this time, the electrical load varied according to mission requirements, and the cabin temperature was observed to cycle between 85° F and 95° F, as indicated in figure 3–18. Reduction in the electrical load during this no-cooling period resulted in corresponding reduction in cabin temperature. It is concluded that cabin cooling was not required during periods in which the Mercury spacecraft electrical system was powered down.

Problems were encountered during MA–9 with the condensate transfer system. The needle of the hand-operated pump, used to transfer liquid from the condensate tank to another container, became clogged with metal shavings from the pump shaft and the condensate could not be transferred. Normally, free water removed by the condensate trap and sponge separator flowed directly to the condensate tank, from which it was then intended to be pumped to storage bags. The condensate tank contained a porous plus to relieve the gas pumped from the sponge into the tank by the action of the sponge separator. Since it was known that this plug could pass water when the tank became nearly filled, the astronaut elected to discontinue operation of the condensate trap when the transfer pump became clogged. This action was taken to stop further flow from the trap to the tank and thereby help to preclude water from being released into the cabin.

No malfunction of the life-support system which compromised the mission or presented a marginal condition to the man occurred during any of the manned Mercury missions. Although minor malfunctions of equipment occurred on these flights, some of which were alleviated by the astronaut, none of these were repeated on successive flights. The suit cooling system has exhibited a history of undesirable operation, characterized by elevated suit inlet temperatures, wet undergarments, and a general lack of astronaut comfort. However, metabolic heat loads were removed sufficiently to keep body temperatures well below a physiologically marginal value. The causes of these cooling system problems for the suit circuit were twofold:

(1) Selection of an improper cooling system control parameter during the initial design period.

(2) Ineffectiveness of the suit-cooling-circuit water separator because of the unpredicted behavior of free liquid in a weightless condition.

Ground testing showed that the steam exhaust duct temperature used in MA–6 and MA–7 missions was not an adequate control parameter for controlling the operation of the heat exchanger. A probe, which sensed the steam temperature at the heat-exchanger dome (see fig. 3–19) between the two coolant evaporating passes, provided a more rapidly responding indication of the heat-exchanger operation. This control temperature parameter was used during the MA–8 and MA–9 flights with satisfactory results. The suit-inlet temperature range of 60° F to 70° F during most of these two flights was more comfortable than the 75° F to 80° F range experienced during MA–6 and MA–7. See figure 3–20 for a summary of suit-inlet temperatures experienced during the four manned orbital flights.

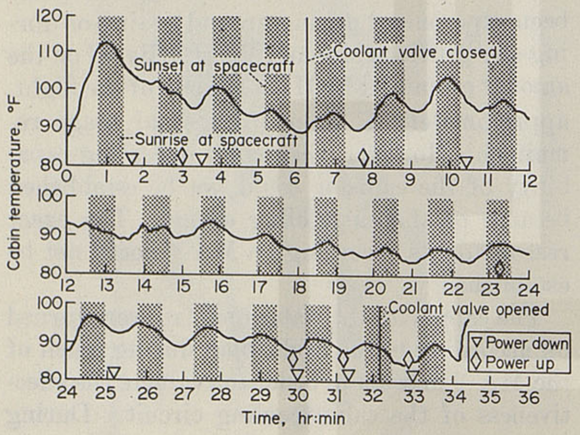


FIGURE 3–18.—Time history of MA–9 spacecraft cabin temperature.

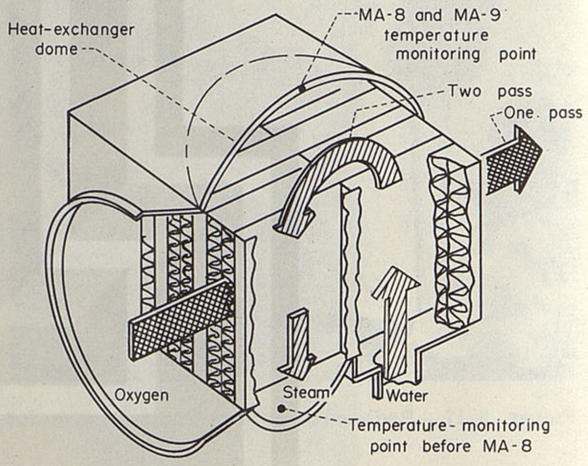


FIGURE 3–19.—Temperature monitoring points on heat exchangers.

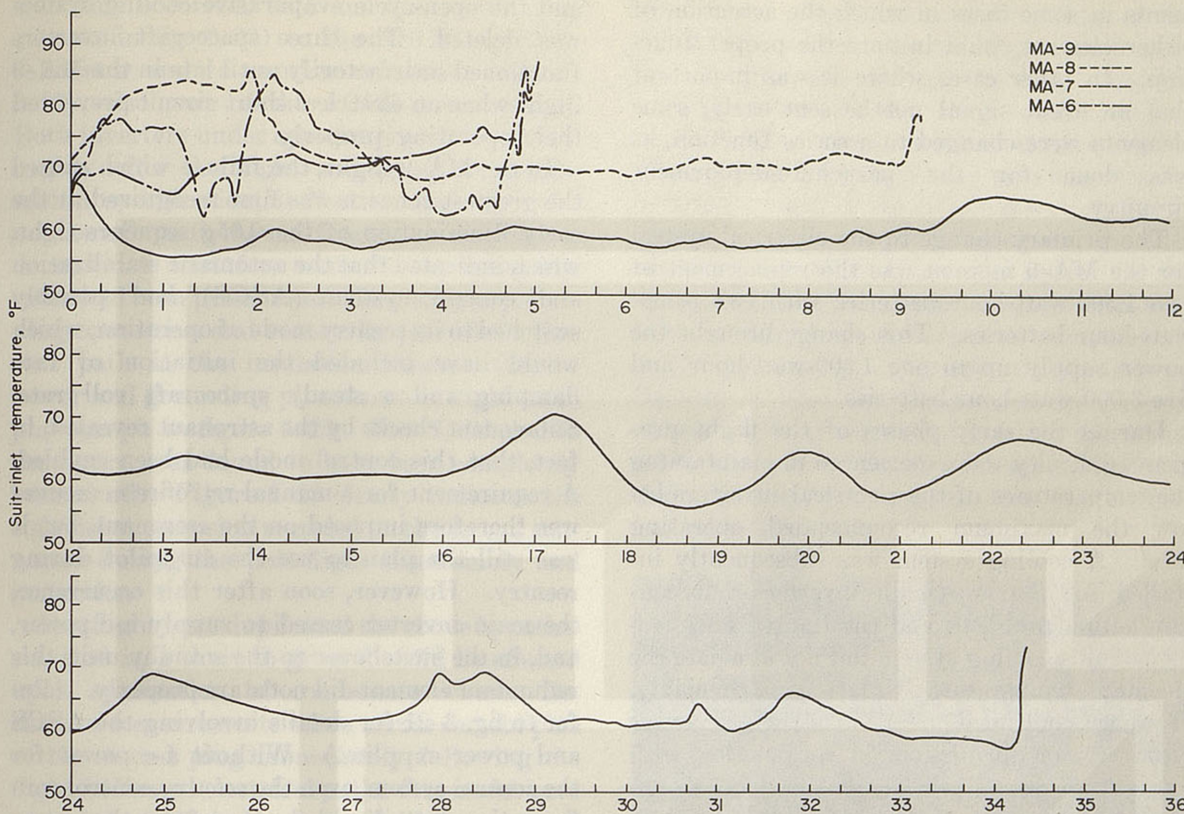


FIGURE 3-20.—Time history for suit-inlet temperature for manned orbital flights.

Other ground tests showed that water in the suit circuit, when condensed from the gas stream in the heat exchanger, was not carried by the gas flow to the sponge separator. This water is believed to have been held under weightlessness to the metal surfaces by surface tension and flowed from the cooling surfaces to the duct walls, thereby probably passing around the sponge in the separator. The condensate trap, which was installed in the MA-9 ECS, verified the need for a trap which will remove free condensate water traveling along the duct walls. Missions of even longer durations will require the extraction of all free condensate to keep the astronaut's body dry and thereby to obtain maximum comfort and hygiene.

Electrical and Sequential Systems

Except for some early development problems in the sequential system, this system group has performed satisfactorily throughout the Mercury program. Although there were no serious sequential problems throughout the manned flight program, there was an early deployment of the main parachute during the MR-4 mission and of the drogue parachute during MA-6. The reasons for these premature deployments have never been fully understood, since no system malfunction could be found during exhaustive postflight testing. During the later manned orbital missions, a modification to the sensing circuits for these sequential functions guarded against premature automatic deployment. The contractor was instructed to conduct a single-point failure analysis, which involved a detailed study of the electrical and sequential circuitry to establish all possible failure modes, and this analysis was conducted for all spacecraft systems before the MA-7 flight. The results of this study were evaluated for failure conditions that would singularly jeopardize flight safety, and appropriate modifications were incorporated into the MA-7 and subsequent spacecraft to improve reliability. The greater portion of these changes involved the electrical and sequential systems because of their unique relationship to critical mission functions. These changes dictated paralleling of redundant sensing elements in some cases in which the actuation of either element could initiate the proper function. In other cases where it was important that an event signal not be sent early, some elements were changed to a series function, as was done for the parachute-deployment circuitry.

The primary change to the electrical system for the MA–9 mission was the replacement of two 1,500-watt-hour batteries with two 3,000-watt-hour batteries. This change brought the power supply up to one 1,500-watt-hour and five 3,000-watt-hour batteries.

During the early phases of the flight program, difficulty was experienced in maintaining the temperatures of the electrical inverters below the maximum recommended operating level. A cooling system was subsequently installed for the two main inverters, but contamination problems and the limited effectiveness of this cooling system did not alleviate the elevated temperature situation appreciably. However, continued operation of these inverters from mission to mission, in conjunction with ground test results, without experiencing a temperature-associated failure, provided sufficient confidence that these units would operate satisfactorily. Finally, for the MA–9 mission, modified inverters with improved thermal characteristics were installed in place of two of the old style units (main 250 v-amp and 150 v-amp) and the open-cycle evaporative cooling system was deleted. The three spacecraft inverters functioned satisfactorily until late in the MA–9 flight when an electrical short circuit prevented their operating properly.

In the MA–9 flight, the failure which caused the greatest concern was first recognized at the early illumination of the 0.05g sequence light, which indicated that the automatic stabilization and control system (ASCS) had possibly switched to its reentry mode of operation, which would have included the initiation of rate damping and a steady spacecraft roll rate. Subsequent checks by the astronaut revealed, in fact, that this control mode had been enabled. A requirement for a manual retrofire maneuver was therefore imposed on the astronaut, but it was still the plan to use the autopilot during reentry. However, soon after this occurrence, the main inverter ceased to supply a-c power, and, in the switchover to the standby unit, this redundant element did not start properly. (Refer to fig. 3–21 for details involving the ASCS and power supplies.) Without a-c power for the control system, even the reentry control configuration was disabled; therefore, the astronaut was required to conduct this maneuver with manual control. This task was further complicated by a corresponding loss of gyro attitude indications because of the a-c power failure. A postflight inspection and analysis

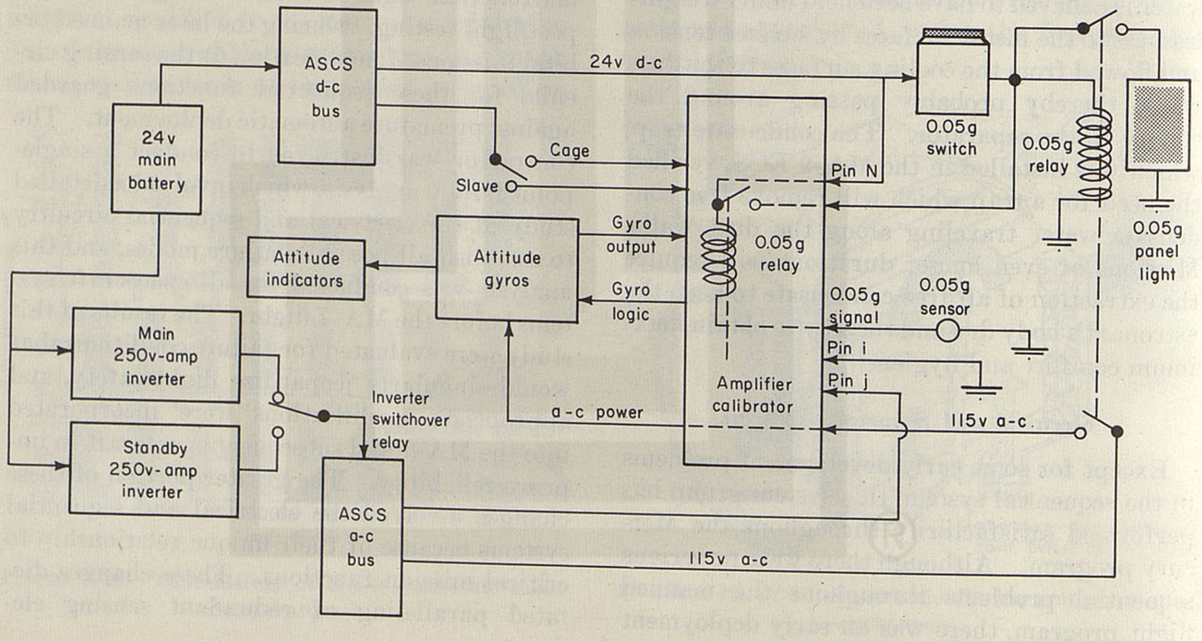
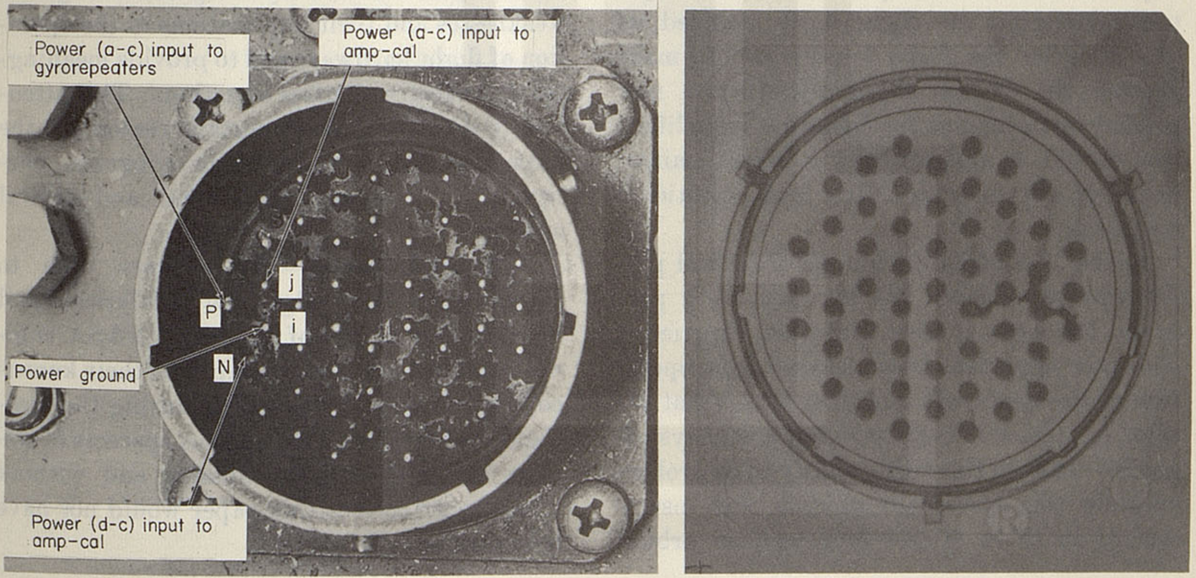


FIGURE 3–21.—Relationship of electrical power to control system autopilot.

of the trouble areas disclosed that a short circuit had occurred, both on the power plug (shown in fig. 3–22) to the ASCS amplifier-calibrator and to another connector (see fig. 3–23), also part of the ASCS power circuit. Both inverters under question were tested thoroughly after the flight and found to operate within specification, indicating that they did not contribute to the malfunction. Strong evidence exists that free water in the spacecraft cabin had been present near the multipin power-plug connection and eventually provided a current path in the insulation between the d-c power and grounding pins shown in right-hand photograph in figure 3–22. Pin N, labeled in the figure, was found to have been completely burned off. Figure 3–23 clearly indicates the significant corrosion revealed on the second connector during the post-flight disassembly and inspection.

Postflight tests duplicated the above hypothesis; that is, a short to ground could be effected upon application of condensate water. Resistance measurements taken across certain pins of the second plug immediately following the flight indicated electrical paths that could have caused the 0.05g indication. A likely source of the liquid which might have caused the electrical short circuit was the porous vent of the condensate tank in the environmental control system. This tank is located in the proximity of the auto-pilot power plugs, and normal cycling of the sponge squeezer during the flight could have forced condensate through the vent. Another possible source of water which could have produced the short circuit is the local condensation of cabin humidity, which may have been present because of a leak in the drinking-water valve or because of water vapor exhaled by the pilot when his helmet faceplate was open. Or the water droplets which leaked from the valve may have somehow been deposited, in part, directly on the power plug. This experience points up the need to minimize or eliminate the presence of free liquid or high humidity in a spacecraft cabin where electrical systems are functioning and to insulate and seal bare electrical connectors more effectively.

Figure 3–23.—Postflight photograph of MA–9 connector-socket rear face.



(a)—Front view showing burnt pin.

(b)—Rear view showing X-rayed current paths in insulation.

Figure 3–22.—Postflight photograph of MA–9 auto-pilot power plug.

Concluding Remarks

The Mercury spacecraft systems design and development phases were conducted concurrently and although this philosophy involved a known risk, it made possible the early realization of the project objectives. During this time, many valuable lessons were learned and exploited in the development and operation of manned space-flight systems.

In the system design, maximum use was made of existing technology and off-the-shelf equipment, and systems concepts were kept simple. However, some important advances in the technology also had to be initiated. It was found that the spacecraft and its systems must be designed for operational conditions. Examples of the design-for-operation standard relating to the preflight activities are system accessibility and the simplification of system interfaces. It is also important in the early system design to allow for an inevitable growth in weight.

During development and qualification testing, the test criteria cannot be compromised in most instances, since an overlooked system inefficiency will inevitably show up later where a redesign is more costly. However, it was also found in Mercury that no single qualification criterion necessarily applies to all systems, and local operational conditions must be individually evaluated for each system. Whenever system components are significantly modified, as was done for the *Faith 7* spacecraft to make possible the 34-hour flight capability, a new ground test program for hardware requalification should be administered to insure maintenance of previous reliability and operational standards.

In the area of hardware operation and performance evaluation, the Mercury flight program has been a most valuable experience. The most important lesson learned from operation of the spacecraft control system is that the pilot is a reliable backup to automatic system modes. 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. Another valuable lesson in both the control system and cooling system designs was the avoidance of components which are especially sensitive to contamination. The small valves used to meter reaction control fuel and environmental control system cooling water should have been designed to employ larger flow areas to reduce susceptibility to particle blockage. Other than guarding against stray voltages and sensitivity to transients, the major lesson derived from the performance of the electrical and sequential systems was the need to seal and insulate effectively all electrical connectors from possible sources of free liquid and humidity in the spacecraft cabin. In the life support system, it was also found that the cooling systems must be designed with adequate margins and that food, water, and waste management devices require particular attention because of plumbing complexity and the effects of weightlessness.

Throughout the Mercury development and flight programs, quality control and rigid manufacturing standards were found to be absolutely mandatory if incidental flight failures and discrepancies were to be avoided. Throughout the project, a careful and continuing attention was given to engineering detail in order to make possible the early recognition of system weaknesses and their implications in the operation of flight hardware and to provide meaningful and effective courses of action. This attention to detail was an important reason for the success of the Mercury flight program, particularly the manned suborbital and orbital missions.

Acknowledgement.—The authors wish to gratefully acknowledge the analytical and documentary efforts of the many NASA engineers and technicians who applied their knowledge and foresight unselfishly during the postflight evaluations of the various spacecraft systems for each Mercury mission and without whose contributions this paper would not have been possible.

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4. MERCURY-REDSTONE LAUNCH-VEHICLE DEVELOPMENT AND PERFORMANCE

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Summary

The Mercury-Redstone launch vehicle was used for the first United States ballistic manned space flights. As a prelude to the orbital flight program, the Mercury-Redstone missions provided an opportunity to evaluate the performance of the Mercury spacecraft, the reactions of the astronauts to brief periods of space flight, and the launch and recovery operations. The first steps toward man-rating a tactical missile were made in a series of design changes and modifications based on ground and flight testing. This paper describes development of the first U.S. manned launch vehicle, including the abort system, the reliability programs necessary for pilot safety, and the performance of the Mercury-Redstone space vehicle.

Introduction

The Mercury-Redstone launch vehicle was the United States' first manned launch vehicle. However, it is only the first of a series of launch vehicles which will exhibit an increasing capability in manned space payloads.

By early 1959, several decisions were made in regard to the performance required of a launch vehicle needed for the first phase of the manned flight program. The vehicle had to have both the reliability and performance to place a manned, 2-ton payload safely into a suborbital trajectory in which at least 5 minutes of weightlessness would be experienced and an apogee of at least 100 nautical miles would be attained. In addition, the vehicle had to be available in time to support the desired flight schedule. These requirements narrowed the choice to launch vehicles which had already been developed for a military mission.

At this time, two surplus Jupiter C missiles were available from the Army Ballistic Missile Agency (ABMA). The Jupiter C was an advanced version of the Redstone, a tactical military missile with a record of over 50 successful flights to verify its reliability. The original Redstone could not meet the mission requirements; however, the Jupiter C had elongated propellant tanks, a lighter structure, and the required performance for Mercury. The Jupiter C launch vehicle had been used for conducting reentry studies and placing the first U.S. satellite, *Explorer I*, into orbit.

Therefore, the Redstone vehicle, in its Jupiter C modification, satisfied the basic Mercury suborbital requirements of availability and performance.

However, the Jupiter C did not incorporate all the necessary safety features; and further adaptation was necessary for use as a manned launch vehicle. This development, which is sometimes referred to as "man rating," had as its three major guidelines safety during launch, satisfactory operation from a human-factors standpoint, and adequate performance margins.

The actual adaptation took place in three phases: basic modifications, modifications after ground tests, and modifications after flight tests. Although there were specific hardware changes during the development, the basic man-rating program and design concepts did not require major alteration.

Basic Vehicle Modification

As noted, some basic modification was necessary to adapt the Jupiter C to the Mercury mission requirements. The required modifications and additions made the new Mercury-Redstone launch vehicle physically distinguishable from both the Redstone and Jupiter C missiles. Figure 4–1 illustrates the differences between these configurations. It should be noted that each successive version of the original Redstone was progressively longer.

