaut found HF communications quite useful during the long periods in which he could not make UHF contact with a network station.

Although the command system has never been exercised for a commanded abort, its performance has been entirely satisfactory during other inflight exercises, such as the reception of signals for instrumentation calibration in all orbital flights and for an emergency voice communications test and a commanded wake-up tone in the MA–9 mission. For the unmanned orbital flights, MA–4 and MA–5, the command system was successfully used to control the operation of the spacecraft and bring it safely back from orbit.

Instrumentation System

The instrumentation system monitored over 100 performance variables and events throughout the spacecraft, and the operation of this system was satisfactory throughout the entire Mercury program. The system was designed with enough flexibility to incorporate required instrumentation changes as the program progressed. In the manned orbital flight phase, it was desired to have a more complete temperature survey at discrete spacecraft points, primarily on the spacecraft afterbody; and a low-level commutator circuit was installed. This unit was deleted from the MA–9 spacecraft as having served its purpose and to save weight. The confidence in the telemetry transmitters through the third manned orbital flight led to a decision to eliminate one of the two redundant units from the *Faith 7* spacecraft to save weight. The onboard recording capacity for the MA–9 flight was extended by changing the tape speed from $1\frac{7}{8}$ inches per second (ips) to $^{15}/_{16}$ ips and reprograming the operation periods such that only essential information was recorded during the expected 34-hour period.

Probably the most widely known system malfunction in the entire Mercury program is that associated with the failure of a limit switch which sensed heat-shield release. During the MA–6 mission, ground-control personnel received a telemetry signal which indicated that the heat shield had been prematurely unlatched from the spacecraft. Although it was believed that this signal was improper and involved an instrumentation failure, a decision was made to reenter with the retropackage attached to insure that the heat shield would not part from the spacecraft during the critical reentry heating period. A postflight examination of the instrumentation revealed that a limit switch had a bent and loose shaft (shown in fig. 3–13) and that manipulation of the sensor without appreciably displacing the sensing shaft would generate an erroneous signal. This experience prompted a change in the installation technique and a directive for tighter quality-control standards to insure that prescribed manufacturing tolerances would be maintained. This type of malfunction did not recur in subsequent flights.

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FIGURE 3-13.—MA-6 limit switch used to sense heat-shield release.

Early in the flight program, beginning with the Little Joe 5 mission, the mechanical spacecraft clock was found to be sensitive to accelerations in excess of 5g. An electronic digital clock was substituted for this unit and operated satisfactorily.

During the MA-7 mission, the blood-pressure measuring system (BPMS) yielded data which were of only marginal value. The system was thoroughly checked out following the flight, and no major system malfunction was found. It was shown, however, that proper techniques, including establishing a proper amplifier gain setting, correlation with clinically measured values, and the fitting of the pressure cuff to the individual flight astronaut, were not well understood. A thorough review of the entire system, its operating characteristics, and the preflight calibration procedures was conducted in the months after the MA-7 flight, and the data quality for the MA-8 and MA-9 missions was correspondingly improved and resulted in usable values. A discussion of this anomaly from a medical standpoint is presented in the Aeromedical Preparations paper.

During the MA-9 mission, the programer, which automatically controls the operation and sequence of events of certain spacecraft systems, exhibited two anomalies, one inherent and the other resulting from a structural failure. The inherent anomaly, evident to varying degrees in previous flights, involved a sensitive control circuit containing transistors which actuated power relays to operate the programer. This circuit was sensitive to certain input voltage transients which occasionally caused undesired programer operation. Prior to the MA-9 flight, a loading resistor had been added to reduce the inherent sensitivity, and an on-off switch had been incorporated so that the pilot could shut the system down if improper operation occurred. On two occasions, the unit was inadvertently triggered and continued to call for instrumentation calibrations, one of its programed functions. On both occasions, the astronaut turned the system off, and no serious consequences resulted, but the need to improve system design for future programs in this area, particularly for transistorized circuits, is exemplified.

The other programer anomaly, although in a separate section of the system, involved the shearing of a pin used to maintain alinement of a gear in the programer drive mechanism. Figure 3–14 depicts the misalined gear, which resulted in an inflight binding of the programer and the preclusion of a significant portion of recorded data during the midpoint of the MA–9 flight until the astronaut switched from programed to continuous operation.