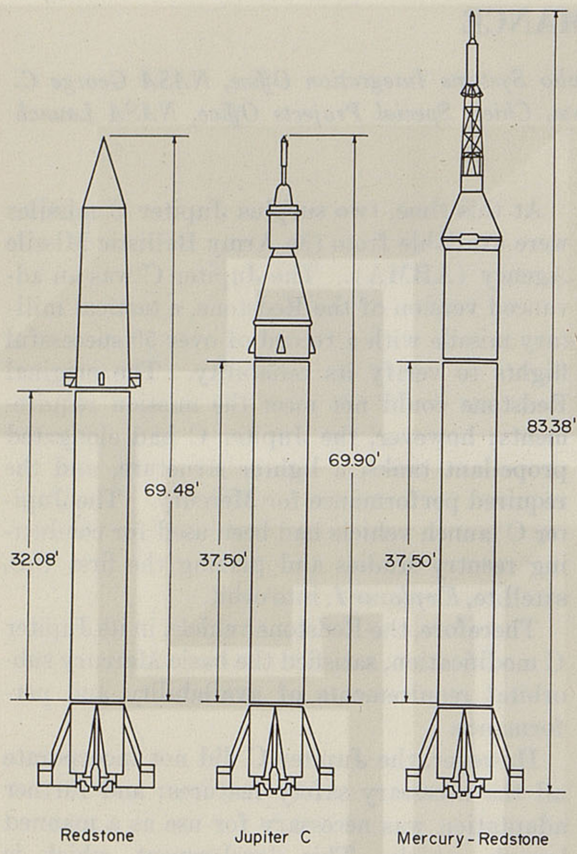

FIGURE 4–1.—Comparison of the three Redstone missiles.

To meet performance requirements, use of the elongated Jupiter C tanks was necessary. These tanks give the Mercury-Redstone launch vehicle a nominal engine burning time of 143.5 seconds, 20 seconds more than the original Redstone vehicle. This greater burning time required the addition of a seventh high-pressure nitrogen tank to pressurize the larger fuel tank and an auxiliary hydrogen peroxide (H_2O_2) tank to power the engine turbopump.

To decrease the complexity for the basic Mercury-Redstone, three changes were made:

(1) The Redstone stabilized platform (ST-80) were replaced by the LEV–3 autopilot for vehicle guidance. The LEV–3 system, although less complex, was more reliable and met the guidance requirements of the Mercury-Redstone mission.

(2) The aft unit, containing the pressurized instrument compartment, and adapter were permanently attached to the center tank assembly. In the tactical version, these units separated with the payload to provide terminal guidance.

(3) A short spacecraft adapter, including the spacecraft-launch-vehicle separation plane, was supplied by the spacecraft contractor. This arrangement simplified the interface coordination.

To prevent major changes midway in the program, the engine was immediately changed from the A–6 to the A–7 model. The A–6 engine was scheduled to be phased out, and a shortage of hardware was expected to occur during the Mercury-Redstone program. This early changeover avoided a foreseeable problem area but required an accelerated test program.

For the Mercury-Redstone launch vehicle, alcohol was chosen as the fuel. Although the Jupiter C had used unsymmetrical diethyltriamine (UDETA) for greater performance, its toxicity was higher than that of alcohol and was considered to be undesirable for manned flights. However, the selection of alcohol led to a problem with the important jet control vanes because of the extended burning time which caused greater erosion of these vanes. Hence, a program was initiated to select jet vanes of the highest quality for use in Mercury.

The prevalves were deleted from the Mercury-Redstone launch vehicle in order to increase mission success. These valves had been used in the tactical missiles between the propellant tanks and the main propellant valves to prevent possible fuel spillage in the event of a main valve failure. However, failure of the prevalves to remain open in flight would have resulted in a mission abort.

To provide for maximum crew safety, an automatic inflight abort-sensing system was added to the launch vehicle and an emergency egress operation was established for the launch complex. These factors were primary considerations in man-rating the Redstone and are discussed in greater detail later.

The Mercury-Redstone was aerodynamically less stable than the standard Redstone. Because of the unique payload characteristics and the elongated tanks, the Mercury-Redstone was expected to become unstable in the supersonic region approximately 88 seconds after lift-off. (See fig. 4–2.) To compensate for this instability to some degree, 687 pounds of ballast were added forward of the instrument compartment.

Changes were also necessary because of the decreased lateral bending frequencies. The configuration and payload changes reduced the Mercury-Redstone bending frequencies to one-fourth those experienced by the standard Redstone. (See fig. 4–3.) As a result, resonance problems appeared during both ground and flight testing. The second bending mode had to be filtered out of the control system to prevent feedback.

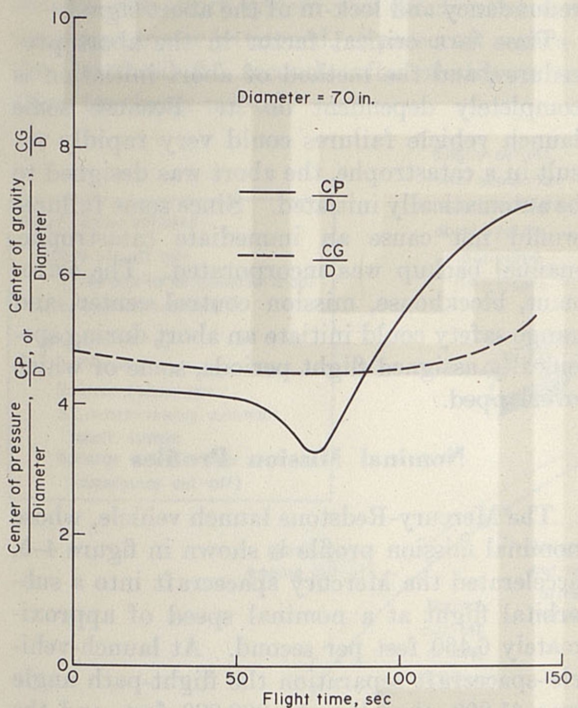

FIGURE 4–2.—Center-of-gravity and center-of-pressure location of Mercury-Redstone during time of flight.

In all, a total of 800 changes were made before the Mercury-Redstone project was completed. The major modifications just described, as well as many minor changes beyond the scope of this paper, resulted in a reliable man-rated vehicle.

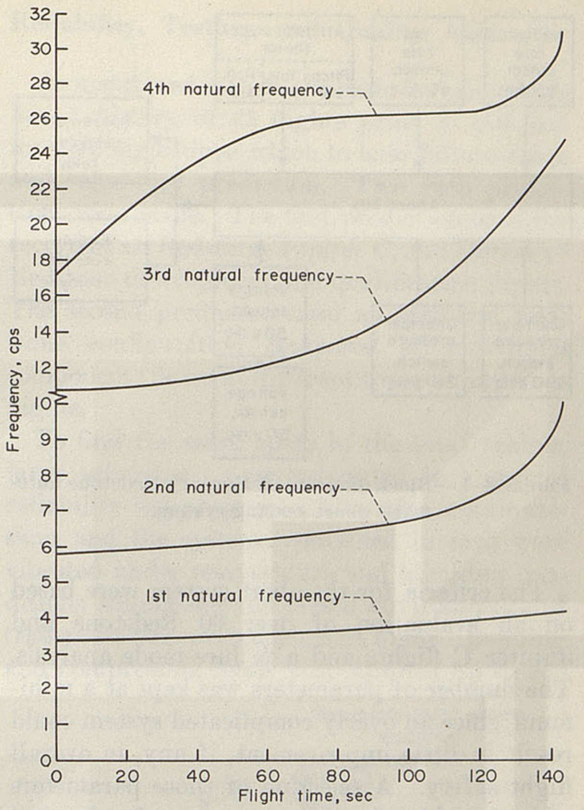

FIGURE 4–3.—Mercury-Redstone lateral bending modes.

Abort System Description

Even though the vehicle was expected to perform properly, a launch-escape system was required for maximum crew safety as long as a catastrophic launch-vehicle failure remained a possibility. Therefore, an automatic inflight system was developed which supplied an abort signal to the spacecraft in the event of an impending catastrophic failure of the launch vehicle. This signal caused engine cut-off, escape-rocket ignition, and spacecraft separation. This cut-off mode was in addition to those sent when the mission conditions were achieved and in the event an emergency command destruct signal had to be sent. Because the vehicle was to be manned, the destruct signal had a built in 3-second delay to allow time for adequate spacecraft separation. The abort system, shown in figure 4–4, sensed and was activated by: unacceptable deviations in the programed attitude of the launch vehicle, excessive turning rates, loss of thrust or loss of electrical power.

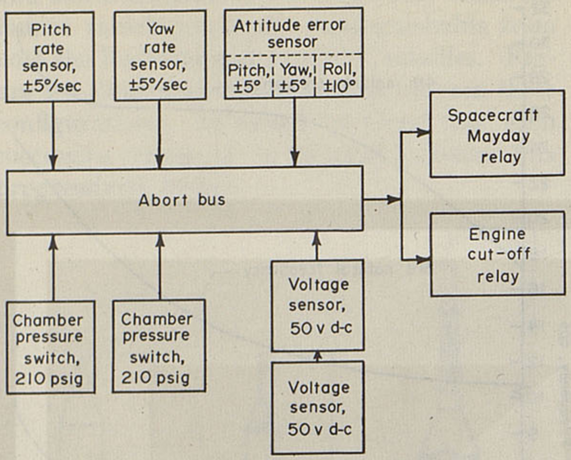

FIGURE 4–4.—Block diagram of Mercury-Redstone automatic abort sensing system.

The criteria for the abort system were based on an evaluation of over 60 Redstone and Jupiter C flights and a failure-mode analysis. The number of parameters was kept at a minimum, since an overly complicated system could result in little improvement, if any, in overall flight safety. A selection of those parameters which would reflect the operation of only vital systems was therefore required. Hence the abort system sensed primarily output or downstream parameters, each of which were then representative of many different types of failures.

For example, a sudden change in the attitude of the vehicle indicated trouble in the control system, regardless of the source of this trouble. It could be the result of a failure in the control computer or some mechanical system or the limits of controllability having been exceeded. By establishing critical values for pitch, yaw, and roll angle, a variety of problems, including the unstable "flip-over" with a subsequent explosion, could be predicted in time for a safe abort. As other examples, loss of thrust, rough combustion, and an impending explosion could be sensed from variations in the combustion chamber pressure. Finally, a loss in electrical power or of the electrical interface between the spacecraft and launch vehicle could be effectively sensed.

As shown in figure 4–4, the abort system circuitry was designed to include adequate redundancy. The combustion chamber pressure (P_c) switches were wired in parallel to assure an abort capability even if one sensor failed. Since the predominant failure mode of electrical voltage sensors is opposite that for a pressure switch, the relays controlled by the voltage sensors were connected in series. Although a single sensor monitored pitch, yaw, and roll attitudes, as well as pitch and yaw attitude rates, redundancy was implicit for these attitude and rate measurements because of their interdependency.

To supply the necessary timing functions to the abort system, relay interlocks were used to prevent arming of the abort system prior to lift-off and to disarm the system at normal shutdown. The P_c switches were armed after engine start and disabled prior to normal shutdown. Here, additional relays also provided circuit redundancy and lock-in of the abort signal.

Time is a critical factor in the abort procedures, and the method of abort initiation is completely dependent on it. Because some launch vehicle failures could very rapidly result in a catastrophe, the abort was designed to be automatically initiated. Since some failures would not cause an immediate catastrophe, manual backup was incorporated. The astronaut, blockhouse, mission control center, and range safety could initiate an abort during specifically assigned flight periods, some of which overlapped.

Nominal Mission Profiles

The Mercury-Redstone launch vehicle, whose nominal mission profile is shown in figure 4–5, accelerated the Mercury spacecraft into a suborbital flight at a nominal speed of approximately 6,460 feet per second. At launch-vehicle–spacecraft separation the flight-path angle was 41.80°, the altitude, 200,000 feet, and the Mach number, 6.30. The maximum acceleration at cut-off was 6.3g.

In figure 4–5, several important launch vehicle sequencing points are indicated. A circuit permitting automatic engine cut-off prior to abort was activated 30 seconds after lift-off. Prior to this time, this circuit was disabled because cut-off in the first 30 seconds would have resulted in an impact of the launch vehicle on land which was undesirable; therefore, only the range safety officer could initiate an engine shutdown. To prevent an early jettisoning of the escape tower the normal shutdown circuitry was not armed until 129.5 seconds. At 131 seconds the velocity cut-off accelerometer was armed. This arming occurred 12 seconds before nominal expected engine cut-off time to allow for higher-than-expected launch-vehicle performance for a non-optimum mixture ratio, which could result in premature propellant depletion. The chamber pressure sensors to the automatic abort system were deactivated at 135 seconds, thus preventing an abort signal at the time of cut-off. Both cut-off activation and pressure switch deactivation were originally scheduled to occur at 137.5 seconds, but as a result of the early shutdown of MR–2, the times indicated in the figure were selected for all subsequent flights.

At engine shutdown, nominally at 143 seconds, the abort system was deactivated and the escape tower jettisoned. Spacecraft separation occurred 9.5 seconds after shutdown to allow for thrust tail-off.

Reliability, Testing, and Quality Assurance

As mentioned earlier, the basic launch vehicle had a history of 69 flights prior to the first manned flight upon which to base failure-mode and reliability prediction. Two such predictions were made. The first prediction used the record of all Redstone, Jupiter C, and Mercury–Redstone development and qualification flights. The second prediction used an artificial Redstone configuration composed of individual components flown at different times on previous flights.

To find the weak spots in the total vehicle, large subsystems were submitted to a special reliability test program. All major missile sections and the systems contained in each were vibrated under temperature and humidity conditions simulating the actual environments of transportation, prelaunch, and flight. Bending and compression loads were applied up to 150

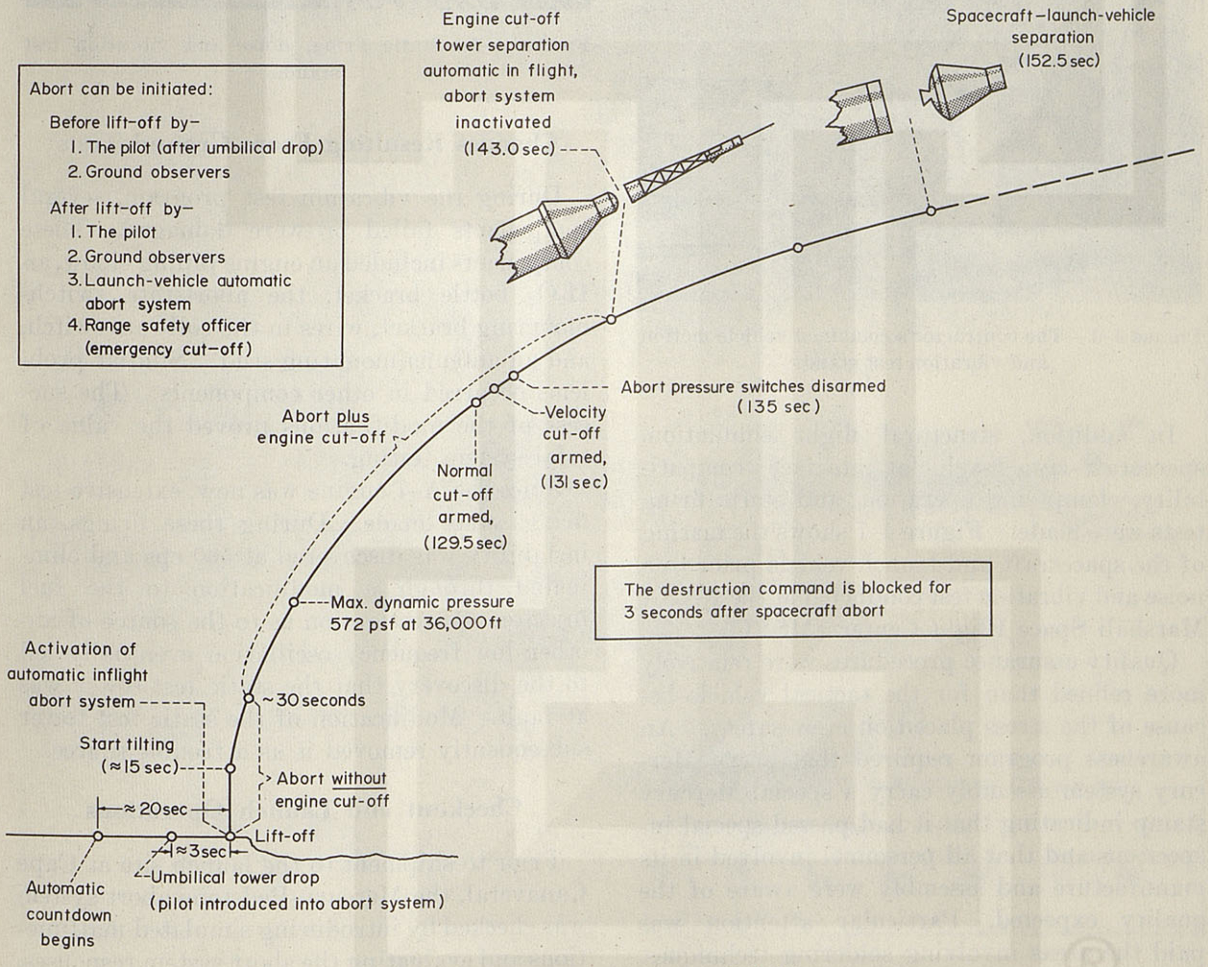

FIGURE 4–5.—Mercury-Redstone powered flight sequence.

percent of maximum flight loads, thereby establishing positive margins of safety. When trouble spots were found, individual component testing was followed up with additional systems tests.

Figure 4–6 shows the vehicle contractor's combined environmental test facility. This facility applied flight vibrations and rigid body motions up to 4g at 2,000 cps simultaneously with temperatures up to 115° F. This testing proved the importance of investigating the interaction of all component masses.

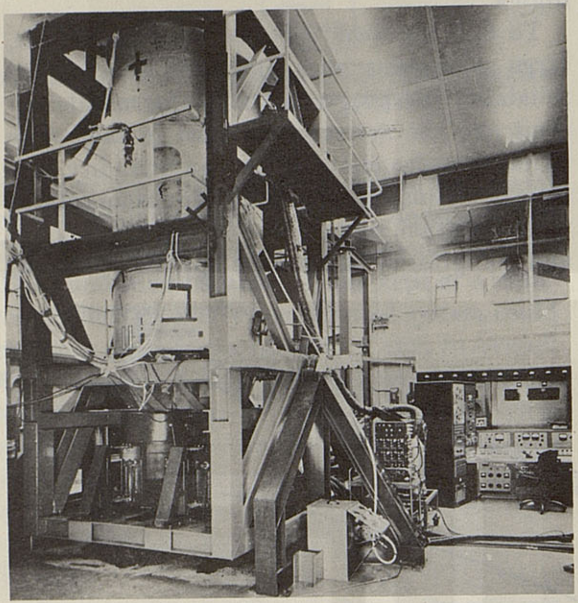

FIGURE 4–6.—The contractor's combined vehicle motion and vibration test stand.

In addition, structural flight simulation, spacecraft–launch-vehicle interface compatibility, clamp-ring operation, and static firing tests were made. Figure 4–7 shows the mating of the spacecraft and launch vehicle prior to a noise and vibration test conducted at the NASA Marshall Space Flight Center (MSFC).

Quality assurance procedures were relatively more refined than for the tactical vehicle because of the stress placed on crew safety. An awareness program required that every Mercury system assembly carry a special Mercury stamp indicating that it had passed special inspections and that all personnel involved in its manufacture and assembly were aware of the quality expected. Particular attention was paid the areas involving soldering techniques, welding repairs, and preparation of instructions.

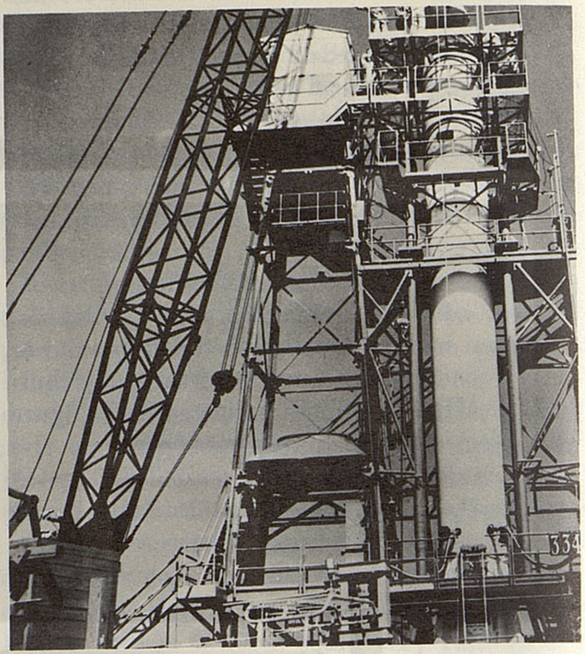

FIGURE 4–7.—Static firing, noise and vibration test stand.

Changes Resulting From Ground Tests

During the vibration test program, several components failed or were damaged. These components included an engine piping elbow, an H_2O_2 bottle bracket, the abort-rate switch-mounting bracket, wires in the roll-rate switch, and an antenna mounting stud. Similar problems occurred in other components. The success of the modifications proved the value of total system testing.

Since the A–7 engine was new, extensive test firings were made. During these firings, an instability was discovered at 500 cps and eliminated through a modification to the fuel injector. Investigation as to the source of another low-frequency oscillation eventually led to the discovery that the static test tower was at fault. Modification of the static test tower subsequently removed it as a trouble source.

Checkout and Launch Operations

Prior to shipment to the launch site at Cape Canaveral, the Mercury-Redstone abort system was checked by introducing simulated malfunctions and evaluating the abort system responses.

The first three launch vehicles were also carefully tested for compatibility with the spacecraft at MSFC.

At Cape Canaveral, the Mercury-Redstone countdown was conducted in two parts with a rest period in between to reduce fatigue of the launch crew. Lox loading was scheduled for completion at 180 minutes prior to lift-off to minimize the possibility of an additional 12-hour delay for lox tank purging and drying during the recycle time in the event of a launch cancellation after lox loading. The astronaut was to be inserted into the spacecraft after lox loading at approximately 120 minutes prior to lift-off. A period of 4 hours was considered to be a tolerable time between astronaut insertion and lift-off to accommodate possible holds in the countdown.