During the MA–9 flight, the respiration rate sensor failed to yield reliable data during and after the fifth orbital pass, but other sources of this information were found to be adequate.

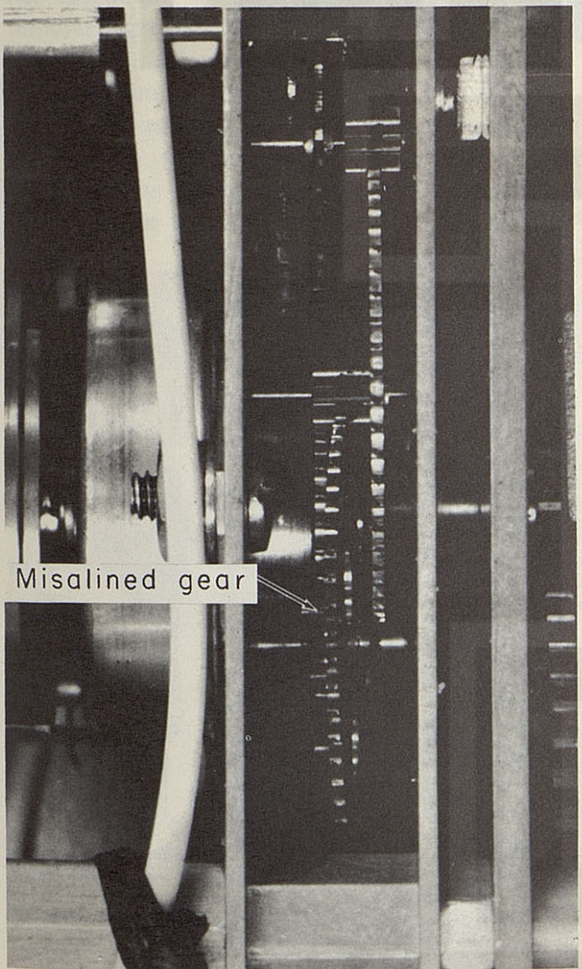


FIGURE 3–14.—Misalined gear in MA–9 programer.

A postflight investigation of the system disclosed a broken solder joint at the attachment point of the sensor lead.

Life-Support Systems

The life-support systems primarily provide for control of the cabin and suit atmospheres, management of metabolic-waste products, and the supply of food and liquid for the astronaut. The major changes to the MA–9 life-support systems, including the environmental control system (ECS) (fig. 3–15), from those of previous missions were accomplished primarily in support of the increased mission time, and the most significant modifications were as follows:

(1) Addition of about 4 pounds of primary breathing oxygen (O_2), stored under pressure, for a nominal total of 12 pounds in the system.

(2) Increase in the carbon-dioxide (CO_2) adsorber, lithium hydroxide (LiOH), quantity from 4.6 to 5.4 pounds. The amount of activated charcoal, as the odor absorber, was decreased from 1.0 to 0.2 pound, which was sufficient.

(3) Increase in the stored coolant-system water from 39 pounds to 48 pounds.

(4) Increase in the capability of the urine collection and storage system.

(5) Addition of an improved condensate collection and storage system, including a new wick-type condensate trap (shown in fig. 3–16) to extract free water from the suit circuit of the ECS.

(6) Increase of the stored drinking water by 4.5 pounds for a total of 10 pounds of potable water.