Emergency Egress and Pad Abort

Special astronaut safety precautions were required after insertion since the launch vehicle was already fueled; therefore, launch pad emergency egress procedures were developed. A study (see fig. 4–8) to determine the best mode to retrieve an incapacitated astronaut indicated the blockhouse-controlled service structure would provide the most expeditious escape. If, however, he were able to exit without help,

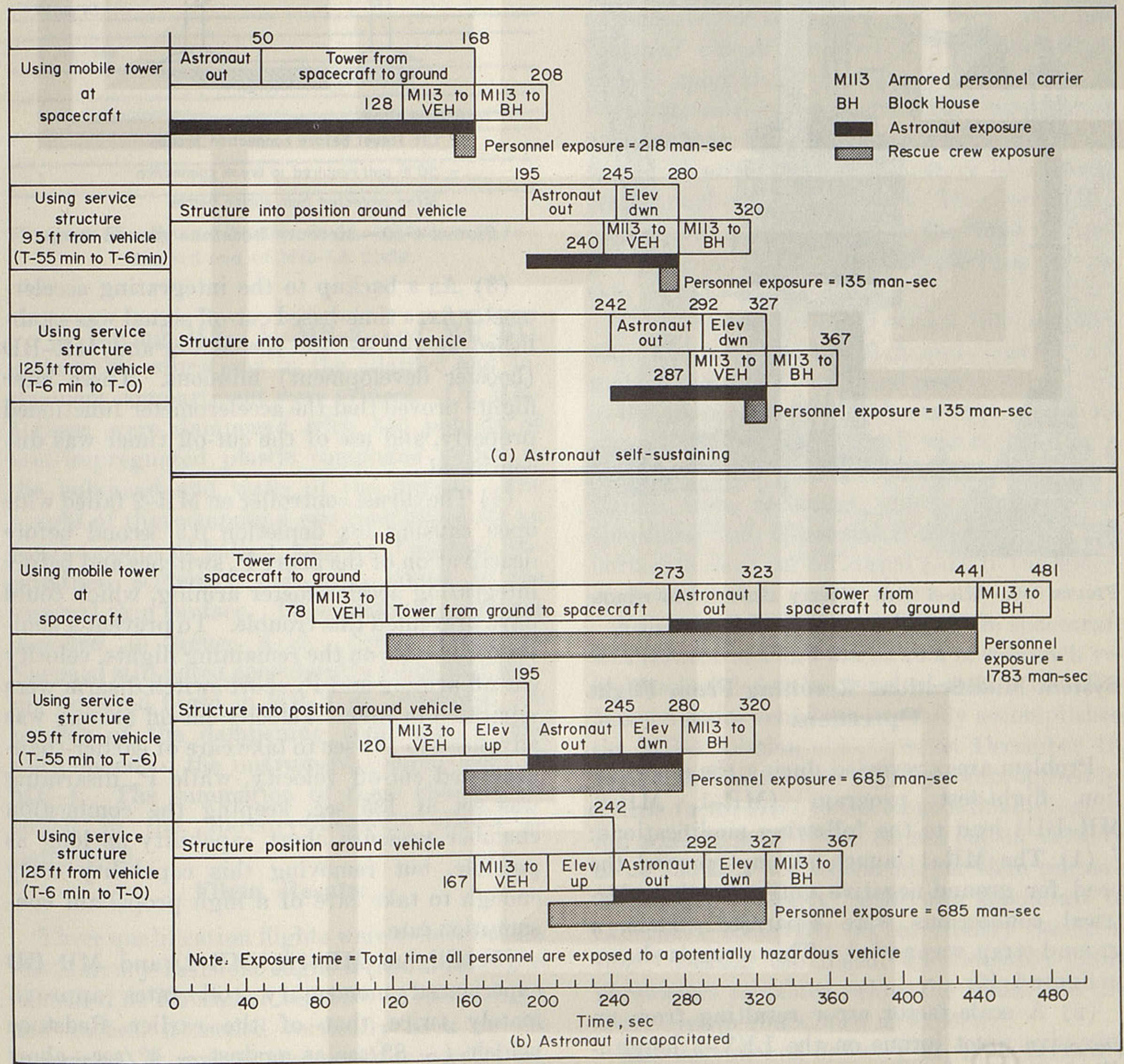

FIGURE 4–8.—Time study of astronaut emergency egress.

he could use the pad escape tower, or "Cherry Picker," shown in figure 4–9. The cab of this specialized escape equipment, which was permanent but extendable, was stationed near the spacecraft hatch until just prior to lift-off. Utilization of this escape device was combined with the use of fire trucks, an armored personnel carrier (M–113), and rescue teams for exit from the pad area. In case of a pad abort, recovery procedures and vehicles, including army helicopters and amphibious craft, were organized and prepared to assist.

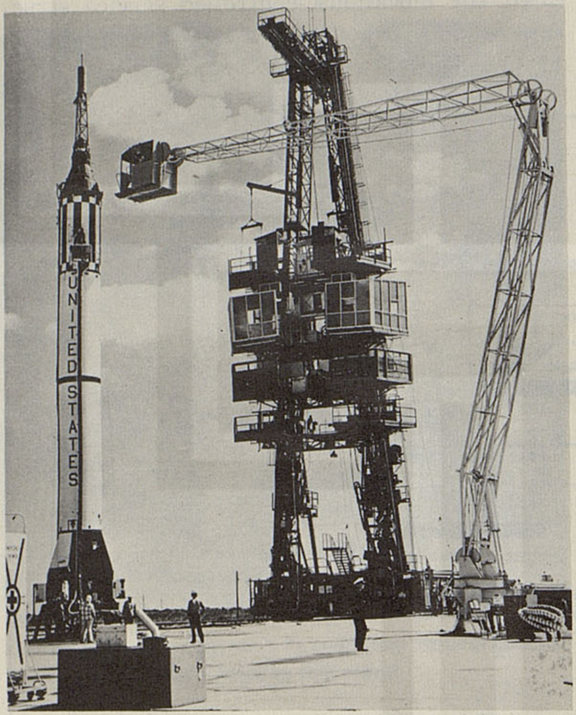

FIGURE 4–9.—MR–3 with "Cherry Picker" and remote controlled service structure.

System Modifications Resulting From Flight Operations

Problem areas revealed during the qualification flight-test program (MR–1, MR–2, MR–BD) lead to the following modifications:

(1) The MR–1 launch attempt proved the need for ground-negative until all other electrical connections were separated. Thus, a ground strap was added. This strap is shown in figure 4–10.

(2) A scale-factor error resulting from an excessive pivot torque on the LEV–3 longitudinal integrating accelerometer caused the MR–1A launch vehicle to experience a cut-off velocity exceeding the nominal value by about 260 feet per second. Use of softer wire and the relocation of the electrical leads eliminated the problem.

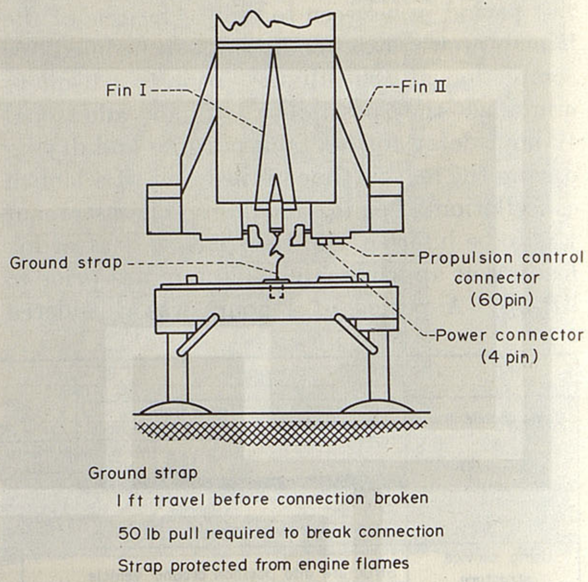

FIGURE 4–10.—Mercury-Redstone ground strap.

(3) As a backup to the integrating accelerometer fix, a time-based cut-off signal was established at 143 sec for the MR–2 and MR–BD (booster development) missions. These later flights proved that the accelerometer functioned properly, and use of the cut-off timer was discontinued.

(4) The thrust controller on MR–2 failed wide open causing lox depletion 0.5 second before deactivation of the abort P_c switches and before integrating accelerometer arming, which could have prevented this trouble. To prevent a similar occurrence on the remaining flights, velocity cut-off arming and P_c abort switch disarm were separated in time. Velocity cut-off arming was advanced to 131 sec to take care of earlier-than-predicted cut-off velocity, while P_c disarming was set at 135 sec, keeping the combustion chamber pressure abort capability as long as possible, but removing this capability early enough to take care of a high propellant consumption rate.

(5) Flights MR–1A, MR–2, and MR–BD experienced momentary roll rates approximately twice that of the earlier Redstone vehicle ($\sim$ 8°/sec as against $\sim$ 4°/sec—abort limits were 12°/sec). Since the missile was not

subject to damage at this rate, the roll-rate abort sensor was deleted after MR–BD to increase mission success. The roll attitude angle abort limit of 10° was retained.

(6) An interaction of the second bending mode with the yaw and pitch axis control required the addition of a network filter to reduce control loop gain between 6 and 10 cps. The interaction was noted on flights MR–1A and MR–2 and is illustrated in figure 4–11.

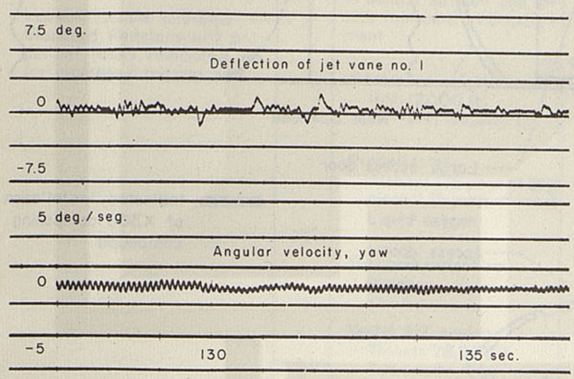

FIGURE 4–11.—Second bending mode oscillations in yaw toward end of MR–1A flight.

(7) During MR–1A, MR–2, and MR–BD, undesirable vibrations in the adapter and instrument compartment were evident. On MR–3 these were dampened with 340 pounds of lead-impregnated plastic compound added to the bulkhead and walls of the section. The weight of this compound was substituted for an equal amount of ballast weight. Fourteen longitudinal stiffeners were also added to the internal skin surface. These improvements are depicted in figure 4–12. Since Astronaut Shepard still noted considerable vibrations during powered flight in MR–3, an additional 102 pounds of the dampening compound, X306, were added to the instrument compartment of MR–4. The summation of these changes resulted in the Mercury-Redstone shown in figure 4–13.

Flight Results

Three qualification flights were conducted for the Mercury-Redstone flight series. MR–1 was launched on November 21, 1960. After rising a few inches, it settled vertically back on the launcher. It proved the need for careful examination of electrical circuitry and led to the addition of a strap for proper electrical grounding.

The sequence of events which led to MR–1's difficulties started during the lift-off when the power and control connectors did not disconnect simultaneously. Because of mechanical adjustments, the power plug disconnected 29 milliseconds prior to the control plug. This permitted part of a 3-amp current, which would have normally returned to ground through the power plug, to pass through the "normal cut-off" relay and its ground diode. The cut-off terminated thrust and jettisoned the escape tower.

The spacecraft did not separate from the launch vehicle because the g-load sensing requirements in the spacecraft were not met. "Normal cut-off" started a 10-second timer which, upon its expiration, was supposed to signal separation if the spacecraft acceleration was less than 0.25g. (This sequencing was designed to minimize the occurrence of a spacecraft launch-vehicle recontact. However, MR–1 had settled on the pad before the timer expired and the g-switch sensing 1g blocked the separation signal.)

The barostats properly sensed that the altitude was less than 10,000 ft and therefore actuated the drogue, main, and reserve parachutes in the proper sequence. The reserve parachute was released because no load was sensed on the main parachute load sensors. To prevent this failure from recurring, engine pressure was monitored and, if normal at 129.5 seconds, the normal booster cut-off signal path to the spacecraft was armed.

Following the MR–1 attempt, the spacecraft was refurbished and mated to a new launch vehicle, scheduled to be launched as MR–1A. The MR–1A space vehicle successfully accomplished the MR–1 mission objectives on December 19, 1960. The launch was slightly compromised by a scale-factor error in the longitudinal integrating accelerometer which caused cut-off velocity to be 260 feet per second higher than normal. This higher velocity caused the spacecraft to experience somewhat higher reentry deceleration. During the flight, all measured abort parameters remained below the limits and the abort system functioned as expected.

MR–2, launched January 31, 1961, carried a chimpanzee named "Ham." On this flight, the

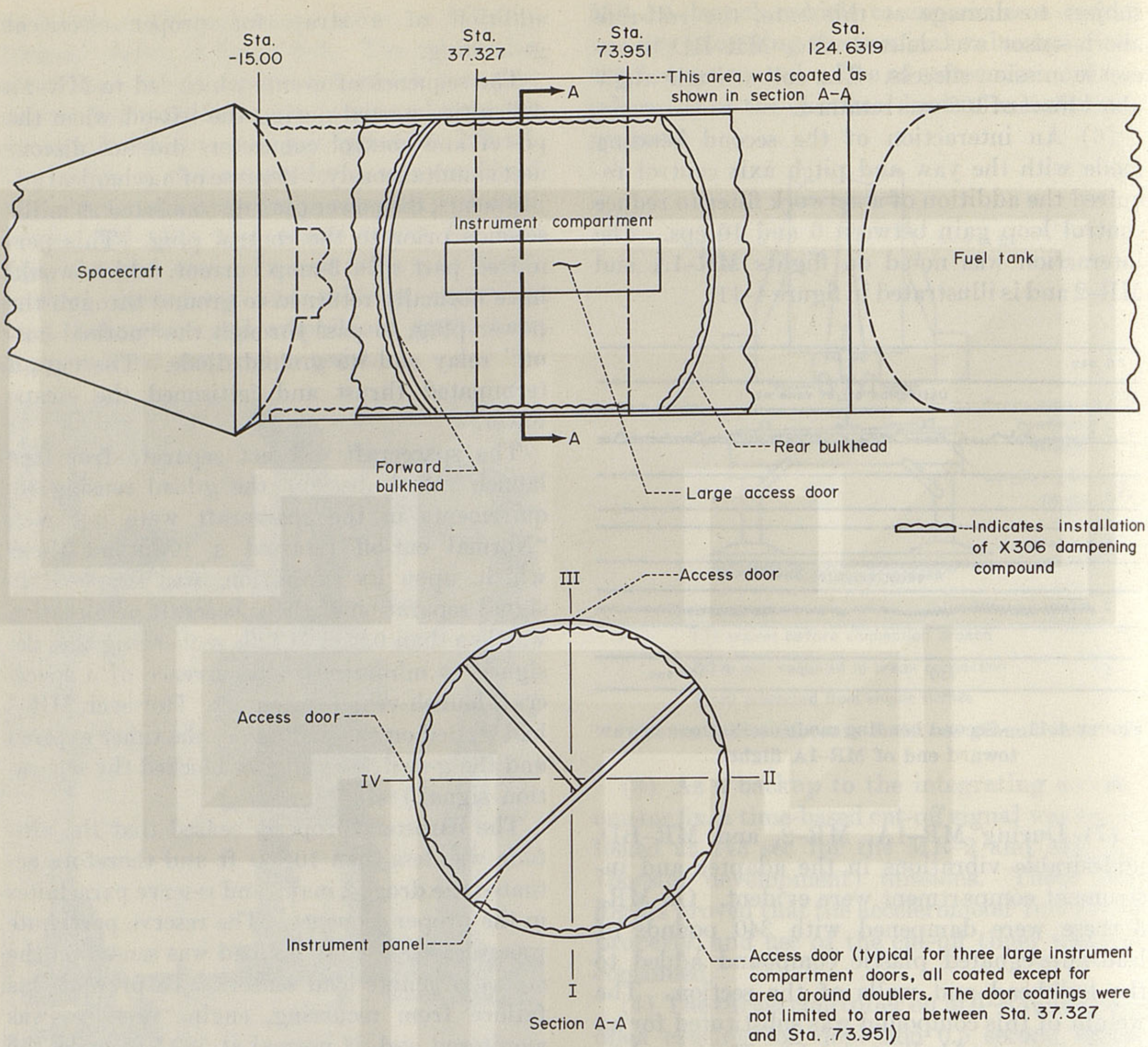

FIGURE 4-12.—Installation of dampening compound in instrument compartment and adapter section for Mercury-Redstone 4 (MR-4).

thrust controller ran above nominal resulting in propellant depletion 0.5 second before abort pressure sensor deactivation. The abort system was able to sense this early shutdown and aborted the spacecraft. The above normal cut-off velocity, combined with the thrust of the escape motor caused the spacecraft to land well beyond the intended recovery area. The simple timing changes explained previously were made to take care of higher propulsion system tolerances.

MR-BD was launched on March 24, 1961, to evaluate a filter network added in the launch vehicle control circuit and modifications incorporated to eliminate the overspeed condition experienced on MR-1A and MR-2. The filter network was intended to dampen the effect of the second bending mode frequency (6 to 10 cps) on the pitch and yaw loop. The flight went exactly as expected and proved the effectiveness of this change.

MR-3 was the first manned flight. With Astronaut Alan Shepard as the pilot, the spacecraft lifted off at 9:34 a.m. e.s.t. on May 5, 1961. All objectives assigned to the launch vehicle were successfully accomplished and no system malfunction occurred. During powered flight,

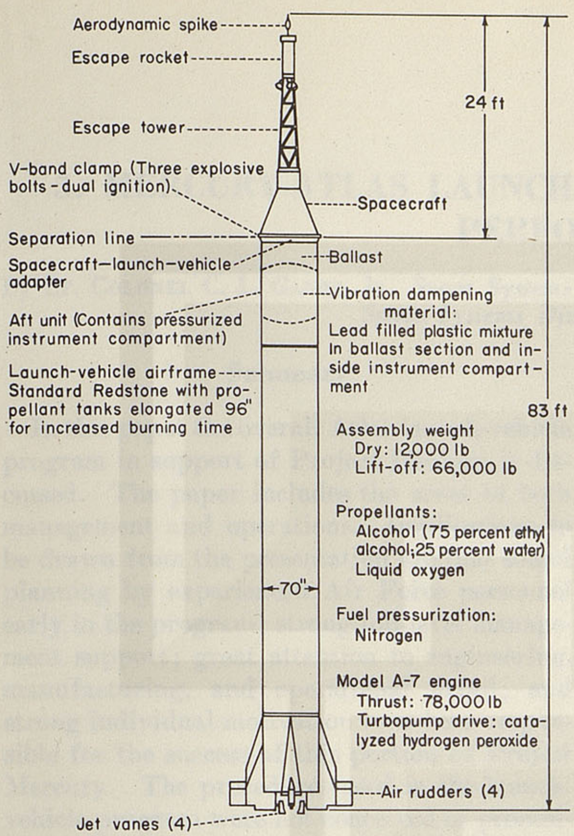

FIGURE 4–13.—Mercury-Redstone configuration.

the astronaut reported buffeting. However, telemetry data indicated lower vibrations than on earlier flights. To reduce these vibrations, additional dampening material was added to the instrument compartment prior to the remaining flight.

Concluding the Mercury-Redstone program was MR–4 carrying Astronaut Virgil I. Grissom in the second manned suborbital space flight. Again, all launch-vehicle systems worked properly and all objectives were achieved. Improved vibration reports indicated that the additional dampening material added to the instrument compartment proved effective.

The Mercury-Redstone flight program was concluded on a positive note with the successful MR–4 mission on July 21, 1961. The first manned flight into space had been accomplished by MR–3 in just over 2½ years from the project's initiation. The initial objectives of providing space flight familiarization and training for astronauts had been accomplished. The spacecraft was exposed briefly to space flight conditions. Of equal importance was the invaluable training of the ground crew in the preparation, launching, and the recovery of a manned spacecraft.

5. MERCURY-ATLAS LAUNCH-VEHICLE DEVELOPMENT AND PERFORMANCE

By Lt. Colonel C. L. Gandy, Jr., *Space Systems Division, U.S. Air Force;* and Major I. B. Hanson, *Space Systems Division, U.S. Air Force*

Summary

In this paper the overall Atlas launch-vehicle program in support of Project Mercury is discussed. The paper includes the areas of both management and operations. Implications to be drawn from the presentation are that sound planning by experienced Air Force personnel early in the program; strong top-level management support; great attention to engineering, manufacturing, and operational detail; and strong individual motivation have been responsible for the success of this portion of Project Mercury. The procedures used in the launch-vehicle program were not conceived or promulgated by any one individual overnight. Rather, they grew from the experience of many and were further shaped by the program itself as it progressed.

Introduction

This paper presents the management aspects of the launch-vehicle system in redirecting a ballistic-missile weapon system into a launch-vehicle system for manned space research. Early agreements between the U.S. Air Force and the National Aeronautics and Space Administration (NASA) established the program responsibilities and identified the management interfaces. Specific guidelines were laid down by the Air Force Chief of Staff to provide effective support to NASA within the military framework of what was then known as the Air Force Ballistic Missile Division. Definitive policies were established to insure maximum launch-vehicle safety for the pilots. The initial overall Mercury systems engineering as it affected the launch vehicle was performed by U.S. Air Force/NASA technical panels and then gradually shifted to the Air Force and its technical contractor, Space Technology Laboratories, and more recently the Aerospace Corporation, for more specific systems engineering.

The basic Atlas "D" system as it existed at the beginning of the program is described to provide a basis for the explanation of the launch-vehicle modifications that were required to support the mission. A brief description is given of the problems that were associated with the individual launch-vehicle flights and the results of the postflight evaluations. A more detailed postflight evaluation is given of the MA–9 flight.

Program Management

During the mid–1950's, the U.S. Air Force conducted a number of studies dealing with manned space flight. Many plans had been formulated and several of the programs had reached a detailed development plan state when, in August 1958, the President directed the assignment of the man-in-space effort to the National Aeronautics and Space Administration. On October 7, 1958, the Space Task Group was organized at Langley Field, Virginia, to manage the then established and later named, Project Mercury.

During the period from October 1958 until April 1959, a series of meetings took place between NASA and the Air Force Ballistic Missile Division to define the AFBMD support required by the NASA-Space Task Group. The problems considered included: definition of the scope of NASA's effort, definition of launch-vehicle requirements, definition of procurement procedures, launch schedules, and launch facilities. It is interesting to note that at the time of the first NASA visit to AFBMD on October 23, 1958, the proposed program envisioned over 25 flights using the Redstone, Thor or Jupiter, and Atlas launch vehicles. Spacecraft orbital weight was to be approximately 2,100 pounds for a 120 nautical-mile orbit. Additional meetings culminated in the issuance to AFBMD of NASA Order HS–24 on November 23, 1958, which specifically requested that the Air Force supply one "C" series Atlas to support Project Mercury. The order specified that this was the initial request of a proposed program which would require approximately 13 boosters of the Atlas and Thor class. On December 8, 1958, AFBMD received NASA Order HS–36 which requested nine "D" series Atlas boosters. Subsequent amendments to HS–36 deleted HS–24, changing the total requirements to 10 Atlas "D" vehicles, later to 14 "D's," eliminating the Thors. Further discussions between the two agencies resulted in the agreement that the Air Force would have full responsibility for the development, procurement, production and launch of the Atlas vehicles for Project Mercury (see fig. 5–1). The final meeting of this series was held between General Schriever, then Commander AFBMD, and Dr. Glennan, Administrator of NASA, on April 7, 1959, in Washington. The basic memorandum of understanding between NASA and the USAF grew from this conference.

A program office was established within the AFBMD to manage the launch vehicle effort, and the services of the Space Technology Laboratories (STL) were requested within the framework of the Atlas weapons system program to support Mercury. Specific guidelines were laid down by the Commander of AFBMD in order that maximum responsiveness to NASA requirements could be assured.

The early systems engineering was accomplished within the framework of technical panels established by NASA. Participants in the panel work were drawn from various NASA organizations, McDonnell, AFBMD, STL and the Atlas manufacturer, General Dynamics/Astronautics. Once the initial problem areas had been defined, technical panels were subdivided into working groups with specific technical areas assigned to assure that thorough treatment was given to all engineering problems. Through the medium of the technical panels, basic trajectory conditions were developed. The launch-escape system concept was born and specific requirements were developed. Reliability goals were established, and systems restraints were imposed. In order to implement, in detail, the general systems approach developed through the technical panels, the Air Force called upon STL to perform these tasks. It was necessary to institute a special systems engineering and technical direction effort for the Mercury/Atlas program, and the STL Mercury Project Office was established in the Fall of 1959 under the direction of Mr. B. A. Hohmann. In the summer of 1960, when the Aerospace Corporation was organized, the task was transferred to this new organization. The majority of the STL Mercury office personnel transferred to Aerospace continued to perform their original jobs. The basic responsibilities of the systems engineering and technical direction group were to develop the technical requirements, monitor the systems and launch-vehicle development, provide trajectory calculations and guidance equations, analyze both ground and flight-test results, assure production acceptability of the launch vehicle, assist in administering the pilot safety program, and provide systems integration of the Atlas associate contractor's systems.

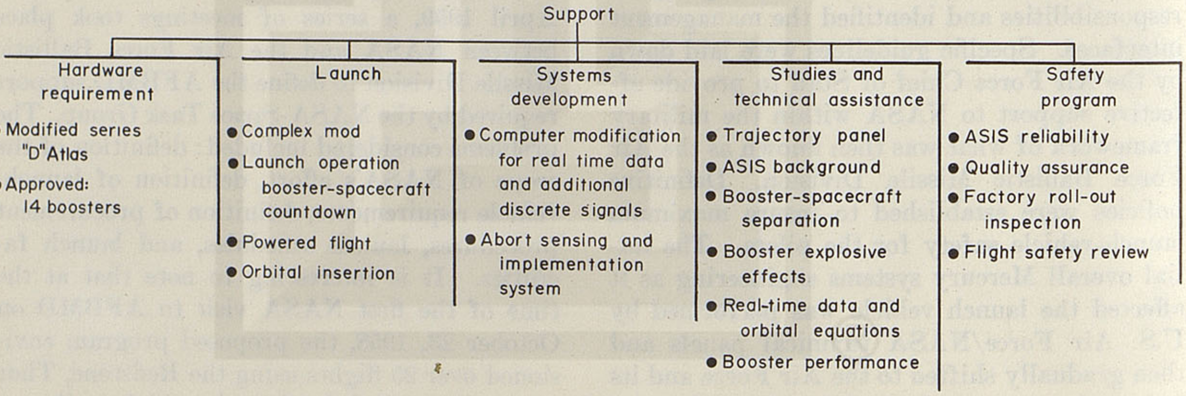

FIGURE 5–1.—Space Systems Division support.

The Space Systems Division and the Aerospace Corporation program offices together were the focal point for detailed management of the launch vehicle program. Program requirements reached this level along a formal path (see fig. 5–2) established from Headquarters NASA to Headquarters USAF, to the Air Force Systems Command (AFSC), to Space Systems Division (SSD), to the Deputy for Launch Vehicles (SSV) to the program offices. A shorter and less formal but equally binding path existed from Manned Spacecraft Center directly to the program offices. Direction received along either path was translated by the program office personnel into action items and routed to the proper agency for accomplishment. Contractual direction and configuration management were controlled by the SSD Program Office originally through the Atlas Weapons System Program Office and later through the SSD Standard Launch Vehicle III (SLV III) Office. Subsystem offices within SSD were responsive to the Mercury launch vehicle program office in the areas of guidance and propulsion systems. Technical direction was handled informally by direct contact between the Aerospace program office and the contractors and formally through the SSD program office.

The Atlas associate-contractor team consisted of General Dynamics/Astronautics (GD/A) who furnished the Atlas airframe and basic vehicle, Rocketdyne Division of North American Aviation (R/D) who furnished the propulsion system, General Electric (GE) who provided both the airborne and ground portions of the guidance system, and Burroughs Corporation who provided the A–1 Computer for inflight guidance in conjunction with the GE system. GD/A performed the launches at the Atlantic Missile Range (AMR) under the supervision of the 6555th Aerospace Test Wing, and the other contractors provided appropriate launch services. Other valuable members of the Atlas team were the Air Force's Western and Eastern Contract Management Regions whose personnel insured the contractors' compliance with contract provisions and performed quality control and technical inspection functions.

Early in the Mercury program, Major General O. J. Ritland, as Commander of BMD recognized that a safety program should be instituted to protect the Mercury pilot. Accordingly, he directed that studies be conducted to determine what efforts were required to insure

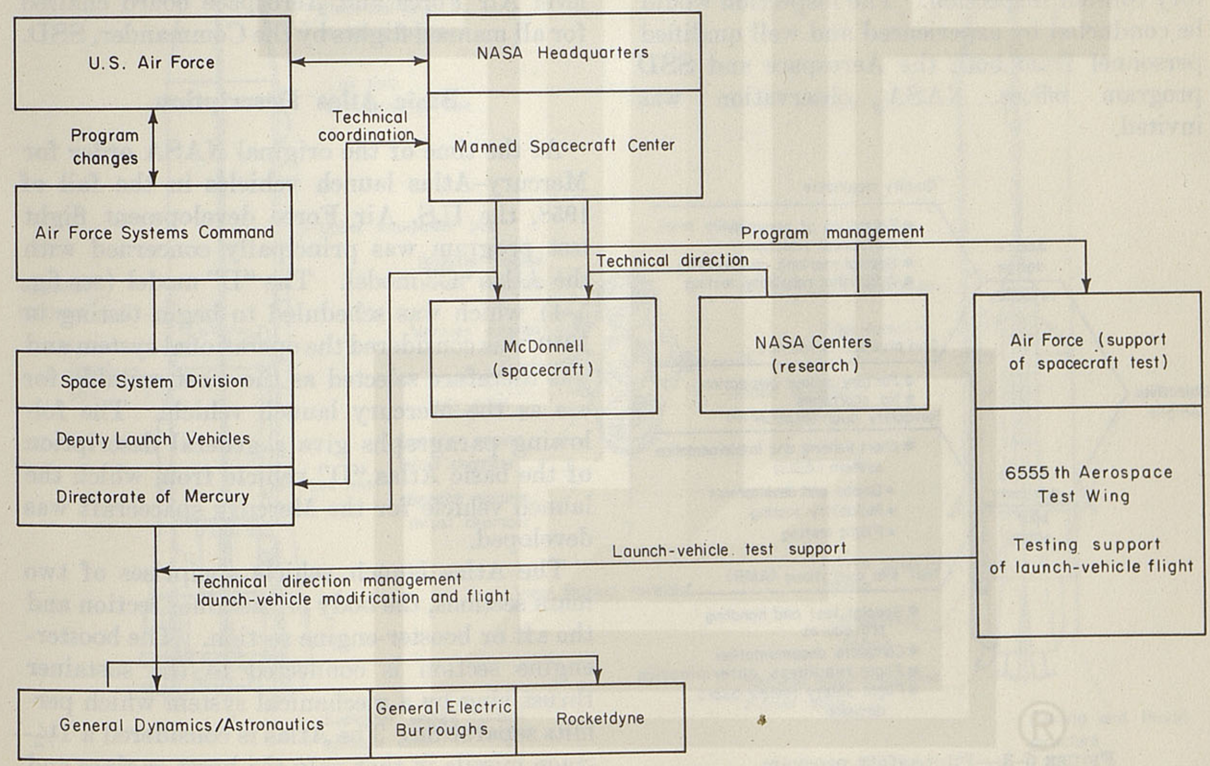

FIGURE 5–2.—Management responsibilities.

safe powered flight and to assure the program management that the launch vehicle was indeed ready for manned flight. This study resulted in the Pilot Safety Program for Mercury-Atlas launch vehicles (see fig. 5–3), a program which has dominated the management of the launch-vehicle portion of Project Mercury.

The basic objectives of the program have been to assure design reliability and adequate pilot safety. Recognizing that the Atlas had been designed as a weapons system and had not been required to meet the reliability expected of a manned system, program personnel established these objectives. The first was to be met through quality of production and end-product excellence. The quality of production would be assured through education and motivation of all personnel associated with manufacture of the hardware, through special component selection and marking procedures, and through special handling techniques. End-product excellence could be assured by requiring that no shortages would be tolerated at the time of launch-vehicle acceptance, and that the vehicle must be complete and up to date with no provisions for field modifications. This assurance would be gained by means of a detailed and highly critical factory roll-out inspection. The inspection would be conducted by experienced and well qualified personnel from both the Aerospace and SSD program offices. NASA observation was invited.

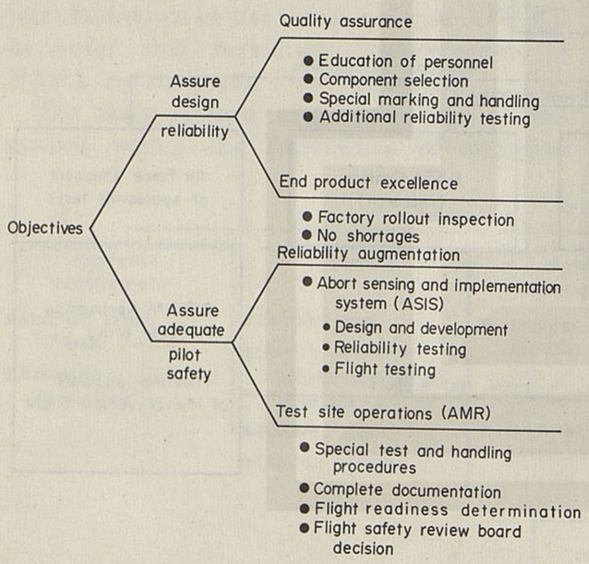

FIGURE 5–3.—Pilot-safety program.

The second objective of assuring adequate safety would be met by providing reliability augmentation and by special test-site operations. The abort sensing and implementation system (ASIS) was designed to bridge the gap between the existing reliability of the launch-vehicle and the near perfection required of a manned system. The ASIS was an automatic system designed to sense an impending catastrophic failure and initiate spacecraft escape prior to the failure. The ASIS itself had to be an extremely reliable system. This reliability was obtained first through a design based upon redundant sensors and circuitry. Then rigid design reviews, stringent ground testing, and finally flight testing were conducted for the system.

The special test-site operations started with unique Mercury handling procedures for the launch vehicle and a requirement that complete documentation be maintained on all prelaunch operations. The documentation, in turn, led to assurance that the vehicle was indeed flight ready upon completion of the required prelaunch testing. The flight readiness was certified by the Mercury–Atlas Flight Safety Review Board. This board was established as a high-level Air Force and Aerospace board chaired for all manned flights by the Commander, SSD.

Basic Atlas Description

At the time of the original NASA order for Mercury–Atlas launch vehicles in the fall of 1958, the U.S. Air Force development flight test program was principally concerned with the Atlas "C" model. The "D" model (see fig. 5–4) which was scheduled to begin testing in 1959, was considered the operational system and was therefore selected as the most suitable for use as the Mercury launch vehicle. The following paragraphs give a general description of the basic Atlas "D" vehicle from which the launch vehicle for the Mercury spacecraft was developed.

The Atlas launch vehicle comprises of two main sections, the body or sustainer section and the aft or booster-engine section. The booster-engine section is connected to the sustainer thrust ring by a mechanical system which permits separation. The Atlas is considered a 1½-stage missile in that only the boost engines and associated hardware are jettisoned at the completion of the first stage of firing.

The sustainer section is made up of a thin wall, fully monocoque structure pressure vessel and derives its rigidity from internal pressurization. The sustainer body is a welded structure of corrosion-resistant stainless-steel sheets varying in thickness from 0.048 inch to 0.015 inch. The tank is approximately 50 feet in length. The forward end consists of a thin dome on which the liquid oxygen boil-off valve is mounted. The base of the dome is joined to the first skin of a conical section whose upper diameter is approximately 70 inches. The conical section joins a cylindrical section 10 feet in diameter. The lower end of the tank is conical, tapering to a point. A hemispherical diaphragm called the intermediate bulkhead divides the tank into a forward section for liquid oxygen and an aft section for RP–1 fuel. A thrust ring joins the conical aft section to the cylindrical portion of the tank. Annular baffles in the tanks serve to dampen propellant sloshing. The sustainer engine with its associated equipment and subsystems is gimbal-mounted to the sustainer thrust cone which is the aft end of the fuel tank. Vernier engine thrust chambers are gimbal-mounted on opposite sides of the structure at the extreme aft end of the cylindrical portion of the tank. Equipment pods containing electronic and electrical units are attached to the tank skin 90° around the tank from the verniers.

The aft section or booster-engine section consists of two booster engines, structure, and associated equipment. It is attached to the thrust ring at the aft end of the tank section by a mechanism which releases it for separation. The motion of this section is controlled during separation by jettison tracks. A radiation shield protects the aft section from the heat radiated from the engine exhaust.

The propulsion system consists of a Rocketdyne MA–2 rocket-engine group made up of two main assemblies: the booster section (see fig. 5–5) consisting of two booster engines having 154,000 pounds of thrust each and the sustainer-vernier group (see fig. 5–6) consisting of

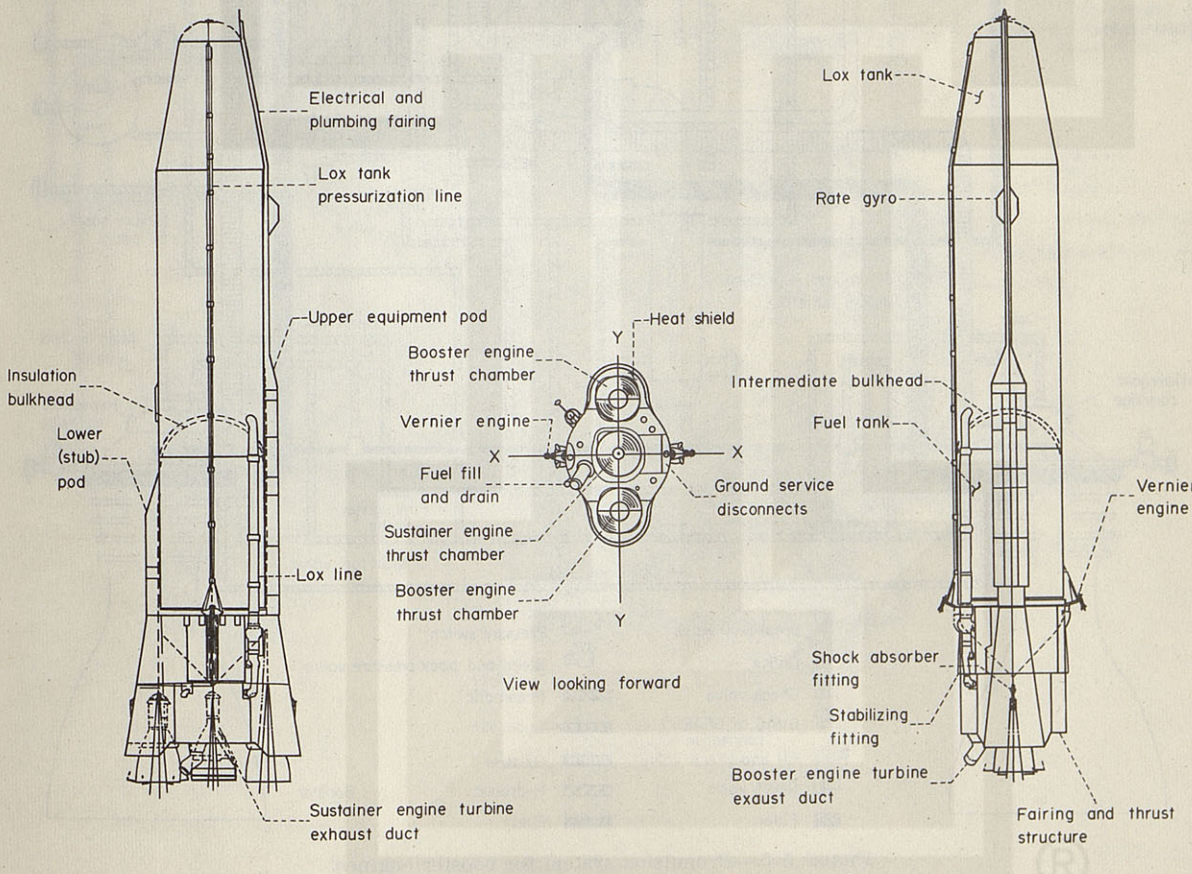

FIGURE 5–4.—Three-view drawing of basic Atlas configuration.

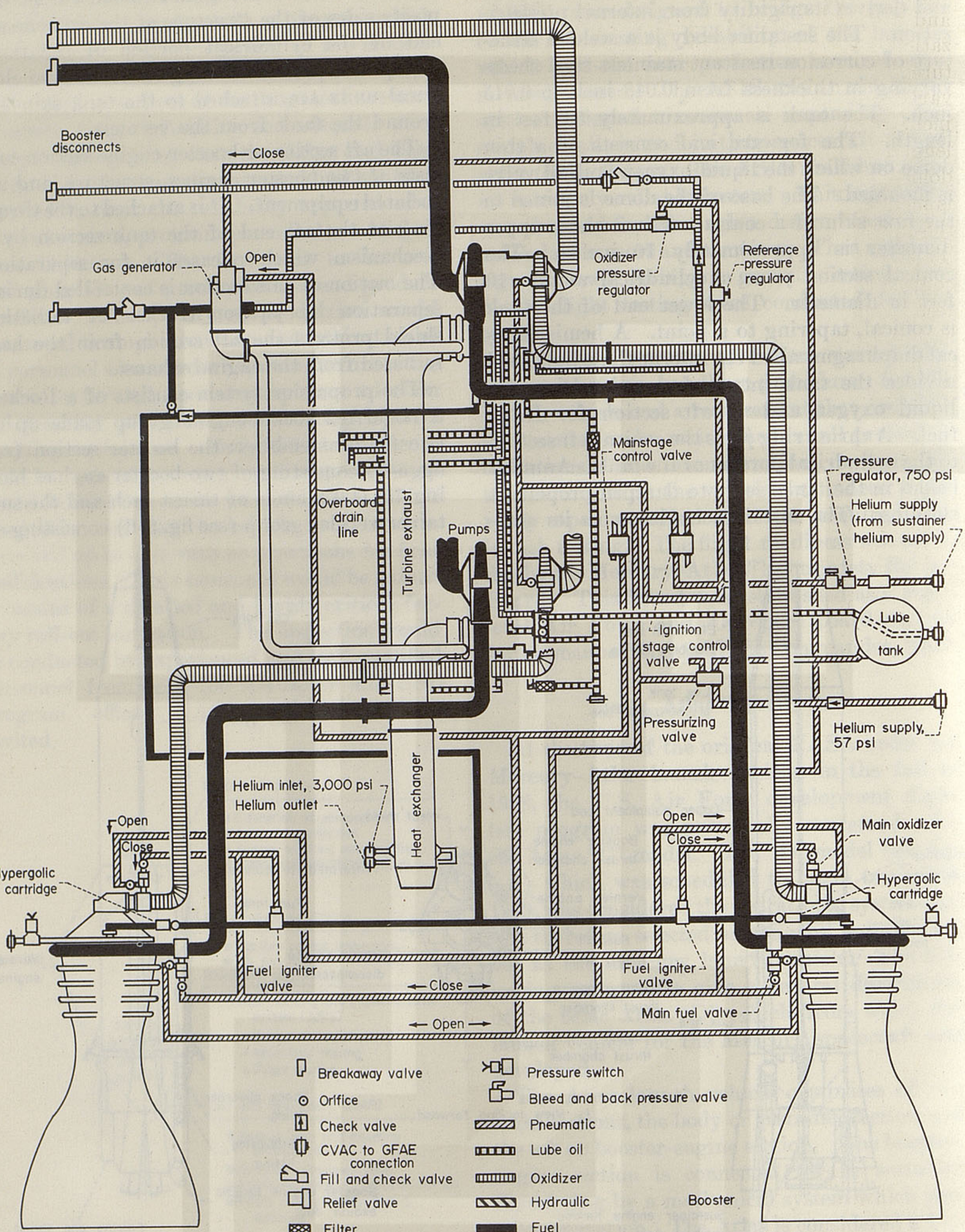

FIGURE 5-5.—Propulsion system for booster engines.