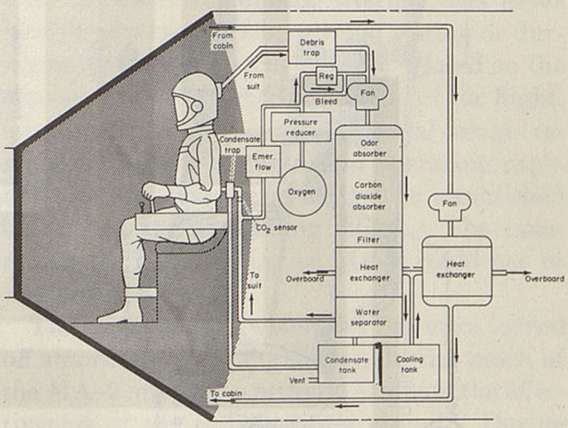
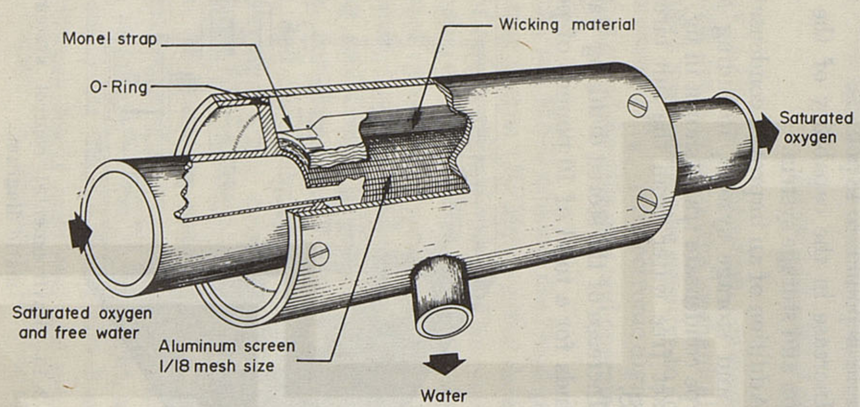
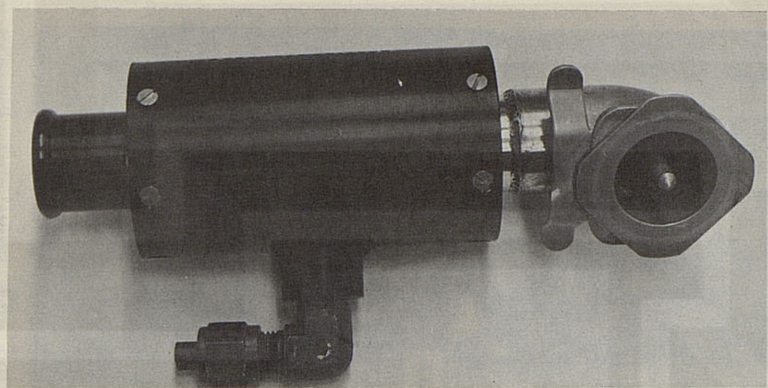


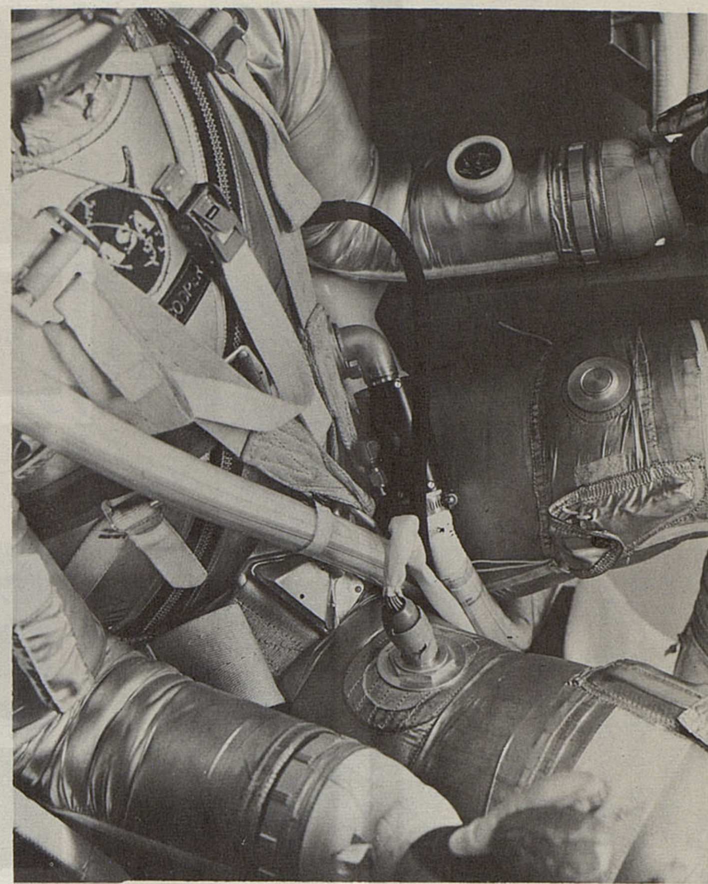
FIGURE 3–15.—Environmental control system schematic diagram.