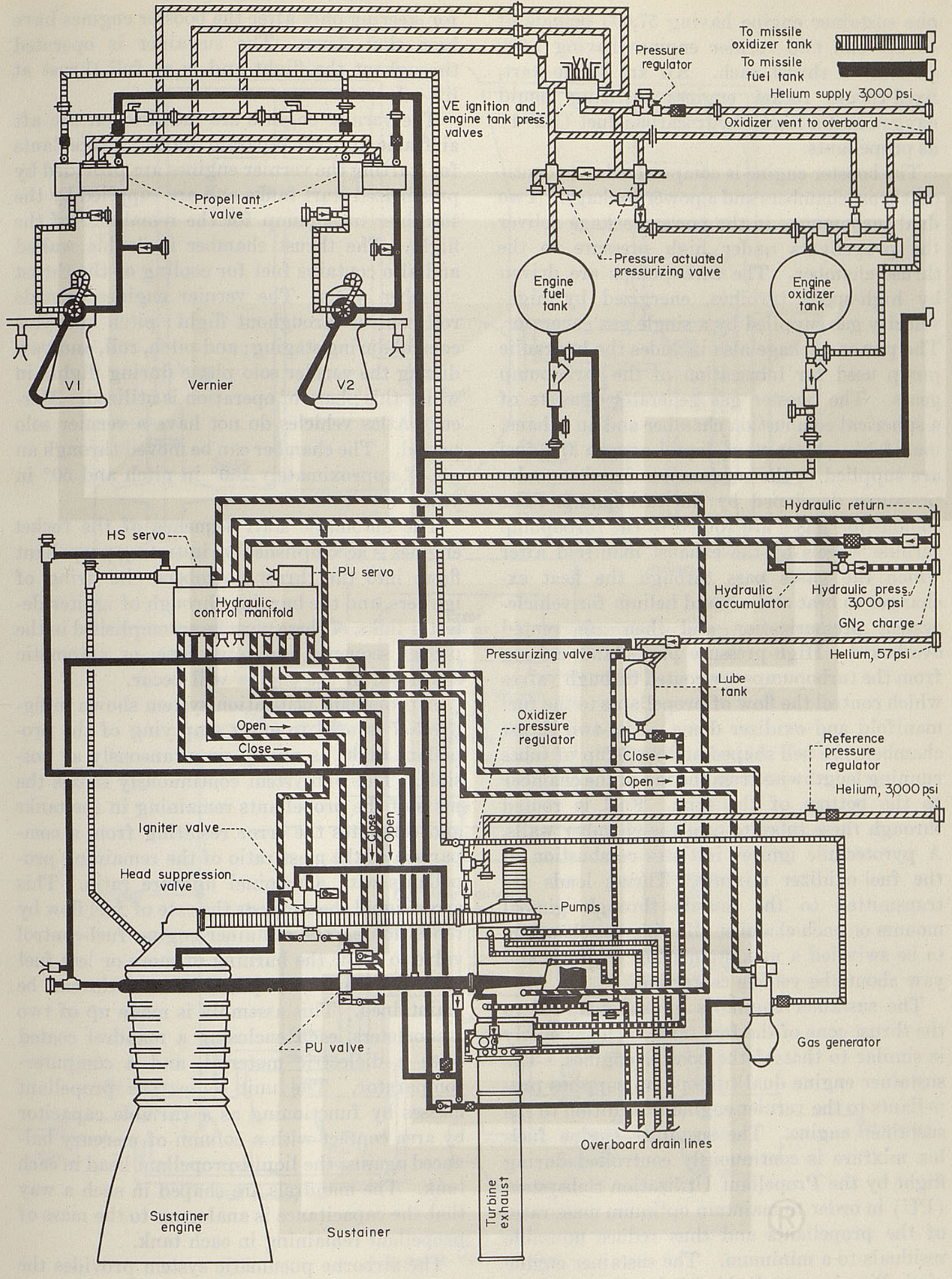

FIGURE 5–6.—Propulsion system for sustainer engines.

one sustainer engine having 57,000 pounds of thrust and two vernier engines having 1,000 pounds of thrust each. All are single-start, fixed-thrust rocket engines utilizing liquid oxygen and a liquid hydrocarbon fuel (RP–1) as propellants.

The booster engine is composed of two identical thrust chambers and a power package. Two dual turbopumps in the power package deliver the propellants under high pressure to the thrust chamber. The turbopumps are driven by high-speed turbines, energized by high-velocity gas supplied by a single gas generator. The power package also includes the hydraulic pump used for lubrication of the turbopump gears. The booster gas generator consists of a spherical combustion chamber and an exhaust manifold. After start, liquid oxygen and fuel are supplied to the combustion chamber under pressures developed by the turbopump. The combustion gases are routed to the turbopump turbine wheels by the exhaust manifold after which the gases pass through the heat exchanger to heat and expand helium for vehicle-system pressurization and then are vented overboard. High-pressure propellants exiting from the turbopumps are routed through valves which control the flow of propellants to the fuel manifold and oxidizer dome. The two thrust chambers are bell shaped and made up of tubes running lengthwise from the top of the chamber to the bottom of the skirt. Fuel is routed through these tubes to cool the chamber walls. A pyrotechnic igniter initiates combustion of the fuel-oxidizer mixture. Thrust loads are transmitted to the missile through gimbal mounts on each chamber allowing the chambers to be swiveled a maximum of 5° in pitch and yaw about the vehicle centerline.

The sustainer engine is gimbal mounted to the thrust cone of the fuel tank. The assembly is similar to that of the booster engines. The sustainer engine dual turbopump supplies propellants to the vernier engine in addition to the sustainer engine. The sustainer engine fuel-lox mixture is continuously controlled during flight by the Propellant Utilization Subsystem (PU) in order to maintain optimum mass ratio of the propellants and thus reduce unusable residuals to a minimum. The sustainer engine gimballing is controlled in pitch and yaw within an arc of ±3°. The sustainer engine is used for steering only after the booster engines have been shut down. The sustainer is operated throughout the flight and is at full thrust at lift-off.

The vernier engines are installed on the aft airframe in two separate units. Propellants for starting the vernier engines are provided by pressurized start tanks and are supplied by the sustainer turbopump for the remainder of the flight. The thrust chamber is double walled and also contains fuel for cooling of the thrust chamber walls. The vernier engines provide roll control throughout flight; pitch and yaw control during staging; and pitch, roll, and yaw during the vernier solo phase during flights in which this phase of operation is utilized. Mercury-Atlas vehicles do not have a vernier solo period. The chamber can be moved through an arc of approximately 140° in pitch and 50° in yaw.

The automatic start sequence of the rocket engines is accomplished by initiating propellant flows into the thrust chambers, the firing of igniters, and the burning through of igniter detector links. These must be accomplished in the proper sequence and total time, or automatic shutdown of the engine will occur.

A propellant utilization system shown in figure 5–7 is used to effect emptying of the propellant tanks as nearly simultaneously as possible. This subsystem continuously senses the mass of the propellants remaining in the tanks and computes the error resulting from a comparison of the mass ratio of the remaining propellants with a nominal mixture ratio. This error signal then adjusts the rate of fuel flow by repositioning the sustainer-engine fuel-control valve to allow the burning of more or less fuel in order that the required mass ratio can be maintained. This assembly is made up of two manometers, each enclosing a mandrel coated with a dielectric material, and a computer-comparator. The unit senses the propellant masses by functioning as a variable capacitor by area contact with a column of mercury balanced against the liquid-propellant head in each tank. The mandrels are shaped in such a way that the capacitance is analogous to the mass of propellant remaining in each tank.

The airborne pneumatic system provides the structural rigidity for the main propellant tanks and also provides the necessary head to

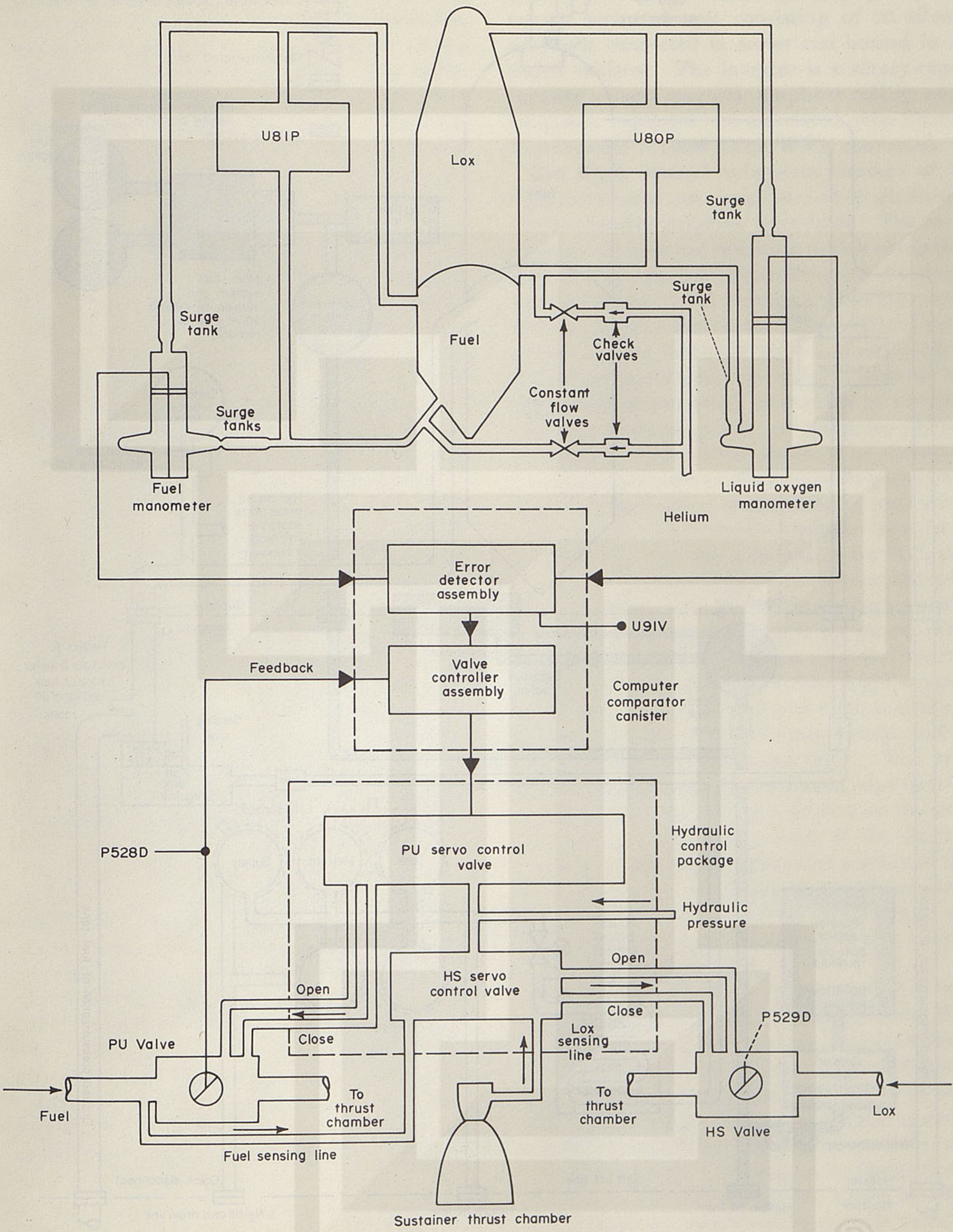

FIGURE 5–7.—Propellant utilization system.

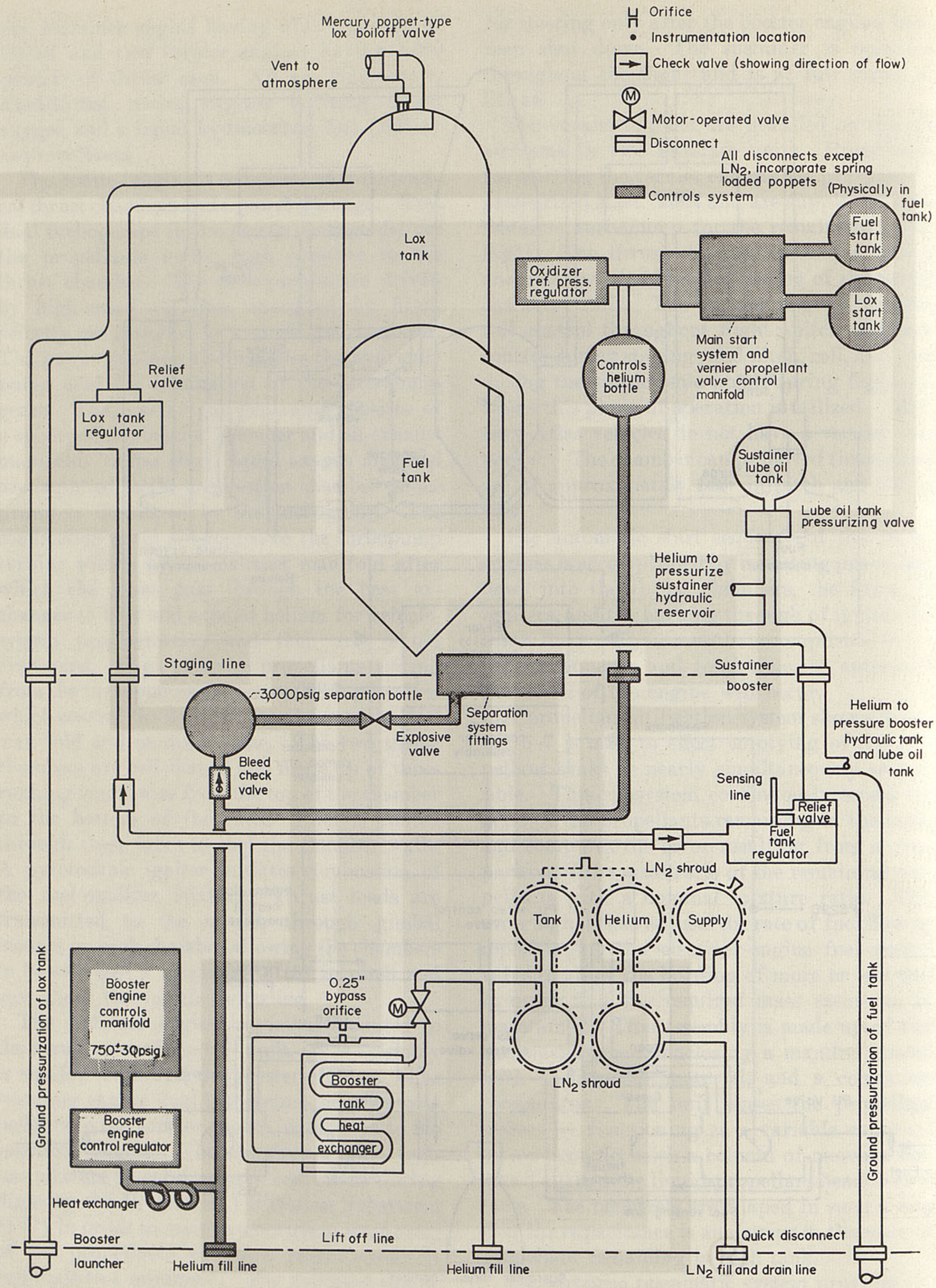

FIGURE 5-8.—Pneumatic system.

prevent the turbopumps from cavitating at low acceleration levels. This pneumatic system, presented schematically in figure 5–8, is used throughout the missile for control, reservoirs, lubricant tanks and the pressurization of the vernier engine propellant tanks. The pneumatic system also provides the actuation force for the first stage separation latches. The pressurization medium is helium, and liquid nitrogen is used to refrigerate the vehicle borne helium supply during the prelaunch phase of the countdown. Five spherical titanium storage vessels are used for the primary supply and are jettisioned with the booster section at staging. The control helium bottle is retained with the sustainer section and provides control pressure for the sustainer section. Tank pressurization is maintained by helium throughout booster-engine operation only. After first stage separation, no helium is required since oxidizer vaporization will keep the pressure in the oxidizer tank above the allowable minimum limits, and main fuel-tank pressure decay will not reduce this pressure beyond the minimum of allowable limits throughout the remainder of the flight. A liquid-oxygen tank boil-off valve is used to maintain proper cryogenic conditions of lox during tanking and holds.

The electrical subsystem (see fig. 5–9) is composed of a 28 v d-c main missile battery and a 115 v d-c three-phase 400 cps inverter. Battery power is provided to the inverter, propulsion subsystem, flight control subsystem, propellant utilization system and abort sensing and implementation system (ASIS). A power changeover switch is used to transfer both a-c and d-c

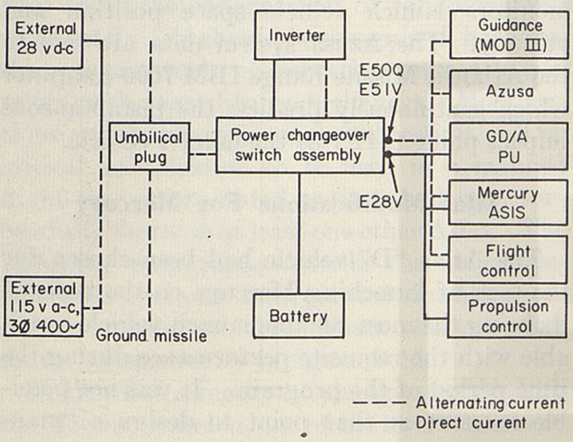

FIGURE 5–9.—Electrical system.

power from external to internal. The position of this switch is manually selected in the launch control blockhouse. The main battery is a remotely activated unit consisting of 20 silver zinc cells connected in series and housed in a sealed canister. The inverter is a rotary-type inverter using a magnetic amplifier voltage and frequency regulator and associated noise filters. The inverter is three phase-WYE connected.

The flight control subsystem consists of a flight programer, an autopilot, and 10 gimbaled thrust-chamber actuator assemblies. The subsystem stabilizes and steers the vehicle along the desired flight path by controlling the direction of the engine thrust vectors. Steering commands are generated on the onboard flight programer during the boost phase. Shortly after first-stage separation, the airborne portion of the guidance subsystem is enabled to provide steering commands to the autopilot for the remainder of the sustainer phase. The autopilot (see fig. 5–10) consists of a gyro package, a servo amplifier package, a programer, an excitation transformer, and engine-position feed-back transducers. On the standard Atlas "D", the main gyro package is located at station 991 and contains three rate gyros, three displacement gyros, and associated electronic equipment. The programer is a transistorized electrical timing device which controls the various flight sequential functions such as roll and pitch programs, staging filter changes, guidance enable, and so forth throughout the entire flight. The programer has two major sequences, the first of which is initiated at 2-inch motion of the missile and the second at receipt of the staging command from the ground-based portion of the guidance subsystem. The servo-amplifier package provides the integrating circuits and includes the necessary filters to insure proper flight attitudes and rates.

The guidance subsystem (see fig. 5–11) consists of the ground-based General Electric Mod III–A X-band radar system, the Burroughs A–1 computer system, and the airborne General Electric Mod III–A guidance group. The Mod III system consists of a position-tracking radar subsystem which determines the position vector of the missile with respect to the guidance station, plus a rate subsystem, which by Doppler techniques measures the missile velocity. In addition, the tracking radar serves as a data

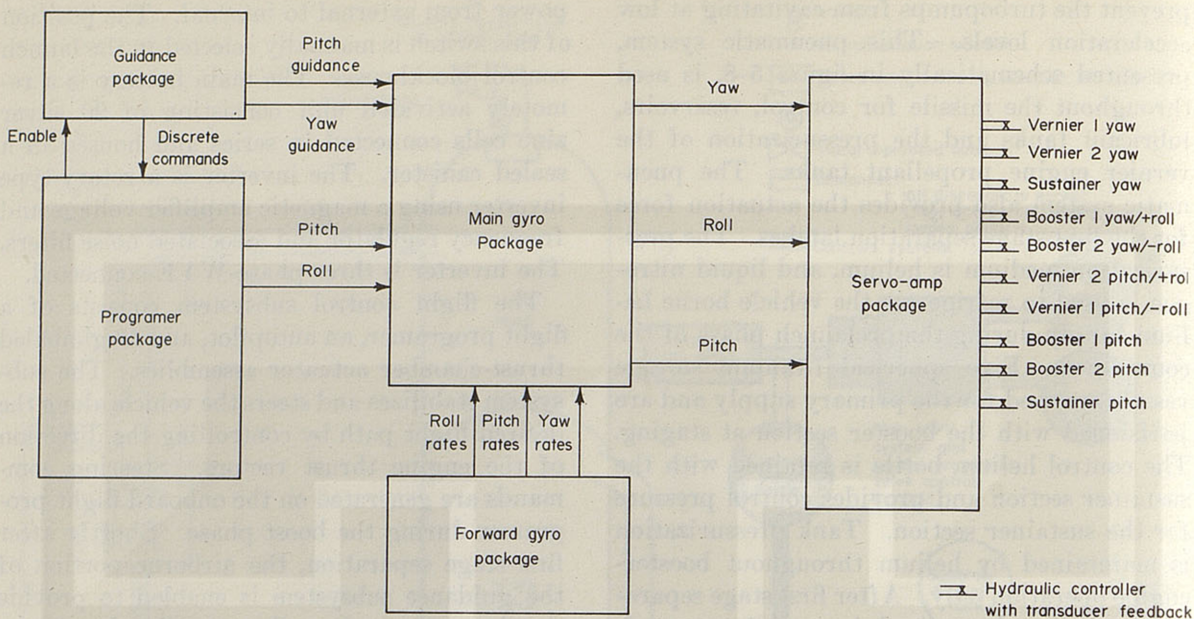

FIGURE 5-10.—Flight control system.

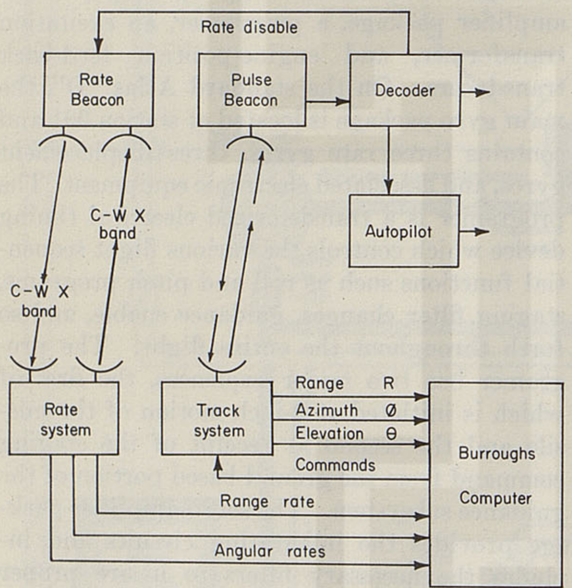

FIGURE 5-11.—Guidance system, MOD III.

link to provide operational commands to the missile-borne equipment. Position and rate data from the radar are transmitted to the Burroughs A-1 computer for processing in accordance with the guidance equations. The computer generates corrective commands which are then fed back into the radar to be transmitted as steering signals to the launch vehicle.

Although weapon-system Atlas vehicles do not require telemetry transmission, research and development vehicles have such a requirement. Two telemetry subsystems were used on Mercury flights. The standard subsystem was used on flights through MA-4 (Atlas 88D). Subsequent flights utilized a lightweight telemetry subsystem (see fig. 5-12) which will be described in the next section.

Two additional systems are installed for the use of range safety personnel. The first is the range safety command system which receives, decodes, and activates the arming, engine shutdown, and destruct functions. The other system is the Azusa radio tracking system which monitors launch vehicle space position and velocity. The Azusa system data are sent to the Atlantic Missile Range IBM 7090 computer which continuously predicts the instantaneous impact point (IIP) of the launch vehicle.

Atlas Modifications For Mercury

The Atlas "D" vehicle had been chosen for the task of launching Mercury on the basis of its being the most reliable launch vehicle available with the requisite performance during the time period of the program. It was not possible to start at that point to design a "man-

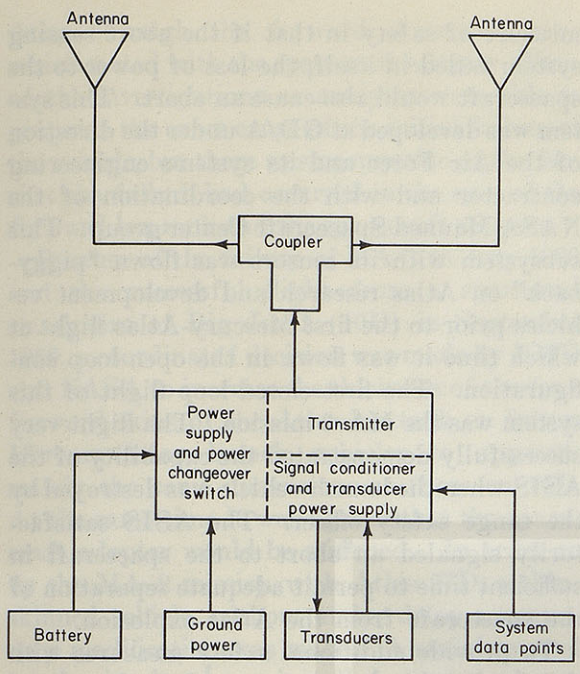

FIGURE 5–12.—Lightweight telemetry system.

rated" vehicle to perform the Mercury flights without several years' delay to the program. Therefore, to capitalize on the reliability inherent in the basic design of the vehicle which had been demonstrated in Atlas development flight tests, a ground rule of the booster program was to make a minimum number of changes to the launch vehicle. Only those changes necessary to adapt the vehicle to the requirements of the Mercury mission or those required to improve the safety of the vehicle for manned flight would be authorized. As with any development program, flight-test experience established the need for incorporation of additional modifications with the major purpose being the enhancement of reliability and pilot safety. It should be recognized, however, that an extremely conservative approach was taken with regard to such changes. Modifications required extensive ground testing, and no critical modification to be used in a manned flight was incorporated until it had been successfully flown on at least one other Atlas. The following paragraphs describe the major system modifications incorporated in Mercury-Atlas launch vehicles. These changes are shown schematically in figure 5–13.

In the first category of changes required by the Mercury mission, one of the most important of the changes was the addition of a new auto-

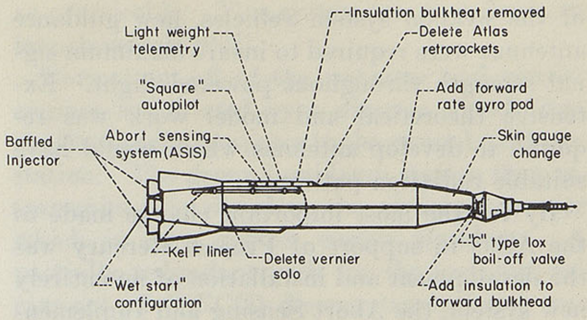

FIGURE 5–13.—Launch-vehicle modifications for Mercury.

pilot rate gyro package in a position considerably ahead of that used on the standard Atlas "D". This addition was dictated by the longer Mercury payload and its effect on the flexible Atlas tank during flight. The modification provided optimum attitude rate sensing with resulting minimum engine deflections for more efficient performance of the launch vehicle. The standard rate gyro installation was retained for abort system sensing.

Additional changes in this category include the deletion of the vernier solo phase of operation and relocation of the retrorockets from the launch vehicle to the spacecraft for use as posigrade rocket motors. In the vernier solo mode of operation the vernier engines remain in operation after sustainer engine cut-off, which allows very delicate adjustments to vehicle velocity. Deletion of this mode permitted a reduction in weight and mission complexity with a resultant improvement in performance and reliability. Relocation of the retrorockets was feasible since the Mercury spacecraft was lighter and the posigrade rockets would thus be more efficient in separating the spacecraft from the launch vehicle. The standard Atlas used these retrorockets to "back off" the launch vehicle from the payload. This relocation of the Atlas retrorockets to the spacecraft retropack required that the thin skin of the lox dome be protected from the rocket exhaust. This was accomplished by developing a fiberglass shield that attached to the mating ring and covered the entire dome. A wet-start technique was also incorporated in the engine starting sequence to minimize starting transients. Another change required for the Mercury mission affected the guidance system. Because the trajectory of the Mercury-Atlas flight differed greatly from that of the weapon system vehicles, new guidance antennas were required to insure maximum signal strength throughout powered flight. Extensive theoretical and model work was required to develop antennas which would have suitable radiation patterns.

By far the most important change made to the Atlas in support of Project Mercury was the development and installation of an entirely new system, the Abort Sensing and Implementation System (ASIS). This system was designed to bridge the gap between the admittedly less than perfect reliability of the basic Atlas weapon system design and that near-perfect reliability desirable for a manned flight system. From a very searching and thorough analysis of Atlas flight data, it was seen that certain missile parameters deviated from a norm sufficiently ahead of catastrophic failure to be used as warnings. It was decided to develop an extremely reliable automatic system to monitor these parameters and to signal the spacecraft escape system when a catastrophe was imminent.

The parameters that were considered the most significant for abort indications (see fig. 5–14)

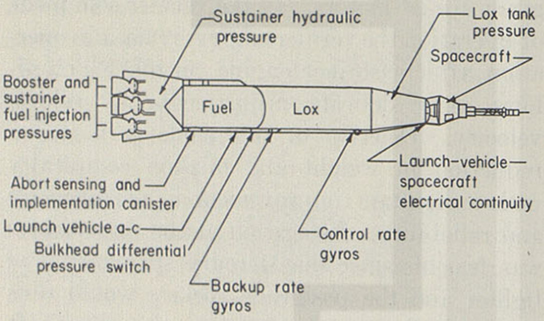

FIGURE 5–14.—Abort system sensors for Mercury-Atlas launch vehicle.

were the liquid oxygen tank pressure, the differential pressure across the intermediate bulkhead, the missile attitude rates about all three axes, rocket-engine injector manifold pressures, sustainer hydraulic pressure, and the launch-vehicle a-c power. Dual sensors for each of these parameters were incorporated into the Atlas system and operation outside a predetermined tolerance band then caused the ASIS to drop out the 28 volt power being supplied to the catastrophic failure detection relays. This drop-out of voltage provided an additional measure of safety in that if the abort sensing system failed in itself, the loss of power to the spacecraft would also cause an abort. This system was developed at GD/A under the direction of the Air Force and its systems engineering contractor and with the coordination of the NASA Manned Spacecraft Center group. This subsystem with its sensors was flown "piggy-back" on Atlas research and development vehicles prior to the first Mercury-Atlas flight at which time it was flown in the open-loop configuration. The first closed-loop flight of this system was the MA–3 mission. The flight very successfully demonstrated the capability of the ASIS when the launch vehicle was destroyed by the range safety officer. The ASIS satisfactorily signaled an abort to the spacecraft in sufficient time to permit adequate separation of the spacecraft from the Atlas explosion.

To provide additional safety measures with the automatic abort, commanded abort, and range safety command destruct, a 3-second delay was incorporated between the signal that commanded engine shutdown and the signal that ignited the destruct package on the launch vehicle. With this change, the launch vehicle could not be destroyed by command for a period of 3 seconds after the engines were shut down. This delay was incorporated to provide adequate separation of the spacecraft from the launch vehicle prior to a command destruct. To provide protection to the launch area, a lockout was incorporated from lift-off to 30 seconds that prevented an abort command from signaling engine shutdown. The spacecraft launch-escape motor had sufficient thrust to provide adequate separation from the Atlas during this period. Immediately after the failure of the MA–1 (Atlas 50D) mission, a special board was convened to investigate the cause of the failure. A number of separate phases of investigation were performed under the direction of the board. These included extensive analyses by Aerospace and GD/A of the thermal environment, discontinuity stresses, and aerodynamic loads. Wind-tunnel tests were performed to gain more knowledge of the aerodynamic conditions imposed on the total flight vehicle in the transonic and maximum dynamic-pressure regions. Analyses conducted by NASA Space Task Group personnel indicated the possibility of concentrated loads being introduced into the

Atlas through the forward structural ring which mated with the spacecraft adapter. None of the investigations or analyses were able to pinpoint the exact cause of the initial failure of the vehicle, but there was no question of the fact that the failure had occurred in the area of the forward lox tank and the spacecraft adapter.

Because of the failure of MA–1 in July 1960 and the successful flight from a structural standpoint of Big Joe I (10D) in September 1959, a coordinated decision was made by BMD and NASA to increase the thickness of the four forward skins of the Atlas lox tank on future Mercury-Atlas launch vehicles to approximately the same dimensions as those on 10D. At the same time it was agreed that the spacecraft adapter would be stiffened. In order to fly the MA–2 mission with Atlas 67D, a thin-skinned vehicle, without undue delay a temporary modification was made. A stainless steel reinforcing band was installed about the lower flange of the mating structure (Station 502 ring) and the first skin aft.

Early in the Mercury program, it was decided to incorporate the electronic "square" autopilot in place of the electromechanical "round" autopilot. The reason for selecting the relatively new electronic system over the proven round autopilot was to obtain improved reliability, improved maintainability due to modular plug in packaging, much increased flexibility to allow for most types of mission changes, and ease of manufacturing by eliminating much of the hidden, point-to-point wiring, and the mechanical setup of the programer. The improved reliability was a result of including such design features as electronic switching in place of mechanical switching, electronic integration in place of electromechanical integration, and improved circuit board design.

Initial flight testing in the Atlas program was accomplished by using an early type of telemetry system. The weight and power requirements to operate the early system were high, and oscillator stability degraded over a short operating time span. A transistorized, lightweight system was developed by GD/A to support the Centaur flight test programs and appeared to be well suited to the Mercury program (fig. 5–12). NASA requested the Air Force to incorporate the new lightweight system as soon as practicable. This system was first flown on launch vehicle 100D.

Normal cut-off of the sustainer and vernier engines is initiated by a discrete signal from the Burroughs computer to the ground guidance station. The ground guidance station then retransmits this signal to the airborne decoder which in turn signals engine shutdown. A partially redundant path for the sustainer-engine cut-off (SECO) discrete transmission was developed early in the program. This path enabled the Burroughs computer to forward the signal to the launch vehicle through the range safety command transmitter, to the airborne receiver and then to the engine relay control. This path was not wholly redundant because no duplication existed in the computer function for generating the SECO time; therefore, a single failure mode still remained. As a result, discussions with the AMR range personnel brought out the capability of the Azusa system to provide a completely redundant SECO discrete signal. The Azusa system in conjunction with the IP 7090 computer continuously computed the instantaneous launch-vehicle impact point (IIP) for Range Safety purposes. With certain modifications to the IP 7090 program it was possible to obtain the time at which orbital velocity was attained. This time was provided electrically by land line to the NASA Flight Director. The Flight Director used this signal as a backup in the event of a failure or malfunction of the Mod III guidance system. This backup SECO system was susceptible to guidance noise; therefore, it was discontinued after the MA–8 mission.

The SECO discrete transmitted to the launch vehicle through the range safety command system as described above, was originally tied to the output of the guidance decoder which obtained a SECO discrete through the guidance system. Both SECO signals used the same path from the guidance decoder and the range safety command receiver to the engine shutdown relays. Additional engineering was required to reroute the signal to provide a completely redundant path.

It is pointed out later in the paper that a problem was discovered with the guidance system at low antenna elevation angles. After a thorough study of the hardware involved, it was concluded that the excessive noise in received signals was cyclic in nature and was caused by an as yet undetermined atmospheric phenomenon. To reduce the effect of the noise in the over-all guidance loop, first the guidance equations were modified to provide additional smoothing, and second, the rate station base legs were increased from 2,000 to 6,000 feet. Although the latter modification did not reduce the actual noise being received, the deleterious effect of the noise on the received signals was reduced by approximately 3 to 1. The third and more complex phase of the study was the development of a mathematical model of the noise to permit a more detailed analysis of the trajectory equation changes that were necessary to minimize this effect. These changes were made to the guidance equations and used on the MA–9 mission.

A fuel tanking test that was being accomplished between the first and second launch attempts of the MA–6 mission brought out a problem that necessitated a major airframe change. The plastic foam material that is used for insulating the base of the liquid oxygen tank from the fuel tank is contained between two hemispherical bulkheads which separate the lox and fuel. A more detailed description of this problem is contained in a description of the MA–6 mission. The limited need for the insulation material coupled with the undesirable feature of removing the bulkhead in the field indicated the need for eliminating the insulation bulkhead from all future Mercury vehicles. A change in the production line stopped further installations of this material.

A major modification in the propulsion system was required to eliminate the possibility of combustion instability. Early in the Atlas program, it was found through flight test experience that combustion instability in the booster engines could cause catastrophic failure of the entire missile. The probability of the occurrence was low; however, the need for maximum safety in the manned space program dictated the need for corrective action. Initially, rough combustion monitors were incorporated and the Atlas was held down for an additional period of time, to allow sensing of the engine vibration characteristics. A rough combustion cut-off (RCC) system then would automatically shut down the engine if combustion instabilities occurred. Again, a thorough ground and flight test program was required before installation on Mercury–Atlas launch vehicles. Another modification provided redundancy in the electrical portion of the propulsion system to insure engine shutdown at SECO. Action was taken also to reroute electrical circuitry to insure proper valve sequencing during start in the high-pressure liquid oxygen plumbing.

Another major modification was made to the booster engine turbopumps. Flight and component testing experience show that incidents had occurred where the lox pump impeller had rubbed against the inlet adapter of the pump. This rubbing caused sufficient heat to ignite the lox and in some cases cause an explosion in the turbopump. Extensive analyses and tests could not pinpoint the exact cause for rubbing; however, the effect of the rubbing could be eliminated by lining the inlet adapter with a plastic material. Months of component and system testing and engineering review were required to provide positive assurance of the suitability of this modification.

Limited changes were made to the pneumatic system specifically for Mercury. Considerable effort was expended however on analyzing tank pressure oscillation that occurs during lift-off under certain payload conditions. The necessary precautions were taken until this problem was resolved. To resolve the entire problem a complex computer model was developed to represent the dynamic conditions existing in the pneumatic system and structure of the Mercury-Atlas vehicle. It was found at the conclusion of the study that earlier characteristics of the helium regulator which controls pressurization gas to the oxidizer tank tended to drive the system into a resonant condition. The new regulator that was used with Mercury did not have the unstable characteristics; therefore, flight restrictions were removed.

The propellant utilization (PU) system was modified to insure an outage of lox rather than fuel in the event abnormal flight characteristics caused the vehicle to expend the total propellant. Early studies had indicated that a safer engine shutdown would be possible in this propellant depletion shutdown case if the lox supply was the first to be consumed. The PU system normally monitors the propellant levels to maintain the proper ratio of onboard propellants. For the Mercury-Atlas the system was modified to drive the mixture ratio to the lox-rich condition at 10 seconds prior to SECO to reduce the ratio of lox to fuel. More recently, a revised method of calibration and a slightly modified mandrel have been developed to provide a more accurate method of maintaining proper propellant ratios.

A normal phenomenon associated with the Atlas vehicle is a roll oscillation that occurs with the missile as the vehicle becomes free of the launcher mechanism. Ordinarily this roll is of small magnitude, and quickly corrected as the autopilot is enabled. A review of flight test history showed that certain vehicles were displaced at roll rates which approached the abort threshold established for the ASIS in roll. Two parallel studies were accomplished to review this problem area. One study reevaluated the abort thresholds to determine if the roll rate limit could be increased. The other study attempted to determine the cause for the roll oscillation in order that a proper modification could be made. It was determined that limited opening of the threshold in roll could be accomplished. The study into the cause for the roll included developing a mathematical model of the launcher mechanism, analysis of control forces required to rotate the missile similar to that demonstrated in flight, base recirculation, engine alinement, and a review of engine acceptance data at Rocketdyne. It was readily apparent that the canted turbine exhaust duct contributed to the clockwise roll moment. This force could cause only half of the roll moment experienced by the missile. Acceptance data from the engine supplier showed that a group of 81 engines had an average roll moment in the same direction of approximately the same magnitude as that experienced in flight. Although the acceptance test-stand and flight-experience data on individual engines did not correlate, it was determined that offsetting the alinement of the booster engines could counteract this roll moment and minimize the roll tendency at lift-off. This change was flight tested and found to correct the roll moment satisfactorily; therefore, the change was incorporated for MA-9 in Atlas 130D.

Flight Test Summary

Big Joe

The first Mercury-Atlas launch was that of Big Joe 1, Atlas number 10D, on September 9, 1959. Atlas 10D was built originally as an R and D vehicle but had received the initial Mercury modifications. The payload was a boilerplate spacecraft. The purposes of the flight were to test the spacecraft's ablative heat shield, afterbody heating, reentry dynamics, attitude control and recovery capability.

Two flight readiness firings (FRF) were performed on Big Joe 1. The first, on September 1, 1959, ended immediately after T–0 because the ignition stage delay timer commanded shutdown of the rocket engines when neither sustainer nor main engine ignition followed normal vernier ignition. There was no booster or stand damage. The second FRF was successfully completed on September 3, 1959, with normal ignition, transition to main stage and shutdown by the engine timer after approximately 19 seconds of running time.

During the launch on September 9, 1959, engine ignition, thrust buildup and lift-off were normal, and launch vehicle performance was completely satisfactory throughout the booster phase. However, after booster engine cut-off (BECO) the booster section failed to jettison and remained attached to the vehicle for the duration of the flight. The sustainer continued to power the vehicle until propellant depletion some 14 seconds prior to normal cut-off. The malfunction resulted in the vehicle failing to achieve planned maximum velocity and in exceeding planned maximum altitude.

Although the injection conditions were considerably different from the preplanned values, the spacecraft reentry satisfied the NASA test objectives. By extrapolating the acquired data, NASA Space Task Group was able to derive the information which was required for spacecraft design. The spacecraft was recovered and returned to Cape Canaveral. Since the data from Big Joe 1 satisfied NASA requirements, a second Mercury launch, Big Joe 2 (Atlas 20D), which had been scheduled for the fall of 1959, was cancelled and the launch vehicle was transferred to another program.

MA–1

The first of the Mercury-Atlas series, MA–1, was launched at 8 : 13 a.m. e.s.t. on July 29, 1960, from AMR Launch Complex 14. The vehicle consisted of Atlas 50D and Mercury Spacecraft number 4, the first production spacecraft, and adapter. The spacecraft primary test objectives concerned structural integrity, afterbody heating and reentry dynamics from a temperature critical abort. Launch vehicle objectives concerned the capability to release the spacecraft at the desired insertion conditions and the evaluation of the open-loop operation of the Abort Sensing and Implementation System (ASIS). A single successful FRF was accomplished on July 21, 1960.

Lift-off and flight of the vehicle were nominal until 57.6 seconds after lift-off when a shock was registered by both the launch vehicle and spacecraft axial accelerometers. The vehicle at that time was at approximately an altitude of 30,000 feet and 11,000 feet down range. The sequence of sensing of the shock indicated that the disturbances occurred in the area of the adapter and the forward portion of the lox tank. All Atlas telemetry was lost at 59 seconds, which is believed to be the time of final missile destruction. Spacecraft telemetry however, continued until 202 seconds, which was the time of landing on the sea, approximately 5 miles downrange. The only launch vehicle primary test objective accomplished was successful evaluation of the open-loop performance of the ASIS which generated an abort signal at 57.6 seconds due to loss of normal a-c voltage.

The failure investigation and results are discussed in the section *Atlas Modifications for Mercury* in this paper.

MA–2

The MA–2 mission was flown by using the Atlas 67D and a production Mercury spacecraft. Test objectives for this flight were concerned with the ability of the spacecraft to withstand reentry under the temperature-critical abort conditions and with the capability of the Atlas to meet the proper injection conditions. This Atlas "D" modified for the Mercury mission, was unique in the program in that it incorporated a stainless steel reinforcing band installed around the vehicle between stations 502 and 510. A thin sheet of asbestos was installed between the reinforcing band and the tank skin. This modification was installed as a precaution against the type of failure which had occurred on the previous MA–1 flight. Atlas 67D had accomplished a successful Flight Readiness Firing on November 19, 1960.

Launch countdown was satisfactory. Although 70 minutes of hold and recycle time were required, none of this time was required for the launch vehicle systems. Lift-off occurred at 9 : 10 a.m. e.s.t. on February 21, 1961. Ignition and transition to main stage were normal, and lift-off was clean. The launch-vehicle flight was uneventful. All test objectives were fully met, and the spacecraft was successfully recovered. This launch was the first one which was preceded by a full Flight Safety Review Board in accordance with the Mercury-Atlas Booster Pilot Safety Program.

MA–3

Atlas 100D, the launch vehicle for the MA–3 mission, was launched from Complex 14 at AMR at 11 :15 a.m. e.s.t. on April 25, 1961. The mission was terminated by the range safety officer after approximately 43.3 seconds due to failure of the launch vehicle to follow its roll and pitch programs. Although the launch-vehicle was destroyed as a result of a malfunction, considerable benefit was derived from the flight test. First, the satisfactory closed-loop performance of the ASIS was demonstrated when the booster engines were shutdown and escape rocket ignition was initiated automatically by the ASIS. The escape was so successful that the spacecraft was recovered some 20 minutes after launch and reused on the next flight.

Second, because of the nature of the failure an intensive reexamination of the complete electrical circuitry and its design, manufacture and installation for both the launch complex and the Atlas was conducted. The malfunction which caused flight termination was isolated to the flight programer or associated circuitry. The programer either failed to start or started and then subsequently stopped without initiating the roll and pitch program. The programer was subsequently recovered, examined, and tested. The most probable cause of the flight failure was traced to contamination of one of the programer pins which under vibration could have caused the failure. The extensive review that was conducted to analyze the flight failure also revealed other deficiencies in the flight control systems. Changes were made to the system to eliminate these possible failure modes and to improve the over-all system reliability.

MA–4

On August 24, 1961, the Flight Safety Review Board for the MA–4 mission (Atlas 88D) performed a thorough review of all pertinent problem areas and all recent Atlas flight test problems. At the completion of the meeting, the Flight Safety Review Board approved the use of Launch Vehicle 88D for the MA–4 mission. The launch was delayed for a 1-week period, and during this period of time a transistor malfunction in one of the flight control canisters aroused considerable concern. An investigation into the factors associated with this failure necessitated an Air Force Program Office decision to delay the flight in order that flight control equipment could be reworked to eliminate this failure mode. The contractor responded to this decision with a concentrated effort to rework and test the equipment in time to support a mid-September launch. On September 12, 1961, the Flight Safety Review Board reconvened. The flight control canister rework was reviewed in detail and the Board concluded that 88D was suitable for launch. The 88D was scheduled for a 250-minute countdown starting at 2:50 a.m. e.s.t. on September 13, 1961. There were four holds and a recycle which resulted in a total count of 374 minutes. Propulsion system performance was normal throughout the start sequence, additional hold-down period and flight. Thrust chamber vibration levels were normal during the hold-down period and chamber pressures were nominal. Lift-off occurred at 9:04 a.m. e.s.t. The flight control systems satisfactorily generated the missile roll and pitchover programs and responded correctly to guidance discrete and steering commands. An oscillation in the pitch plane was evident from T+15 seconds to T+21 seconds. Missile bending was evidenced by an accelerometer located on the lox-dome, launch-vehicle flight control rate gyros, and by spacecraft rate gyros. A change to a launch-vehicle automatic hydraulic actuator had been incorporated on the MA–4 launch vehicle, and the flight control gains had been modified. A postflight modal analysis of the MA–4 data showed that marginal stability characteristics existed with these changes; therefore, additional filtering was deemed to be necessary for future Mercury flights. Propellant slosh amplitudes during the booster phase were low and considerably less than that observed on launch vehicle 67D. The spacecraft injection conditions on the flight of 88D were of the poorest quality of all Mercury-Atlas flights. Tolerance limits were not exceeded; however, a thorough study was required to determine the cause. An analysis of the flight data brought to light tracking phenomena associated with low incident angles. Under certain conditions the guidance system could be affected by varying atmospheric refraction towards the end of flight when the vehicle was approaching the horizon. Limited experience had been obtained at these low elevation angles with the Mod III guidance system. A continuing study was conducted by SSD, GE, Aerospace Corporation, and Space Technology Laboratories in conjunction with the AF Electronic System Division and its technical staff to determine the source and limitations of this phenomenon. Knowledge gained from this study was later used to rewrite the trajectory equations to reduce the effects of refraction anomalies. The postflight evaluation of the launch vehicle 88D mission indicated that all flight objectives were successfully achieved.