(a) Condensate trap details



(b) Condensate trap



(c) Condensate trap installation

FIGURE 3-16.—MA-9 inline condensate trap.

A parallel coolant control valve (CCV) shown in the upper right corner of figure 3–17 was added in the suit cooling-water circuit for redundancy with the primary valve (top-left on the control plate) in the event of a serious valve blockage by contamination, which was experienced in the MA–8 mission.

The operation of the life-support equipment during the MA–8 mission was normal, except that the suit-circuit CCV was partially blocked by solidified lubricant and delayed the astronaut's stabilization of the cooling system at a comfortable level. Preflight procedures were changed for the MA–9 mission so that the CCV's were cleaned and properly lubricated prior to flight, but after the manned systems tests. The cooling water was also passed through a 0.15 micron filter before being transferred into the spacecraft. Blocking of the CCV during the MA–9 flight was not experienced. However, the astronaut was required to make a large number of minor changes to the suit CCV setting in an attempt to maintain the heat-exchanger dome temperature, which was the cooling system control parameter, within the desired range. No system deficiencies or hardware malfunctions were found during the postflight inspection or testing. It is a characteristic of

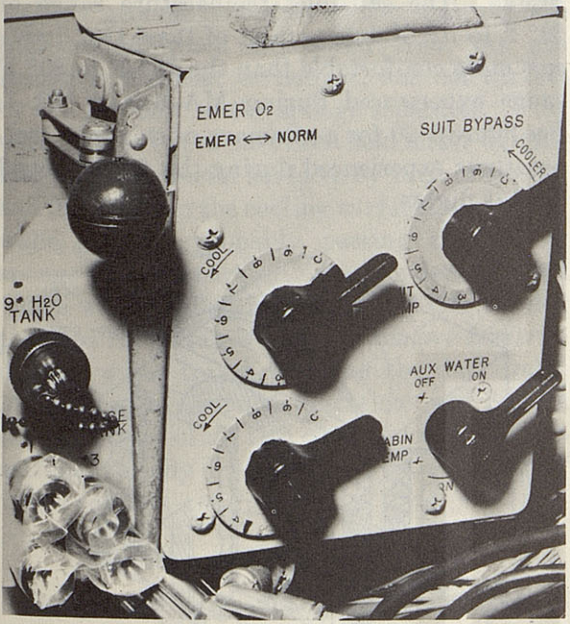


Figure 3–17.—Redundant coolant control valve for MA–9.

the system that changes in metabolic and external suit-circuit heat loads as a result of changes in the astronaut's level of activity, open visor operation, solar heat on the spacecraft, and internal spacecraft equipment heating will be experienced and will be reflected in the coolant requirements for the suit heat exchanger. These heat-load changes are not radical under normal conditions and the corresponding coolant flow changes would be small compared with the capacity of the CCV. It is quite possible that the sensitivity of this small-orifice valve, together with the astronaut's normally varying metabolic heat loads, could have resulted in the need for frequent coolant-flow adjustment.

An inline condensate trap, shown in figure 3–16, was designed to remove excess water from the suit-inlet hose and was installed near the entrance point on the suit. The condensate trap was activated periodically according to the flight plan by the astronaut's opening a hose clamp on the water outlet line from the trap. Condensate water was observed by the astronaut to have been flowing through this line, indicating that free water had probably passed around the sponge.

During the 21st orbital pass, the carbon dioxide (CO_2) level at the LiOH canister outlet began to show an increase on the CO_2 meter. Postflight chemical analysis of the canister showed definite channeling of the flow through the canister. Channeling is the localized or restricted passage of gas through the canister, rather than a uniform flow for maximum CO_2 adsorption. This channeling, which could reduce the effective canister lifetime, has never been experienced during ground testing or during any previous Mercury flight. Based on the amount of unused LiOH at the end of the flight, approximately 27 hours of normal usage remained. However, the actual operating capability of the canister could not be established because of the channeling effects. The exact reason for its occurring on MA–9 could not be established.

The cabin coolant water and fan were turned off according to the flight plan during much of the MA–9 mission in order to evaluate the effectiveness of the cabin cooling circuit. During