MA–5

On November 28, 1961, the Flight Safety Review Board met to consider all aspects of the MA–5 (93–D) mission. Included in the Board review were the autopilot changes that resulted from the previous flight and a thorough discussion of the activities and studies conducted in the evaluation of the guidance phenomena. Additional problems associated with other Atlas space and weapons flight test were reviewed. The Board committed the vehicle to launch.

A number of holds were required during the countdown on November 29, 1963. The data link between the GE ground guidance station and the Mercury Control Center dropped out temporarily, requiring a 4-minute hold, and a 3-minute hold was called at T−7 minutes to resolve a pulse beacon anomaly. Ignition and transition into mainstage were accomplished satisfactorily and within expected limits. There was no indication of the pitch oscillation observed on the launch of 88D. Following lift-off a slight oscillation was noted in the pitch channel during the roll program which is common to all launches. The usual flight oscillation due to slosh was observed from T+86 seconds to T+100 seconds. Staging transients were normal. Approximately 30 seconds before sustainer engine cut-off, a slight oscillation appeared in the pitch channel. This condition persisted for 15 seconds, but the magnitude of the oscillation was of no significance. All flight test objectives were met and the performance of the launch vehicle was within expected tolerance limits.

MA-6

The historic flight of Astronaut John Glenn was conducted on board Atlas launch vehicle 109D and Mercury Spacecraft number 13. This was the flight for which the Atlas Pilot Safety Program had been conceived and for which the launch vehicle team had been preparing so long. Major General O. J. Ritland, then Commander, SSD, convened the Fight Safety Review Board on January 26, 1962, to determine the suitability of Atlas 109D for support of the MA-6 mission. In addition to reviewing the readiness of 109D, the Board reassessed the critical problem areas in the development of the Atlas in support of the Mercury program. This reassessment included all major developments, flight-test incidents and corrective action, the results of additional reliability tests and analyses conducted specifically for Mercury, the performance and test status of the abort system, performance margins experienced on past flghts and the prediction for MA-6, the configuration differences between the previous Mercury vehicle and 109D, and the production and test history of 109D prior to its arrival at AMR. One minor, last-minute problem with a faulty pin connection in the staging umbilical necessitated a second session of the board on January 26, 1962. The condition was repaired, and a complete series of tests to validate all the pin connections in the connector was satisfactorily accomplished. After the second session the Board committed 109D for the launch of MA-6. Adverse weather in the launch area forced the cancellation of the first launch attempt on January 27, 1962. After a tanking test was conducted on January 30, fuel was detected in the insulation bulkhead located between the fuel and liquid oxygen tanks. The insulation bulkhead is located beneath the intermediate bulkhead that structurally separates the two tanks and is composed of a plastic foam material vented to the fuel tank and supported by a thin steel membrane. Test of the plastic material indicated that sufficient fuel could be retained in the insulation material to overload the membrane supporting the insulation bulkhead under flight accelerations. Inasmuch as it was not possible to assess the amount of saturation accurately, a decision was made to remove the insulation material and the supporting structure. The extent of the repair on Atlas 109D at AMR constituted a major but necessary rework of the vehicle in the field. Because of the extent of the repair a highly qualified group of personnel from Aerospace, 6555 ATW and GD/A were selected as a special committee to review all procedures associated with the task. This group also was responsible for validating of the complete task. The primary reason the task was authorized as a field modification was because it had been successfully performed in the field only weeks before on Atlas 121D, the Ranger 3 launch vehicle which flew successfully.

The combined Atlas-Mercury countdown was begun at 11:30 p.m. e.s.t. on February 19, 1962. A built-in hold of 90 minutes was scheduled to begin at T−120 minutes. At T−280 minutes, a telemetry check indicated the Azusa impact predictor was "no-go." The ground station was checked and found to be operating satisfactorily. The tower decks around the transponder were raised, but still the Azusa system could not achieve a satisfactory lock. A decision was made to change the transponder which was accomplished by T−273 minutes. The test was resumed and Azusa was declared "go" at T-213 minutes. No hold time was involved. At T−149 minutes, during the flight control system test, there was a sudden drop in the rate beacon automatic gain control (AGC). The first backup beacon was substituted for the original unit during the built-in hold. This hold was extended for 30 minutes and then extended another 15 minutes to complete installation and retesting. Ten additional minutes of hold were required for the spacecraft. At T−60 minutes a 30-minute hold was requested by Mercury Control Center which was then extended an additional 5 minutes. At T−45 minutes a 15-minute delay was instituted to catch up with the countdown procedures. Lox tanking began at 8:30 a.m. e.s.t. Lox pump problems caused a 25-minute delay in the count. A 2-minute hold at T−6.5 minutes was requested by Mercury Control. The count then proceeded normally to T−0. Lift-off of 109D and Astronaut Glenn occurred at 9:47 a.m. e.s.t. Propulsion system operation during ignition was satisfactory. The longitudinal oscillation normally expected at lift-off were nominal and damped out by approximately 25 seconds after lift-off. Performance of the guidance system was satisfactory. The missile was acquired by radar at the normal time, and tracking was maintained continuously throughout SECO. Steering began at 155 seconds with 60-percent pitchup and 23-percent yaw right commands of 10 and 5 seconds duration, respectively. These initial commands were acceptable for the planned trajectory. Thereafter pitch steering did not exceed 10 percent and yaw steering 5 percent until the end of the flight. Flight control system performance was satisfactory. All monitored programer pitch functions occurred at the proper time. Staging sequence was normal and no evidence of pitch oscillation buildup occurred during the flight. Insertion accuracies were good and well within the tolerance requirements established by NASA. Postflight evaluation of the mission indicated that all systems functioned satisfactorily, and no significant anomalies were apparent.

MA–7

Atlas 107D was shipped to AMR on March 7, 1962, to support the MA–7 flight of Astronaut Carpenter in spacecraft number 18. The vehicle was erected on March 14, 1962, and no serious problems were found during the prelaunch activity. A joint spacecraft and launch-vehicle flight-acceptance composite test (FACT) was conducted on May 4, 1962. The Flight Safety Review Board met on May 23 under the chairmanship of Lt. General Estes, then Commander of SSD, for the purpose of determining the readiness of 107D to support the second Mercury manned orbital launch. The combined Atlas-Mercury countdown began at T−390 minutes at 11:00 p.m. e.s.t. May 23, 1962. The count proceeded very smoothly and without delay until T−11 minutes when the NASA flight director called a 15-minute hold because of unfavorable ground visual conditions. An additional 15-minute hold for the same reason was requested. At 7:17 a.m. e.s.t. an additional 10-minute hold was requested to analyze airborne refractometer test data to determine its effect on the ground guidance system. At 6:28 a.m. e.s.t. an additional 5-minute hold was called to complete the analysis of the refractometer data. Countdown was resumed at 7:34 a.m. e.s.t. and proceeded normally to T−0.

The Atlas vernier-sustainer and booster ignition and transition to mainstage were normal. Lift-off transients were very small and the normal pitch oscillation seen during the roll program was of minimum magnitude. Guidance lock-on was normal. No yaw command was necessary at the time of guidance enable. A slight pitchup was commanded, after which no steering commands were required until just before SECO. Staging transients were very small. An anomaly occurred in the sustainer hydraulic system when at T+192 seconds telemetry data showed that the sustainer engine control hydraulic pressure had begun to drop. The number two ASIS pressure switch activated at T+265.1 seconds when system pressure dropped below the abort level. The number one ASIS switch, which is on a separate sensing line, did not activate and therefore no abort signal was generated. Other telemetry measurements did not show corresponding hydraulic pressure drop. Test simulations conducted after the flight duplicated flight test indications when the sense line was cold soaked at liquid oxygen temperatures. Action was taken to modify future Mercury vehicles by insulating the sense lines. Guidance accuracies for the flight were improved as a result of the extension of the ground based rate system base legs. This was the first Mercury flight to incorporate this modification.

MA–8

Atlas launch vehicle 113D scheduled to support the MA–8 mission on October 3, 1962, incorporated the baffled injector modification in the two booster engines. Sufficient ground and flight test experience had been conducted to provide adequate assurance of the additional flight safety possible with this modification. However, recent ground and flight test failures of the sustainer turbopump created a new atmosphere of concern in the engine area. Investigation of these failures did not reveal any specific cause. Therefore, additional testing was required to determine the susceptibility of 113D to a similar malfunction. An extensive analysis of these past failures did point out that two conditions were common to the failures. The first condition was that the failure occurred during the period of time the fuel control valve was moving into the control position during start. Secondly, the malfunction had always occurred during the initial test of the system in that configuration. For these reasons it was determined that conducting an FRF on 113D in its launch configuration should expose the turbopump to this failure mechanism. Accordingly, an FRF was conducted on September 8. Post FRF evaluation indicated that the propulsion system was flight ready.

Major General Ben I. Funk, Commander, SSD, conducted the Flight Safety Review for the MA–8 mission at 9:30 a.m. e.s.t. on October 2, 1962, to determine the flight readiness of Atlas 113D. NASA concurred with the board's recommendation that the vehicle was in suitable condition to support the MA–8 mission.

MA–8 (Atlas 113D) was launched at AMR Complex 14, 7:15 a.m. e.s.t. on October 3, 1962. The performance of the propulsion system was satisfactory. Telemetered values of all measurements were indicative of normal system operation. Because of the incorporation of the production baffled thrust chamber injectors on the booster engines the missile hold-down time was not extended, and the rough combustion cut-off system was installed open loop on the booster and the sustainer engine for instrumentation purposes only. Flight control data indicated the usual clockwise roll transient at lift-off; however, in this case the transient condition approached 80 percent of the abort threshold. Longitudinal oscillations and pitch oscillations during the initial portion of the flight were nominal and slosh amplitudes were within expected values. All monitored programer switch functions occurred at the proper times and staging sequence was normal. A low amplitude roll limit cycle was apparent from approximately 252 seconds to SECO. Performance of the guidance system was satisfactory with negligible steering commands required after responding to the initial inputs. Insertion conditions were very close to nominal.

MA–9

Atlas 130D was the sixth consecutive launch vehicle to place a Mercury spacecraft into earth orbit. It was the tenth and final launch vehicle used in the Mercury-Atlas program. 130D was accepted at the General Dynamics/Astronautics plant at San Diego, California, on March 15, 1962. Acceptance of this vehicle marked the attainment of a long standing goal of the SSD–Aerospace launch vehicle program offices: acceptance of a Mercury-Atlas launch vehicle without discrepancies or contractual deviations.

The Flight Safety Review Board convened on May 13, 1963, with Major General Ben I. Funk, Commander, SSD, as chairman, to review the status of Atlas 130D to support the MA–9 mission. The MA–8 launch-vehicle performance and the MA–9 launch-vehicle predicted performance were reviewed. All differences between the MA–8 and MA–9 vehicles were discussed, as well as the flight qualification of these changes. The history of manufacturing and testing of 130D at the manufacturer's plant and the prelaunch history at AMR were reviewed. Atlas flight-test experiences were updated to insure that no related problems existed and the board agreed that 130D was ready for flight. An initial launch attempt was made on May 14, 1963; however, the diesel engine used for retracting and stowing the gantry caused a delay in the count when it malfunctioned. Subsequently, the launch was postponed until the following day because of a malfunction in the radar at Bermuda.

The Atlas prelaunch operation, which began on time at midnight of May 14, 1963, was scheduled for a 390-minute countdown plus one planned hold of 90 minutes duration at T—140. There was one unscheduled hold of 4 minutes duration at T—11 minutes 30 seconds, to investigate a signal fluctuation in the Mod III ground guidance system. The anomaly was attributed to an outside source of radiation, and the countdown was resumed. The whole launch vehicle countdown had been exceptionally smooth, and no further delays were encountered. Ignition, transition to mainstage and lift-off were normal with no additional hold-down beyond the normal approximately 2 seconds between flight lock-in and release. Lift-off occurred at 8:04:13 a.m. e.s.t., on May 15, 1963. As the vehicle came off the launcher arms it rolled counterclockwise approximately 0.3° before this minor transient was corrected by autopilot control initiation at 40″ motion. The expected slight longitudinal oscillation associated with lift-off occurred during the first few seconds of missile motion and damped normally. At two seconds after lift-off the roll program was enabled and 130D rolled toward its climbout heading of 72°. The roll program was completed at 15 seconds, and the booster pitch program was enabled. Slight lofting took place during the early portion of the booster powered flight; however, the vehicle intercepted the planned trajectory at 125 seconds. Propellant sloshing became noticeable at 55 seconds, reaching a maximum amplitude at 98 seconds and decaying to a negligible value by 120 seconds. Propellant slosh during this period of time is normal, but the amplitudes on this flight were higher than on most previous Mercury launches. Postflight review of the 130D flight control gains indicated they were within tolerance but below nominal. Higher than normal propellant slosh amplitudes could be expected under these conditions. Booster engine cut-off (BECO) was accomplished at 132.5 seconds with booster section staging at 135.4 seconds. Space position at BECO was very close to planned. At BECO the sustainer engine was nulled in pitch and yaw to assure proper clearance of the booster section during the jettison phase. After booster jettison the sustainer was reactivated in pitch and yaw. The sustainer-stage pitch program was initiated at staging plus 5 seconds and was completed at 159 seconds after lift-off. Entrance into the guidance steering mode was relatively smooth with the initial steering response being slightly up and to the right. After the initial correction, only extremely small steering commands were transmitted. SECO occurred at 303.03 seconds, approximately 1 second earlier than planned. Burnout conditions of the launch vehicle were very close to those planned and were within a few feet per second high in velocity, 500 feet low in altitude, and 0.005° low in flight path angle.

A detailed analysis of flight test data has shown that the launch vehicle performance was very close to nominal. An over-all vehicle postflight trajectory simulation did indicate that the effective specific impulse of the total launch vehicle system was within, but on the high side, of the tolerance band.

The pneumatic system operated satisfactorily, and no anomalies were noted. The tank pressure oscillation which normally occurs at lift-off was of very low magnitude and of no significance to the flight. Adequate pressures were maintained in both lox and fuel tanks and well above the abort limits at all times.

The propellant utilization system exhibited very smooth characteristics throughout the flight and was holding at the nominal position during the period prior to sustainer engine cutoff, indicating that the propellant mass ratio was correct. The PU system on this flight utilized a slightly reshaped mandrel and improved calibration techniques compared to previous Mercury flights.

The sustainer and booster engine hydraulic systems behaved in a normal manner with only slight booster position response to auto-pilot system demands occurring during the propellant sloshing period.

The a-c power supply frequency and the main battery voltage were within specified limits through powered flight. The a-c voltage ran 0.4 to 0.7 volt above the nominal but within the tolerance band. Slight vehicle lofting occurred as a result of this minor shift in a-c voltage.

The flight control system functioned satisfactorily and properly stabilized the launch vehicle. All guidance discrete and steering command functions of the flight control system were properly carried out. GE and Azusa data indicated that the total magnitudes of the booster phase roll and pitch programs were extended slightly beyond nominal but were still well within allowable limits. The major contributor to these excesses was the higher than normal inverter voltage output during the launch to BECO phase of powered flight. It should be noted that the effect of higher than nominal engine performance during boost phase tended to counteract the effect of higher than nominal inverter voltage on the pitch program. As previously pointed out, the propellant slosh was greater than that on most previous flights but its effect on attitude rates was negligible. A low-amplitude roll limit cycle was evident from BECO to SECO. This motion had been noted on previous Mercury-Atlas flights and was not considered detrimental to the mission.

All instrumentation measurements functioned properly throughout the flight, and the telemetry quality was such that a very thorough analysis of all flight parameters was possible.

The range safety command system was not required until the auxiliary sustainer cut-off signal (ASCO) was transmitted 0.04 second after the BECO guidance discrete signal in accordance with the computer program logic.

Performance of the ASIS was satisfactory. Review of launch-vehicle data did not reveal the existence of any undetected abort condition. Switching functions to change abort logic and parameter levels were accomplished in the planned manner from launch throughout powered flight to SECO.

6. RELIABILITY AND FLIGHT SAFETY

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Summary

This paper summarizes the reliability and flight safety features of the Mercury Project. The difference between reliability and flight safety is briefly discussed. The basic concept that no single failure would cause an abort, and that no single failure during an abort would result in loss of the pilot, dictated the need for redundancy and manual over-ride capabilities in spacecraft critical systems.

An existing missile was modified to provide the launch vehicle, and its reliability was augmented by a program of special testing and by careful selection of components. In addition, an abort sensing system was developed for the launch vehicle to provide for sensing of impending catastrophic failure and activation of the spacecraft escape system.

A conservative design approach was used for the spacecraft, incorporating redundancy in all critical systems where possible, in order to provide reliability. Off-the-shelf proven components were used where possible to avoid development problems, and standard design practices were used for designing components where proven components were not available.

The success of the flight-test program proved the effectiveness of the ground test program in disclosing essentially all "early development" and human induced type failures.

Flight safety reviews for the launch vehicle and the spacecraft, and a mission review for all aspects of the mission, were conducted prior to each mission and proved to be effective.

Introduction

The Mercury approach to reliability and flight safety was a practical approach to the problem of achieving manned orbital operation with a reasonable degree of reliability and safety at the earliest possible time. It was an all-out effort to apply knowledge and experience accumulated in years of aircraft and missile flight to get the best chance of mission success and flight safety from parts and components that already existed, or would be brought to the flight stage in, roughly, 2 years. The success of manned space flight required an extensive effort involving dedication of many individuals and their unstinting use of time, there being no sophisticated shortcuts to the disclosure of the many problems and the solving of these problems to assure success of each flight. Consideration of cost, manpower, or schedule were never allowed to influence any decisions involving mission success or flight safety.

Throughout the program, there proved to be a need for stringent attention to details of design, fabrication, quality control, testing and training; emphasis was placed on streamlining the failure analysis and corrective action procedures, incorporating on-the-spot failure analysis at the launch site.

Reliability and flight safety, although closely related, are not exactly the same thing. The former refers to the probability that a given mission will proceed to completion without mishap. This probability combined with the reliability of the escape system provides the overall flight safety or probability of crew survival. It may be pointed out that flight safety can be achieved by building a high reliability vehicle with little or no provisions for escape, as in the case of a commercial airliner, or by attaching a highly reliable escape system to an unreliable vehicle.

Two key design philosophies or guidelines can be postulated:

(1) No single failure shall cause an abort.

(2) No single failure during an abort will result in the loss of life of the crew.

Obviously certain items fall outside the scope of these rules. These are such passive subsystems as the ablation shield and the spacecraft structure as well as some large active elements having a background of high reliability such as the launch escape rocket.

What might be termed the Mercury approach to mission accomplishment and crew safety is outlined in the figures accompanying this report. It may be described conveniently under three main headings, the launch vehicle, the spacecraft and the operational procedures and philosophy.

The success of the mission and safety of the crew also depended on a number of other considerations such as the efficiency of the worldwide network of communications and the recovery operations, both of which are discussed in other papers.

Launch Vehicle

The launch-vehicle reliability and flight safety features are shown in figure 6–1. The main features indicated here are the use of an existing missile modified for Mercury requirements and augmented by a special pilot safety program and an abort sensing system. Although the following discussion centers around the Mercury-Atlas program, similar procedures were put into effect for the Mercury-Redstone program.

Existing Missile

The Atlas and Redstone missiles were chosen as launch vehicles because they were already far along in their development phases and would thus require only minor modification to adapt them to the Mercury requirements. This choice had a number of important implications as to reliability and crew safety, some favorable and some unfavorable. On the credit side, the particular vehicles chosen were well along on their development cycles, had considerable flight experience behind them, and had already demonstrated their abilities to meet the performance requirements. Another favorable feature of the Atlas launch vehicle was the fact that all engines were started, and satisfactory engine operation was verified, before lift-off.

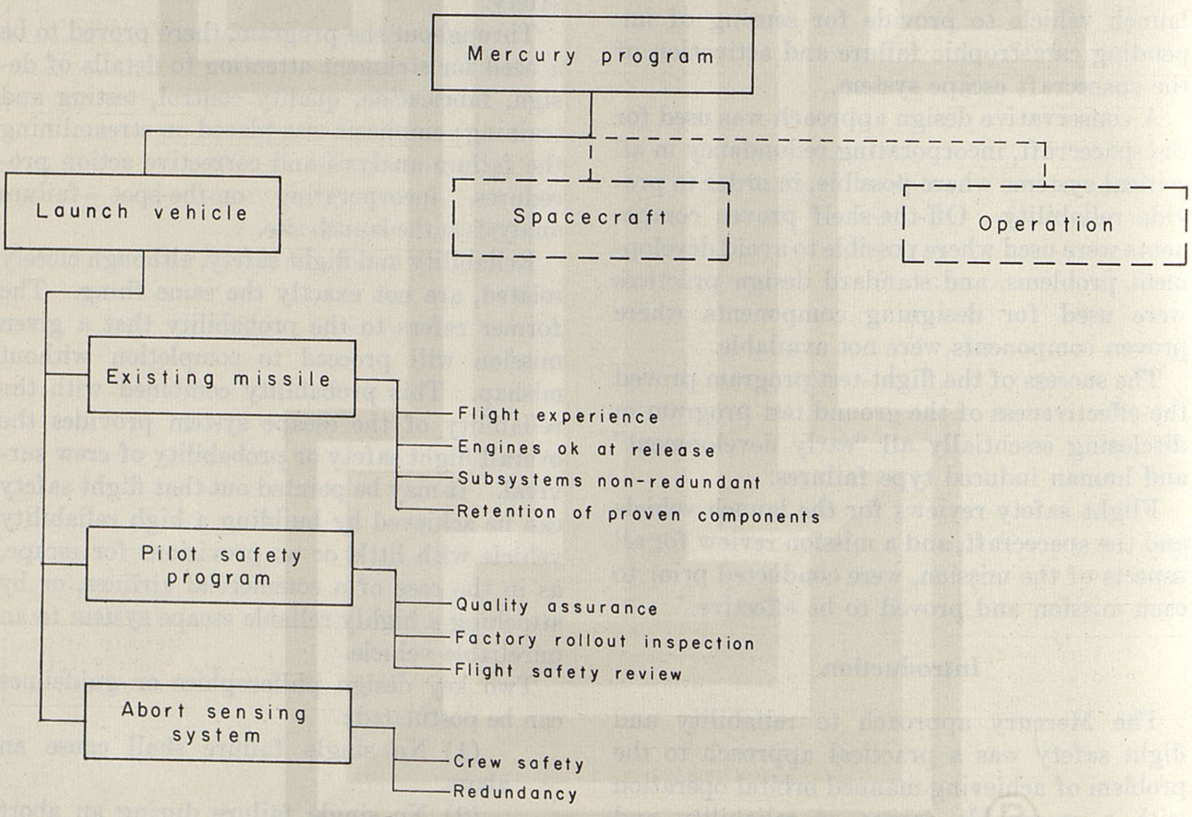

FIGURE 6–1.—Launch vehicle reliability and flight safety features.

Since the Mercury-Atlas vehicle was used as the launch vehicle for the orbital missions, the following discussion will be centered around this vehicle.

A determined effort was made to retain the proven components on the launch vehicle since the development of new components would have resulted in the loss of much of the advantage of using a developed launch vehicle.

Pilot Safety Program

The Pilot Safety Program (see fig. 6–1 and 6–2) was added in the Mercury Project to augment the reliability and safety of the basic Atlas system. This program was developed by the Air Force for the selection and preparation of the Atlas launch vehicles for manned Mercury flights. It was recognized that major design changes to increase the reliability potential of the basic design could not be accomplished within the life of the Mercury Project, and therefore special efforts would be necessary to make certain that the maximum reliability of which the design was capable would actually be achieved in Mercury operations. The program that resulted involved three parts, a Quality Assurance Program, a Factory Rollout Inspection Program, and a Flight Safety Review Program at the launch site.

The Quality Assurance Program consisted of two major areas: An educational program for contractor and sub-contractor personnel; and a critical parts selection program.

Training conducted by the contractor created an awareness of the importance of the Man-in-Space Program and the high reliability required of the Mercury-Atlas launch vehicle. High quality through careful workmanship was stressed.

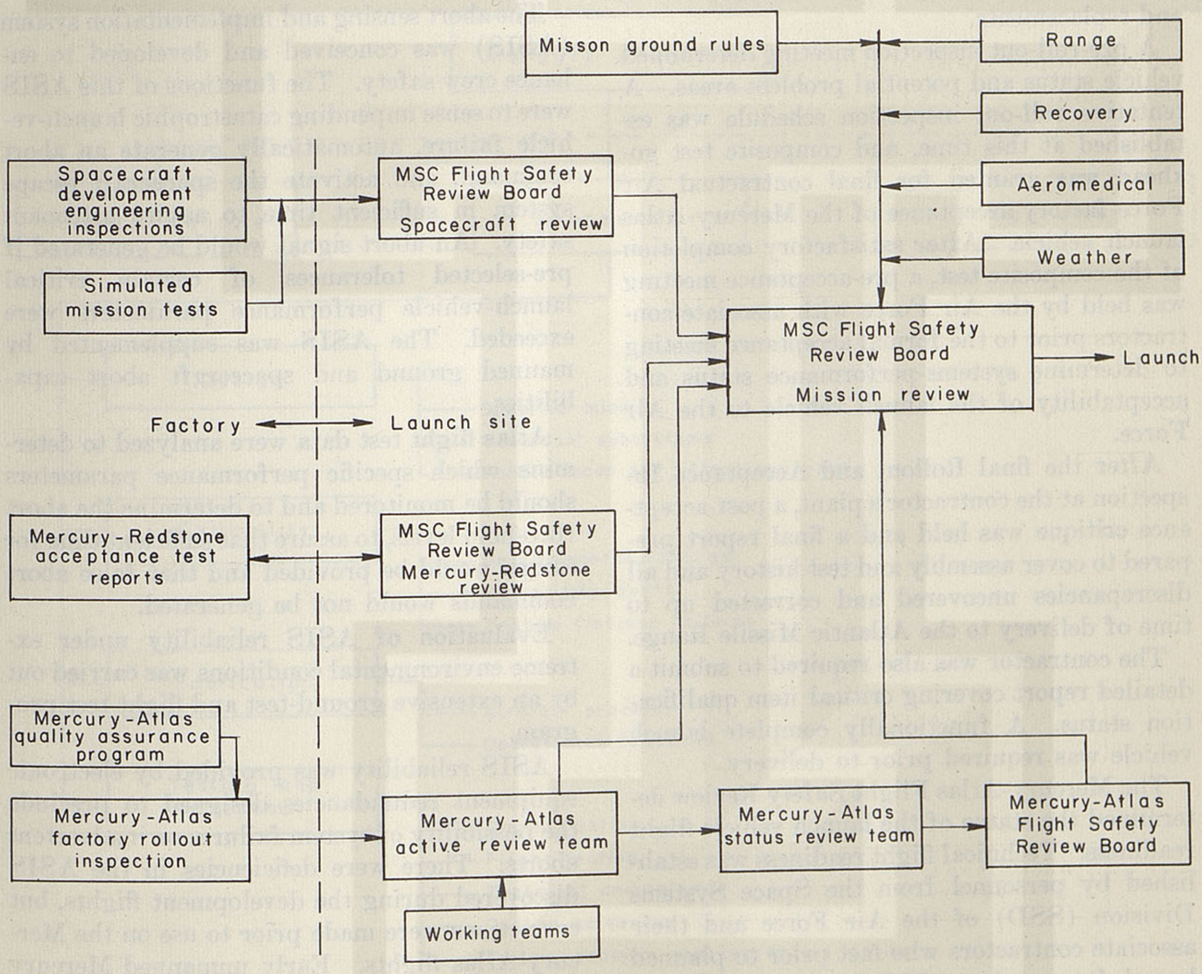

FIGURE 6–2.—Mission review activities.

The result of the critical parts selection program was the rejection of components and subsystems with excessive operating times, on nonstandard performance, or questionable inspection records. Choice of Mercury-Atlas launch-vehicle engines was limited to those standard Atlas engines whose performance parameters most closely met the exact specification requirements. Spare parts were also selected with the same care given to flight hardware. All selected units were specifically identified as accepted Mercury hardware and stored in a specially designated and controlled area.

The Factory Roll-Out Inspection assured that the Mercury-Atlas launch vehicle was complete, functionally acceptable, and ready for delivery. The technical roll-out inspection team consisted of specialists in the technical areas of each flight system. General launch vehicle progress was analyzed on a continuing basis, with special emphasis on hardware status and replacements.

A pre-roll-out inspection meeting determined vehicle status and potential problem areas. A tentative roll-out inspection schedule was established at this time, and composite test go-ahead was granted for final contractual Air Force factory acceptance of the Mercury-Atlas launch vehicle. After satisfactory completion of the composite test, a pre-acceptance meeting was held by the Air Force with associate contractors prior to the formal acceptance meeting to determine systems-performance status and acceptability of the launch vehicle to the Air Force.

After the final Rollout and Acceptance Inspection at the contractor's plant, a post-acceptance critique was held and a final report prepared to cover assembly and test history and all discrepancies uncovered and corrected up to time of delivery to the Atlantic Missile Range.

The contractor was also required to submit a detailed report covering critical item qualification status. A functionally complete launch vehicle was required prior to delivery.

The Mercury-Atlas Flight Safety Review determined the status of the launch vehicle flight readiness. Technical flight readiness was established by personnel from the Space Systems Division (SSD) of the Air Force and their associate contractors who met prior to planned launch for complete vehicle history review since arrival at AMR. The team determined that all possible efforts to insure a successful mission had been made and that the vehicle was in a state of technical readiness. Complete review of all facts yielded a "go" or "no-go" recommendation to the Mercury-Atlas Flight Safety Review Board, which was chaired by the Commander, SSD, for the manned orbital flights. This Review Board meeting was attended by NASA observers, including the NASA Operations Director and one of the astronauts. The findings of this board were subsequently conveyed officially to the NASA Operations Director in the Mission Review.

The total scope of the Pilot Safety Program resulted in expenditure of about twice the standard Atlas fabrication time, and more than three times the normal checkout time and attention.

Abort Sensing and Implementation System (ASIS)

The abort sensing and implementation system (ASIS) was conceived and developed to enhance crew safety. The functions of this ASIS were to sense impending catastrophic launch-vehicle failure, automatically generate an abort command, and activate the spacecraft escape system in sufficient time to assure astronaut safety. An abort signal would be generated if pre-selected tolerances of certain critical launch-vehicle performance parameters were exceeded. The ASIS was supplemented by manned ground and spacecraft abort capabilities.

Atlas flight test data were analyzed to determine which specific performance parameters should be monitored and to determine the abort threshold levels, to assure that sufficient time for escape would be provided and that false abort commands would not be generated.

Evaluation of ASIS reliability under extreme environmental conditions was carried out by an extensive ground-test and flight-test program.

ASIS reliability was provided by electronic equipment redundancies designed to preclude the possibility of system failures or inadvertent aborts. There were deficiencies in the ASIS discovered during the development flights, but corrections were made prior to use on the Mercury-Atlas flights. Early unmanned Mercury flights proved out the entire system; successful abort was initiated on the MA-3 flight, saving the spacecraft which was flown again on MA-4. There were no manned Mercury flights which required an abort action by the ASIS, nor were there any false ASIS abort signals.

ASIS was supplemented by the following manned abort capabilities:

(1) Off-the-pad aborts could be initiated by the test conductor, through direct electrical circuitry, until the vehicle had lifted 2 inches from the pad.

(2) From the point of 2-inch vertical ascent through the end of powered flight, an abort could be initiated through the Mercury Control Center (MCC) radio-frequency link.

(3) The mission could be terminated at any time throughout the entire powered flight by the astronaut.

(4) Indirect abort capability was provided the Range Safety Officer. The automatic airborne abort system could be activated by supplying a manual engine cut-off command. A 3-second airborne time delay was integrated with the airborne range safety command receiver to insure a safe separation of the spacecraft in the event that a command destruct signal became necessary.

Spacecraft

The size, complexity, and cost of the spacecraft and related operational activities including recovery precluded a program of using general flight testing to uncover design and systems weaknesses. It was necessary to produce the first and following spacecrafts with sufficient reliability to assure that each flight would complete its mission. The following discussion covers the reliability and flight safety features of the effort expended in Mercury to accomplish this result. The features are shown on figure 6-3 and may be described under the four head-

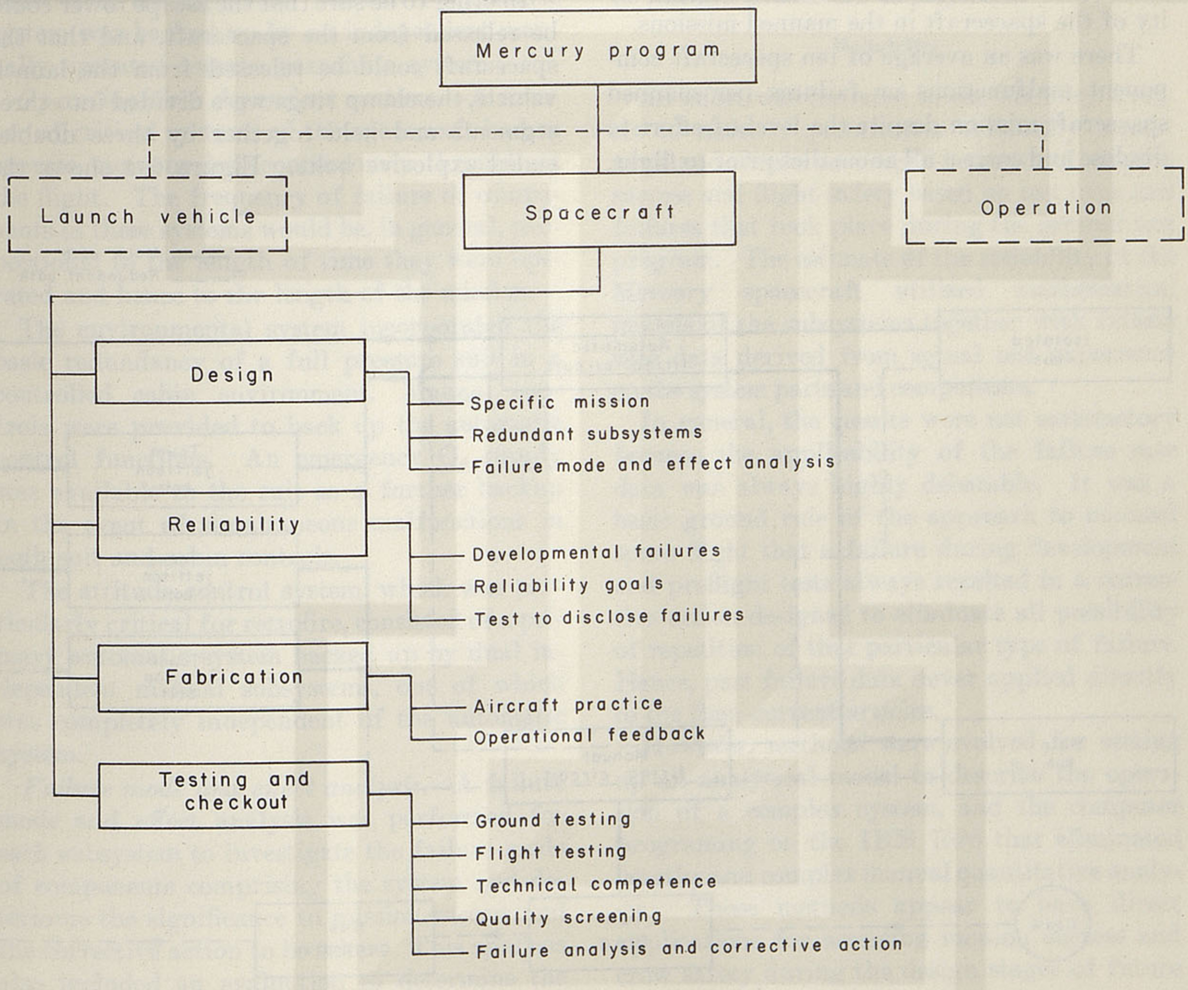

FIGURE 6-3.—Spacecraft reliability and flight safety features.

ings of design, reliability, fabrication, and testing and checkout.

Design

The spacecraft was designed specifically for manned orbital flight with virtually no background of applicable experience to serve as a guide. A very conservative design approach was adopted to provide redundancy in all critical subsystems where possible. The original design was required to provide for normal manned operation, unmanned operation, and operation with an incapacitated man aboard. Much of the redundancy, particularly in the smaller items such as explosive bolts, igniters, etc., was functional in both the unmanned and manned vehicles, but for manned flights the major subsystems such as the attitude control system and landing system relied on pilot operation of the backup mode; hence, the presence of the pilot substantially increased the reliability of the spacecraft in the manned missions.

There was an average of ten spacecraft component malfunctions or failures per manned spacecraft mission despite the level of effort to disclose and correct all anomalies prior to flight. However, in no case did these failures, some of which were critical, result in mission failure. The adopted design approach utilizing equipment redundancy and pilot back-up modes proved its effectiveness.

Insofar as reliablity and safety were concerned, components selected or fabricated for use in the subsystem were representative of the state-of-the-art at the time of the design freeze. Standard design practices were utilized for designing components for specific applications where proven components were not available.

The philosophy of designing redundancy into Project Mercury is best described by the following examples:

One-time-only operating devices.—A number of subsystems are required to operate only once during a mission, and thus the frequency of failure of these subsystems is independent of mission duration.

In order to be sure that the escape tower could be released from the spacecraft, and that the spacecraft could be released from the launch vehicle, the clamp rings were divided into three segments and held together by three double-ended explosive bolts. Figure 6–4 shows the

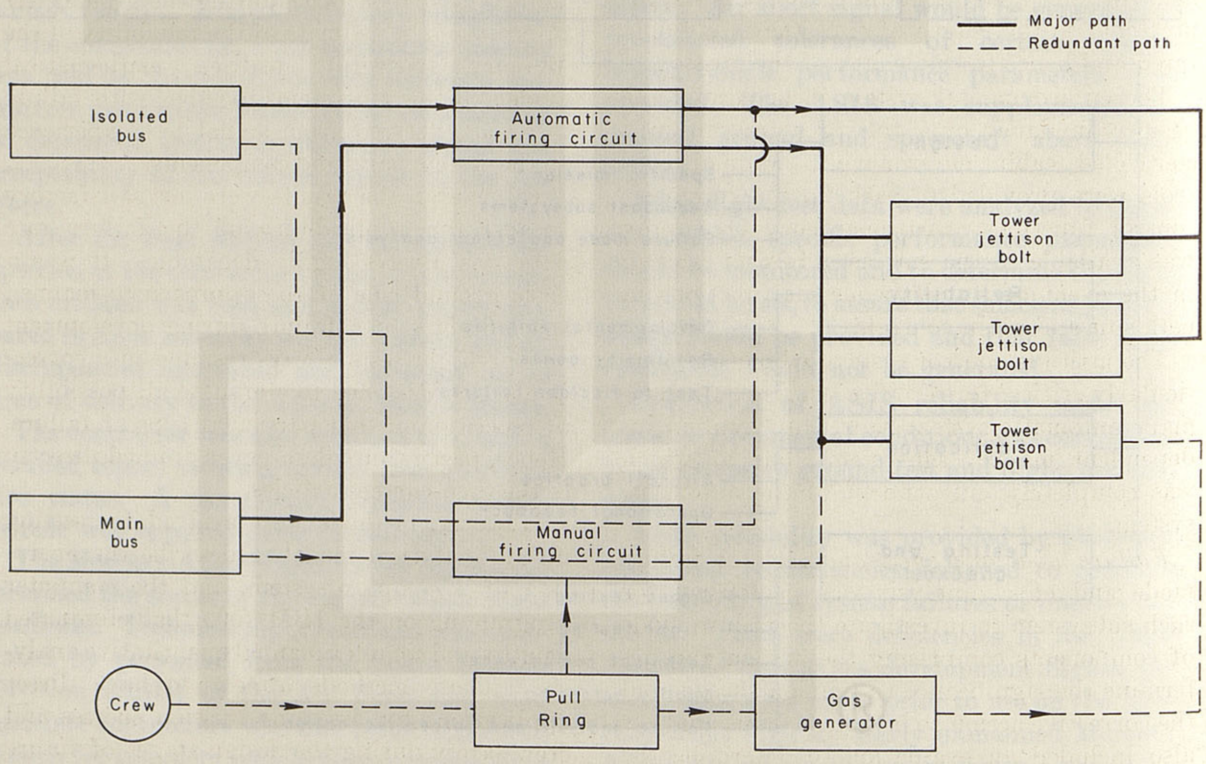

FIGURE 6–4.—Automatic and manual initiation of tower-jettisoning-bolt, pyrotechnics.

escape-tower clamp ring bolt-firing functional arrangement. Firing any end of any bolt could effect the release. The automatic system could fire one end of each bolt from one electric circuit and the opposite end of two bolts from a completely independent circuit; an astronaut manually operated backup could fire the opposite end of the third bolt through a percussion device, and in addition, could send electrical signals through the two automatic electric circuits.

For retroimpulse there were three solid fuel rockets with dual igniters fired by dual circuits. They could be initiated automatically, or by either astronaut or ground command. Only two of the three retrorockets were required to effect a satisfactory reentry.

The primary parachute system was fully automatic. It incorporated dual barostats, dual power sources, and manual backup of each main function in the sequence. The entire automatic system was backed up by an independent manually operated reserve parachute system.

Operating-time dependent systems.—A number of critical systems of the spacecraft had to operate more or less continuously throughout the flight. The frequency of failure of components in these systems would be, in general, proportional to the length of time they were operated and hence to the length of the mission.

The environmental system incorporated the basic redundancy of a full pressure suit in a controlled cabin environment. Manual controls were provided to back up the automatic control functions. An emergency O_2 supply was available to the suit as a further backup in the event of simultaneous malfunctions in both suit and cabin controls.

The attitude control system, which was particularly critical for retrofire, consisted of a primary automatic system backed up by dual independent manual subsystems, one of which was completely independent of the automatic system.

Failure mode and effect analysis.—A failure mode and effect analysis was performed for each subsystem to investigate the failure mode of components comprising the system and determine the significance to mission success and the corrective action to be taken. This analysis also included an evaluation to determine the action that should be taken in case the systems-performance indications displayed to the pilot and transmitted to the ground stations were in disagreement. An important consideration was the probability that the sensors and indicators may malfunction and thus incorrectly dictate the need of an abort.

A concentrated effort was made to identify single point failures; first, those which would in themselves be catastrophic or prevent subsequent operation; and second, those which would cause a premature operation.

An example of a subsystem revision resulting from this effort was the change in arrangement of the dual barostats that functioned to close the circuit to the parachute deployment sequence. Originally, the dual barostats were in parallel; a failure to the closed position of either would initiate the deployment sequence. The revision placed the barostats in series, thereby requiring both to fail closed in order to initiate premature deployment.

Reliability

An effort was initiated in the Mercury Project to make a quantitative reliability assessment and obtain an overall estimate of mission success and flight safety based on test time and failures that took place during the ground test program. The estimate of the reliability of the Mercury spacecraft utilized mathematical models of the subsystems together with failure rate data derived from actual test experience on the system parts and components.

In general, the results were not satisfactory because the applicability of the failure rate data was always highly debatable. It was a basic ground rule of the approach to manned space flight that a failure during development and preflight tests always resulted in a corrective action designed to eliminate all possibility of repetition of that particular type of failure. Hence, past failure data never applied directly to the then-current articles.

However, methods were evolved for setting up an analytical model to describe the operation of a complex system, and the computer programing on the IBM 7090 that eliminated lengthy and complex manual quantitative analysis. Those methods appear to have direct applications for assessing mission success and crew safety during the design stages of future space programs.

Mathematical models were used to some degree in the design stages of the Mercury Project. Catalogued values of failure rates that had been established by the manufacturers or various testing agencies as being representative of the random or statistical type of failure that predominates in fully developed parts comprised the inputs to these models. Reliability values obtained in this way tended to reflect the ultimate goal: that is, the minimum failure rate that may eventually be obtained with the design.

The first Mercury space flights with new systems could not be delayed pending statistically rigorous reliability tests to assure demonstration of reliability goals. The problem was therefore to decide, by a combination of engineering judgment, common sense, experience, and intuition, just when the last serious "early development" types and human-induced types of failure had been eliminated. The early development type of failure arose from design errors, interaction effects between parts and components, unanticipated environmental effects, or errors in estimating environments. The human-induced type were those associated with faulty fabrication, quality control, failure diagnosis, handling, installation, and carelessness.

As a result of the experience in the Mercury Project the role of numerical reliability assessment in manned space programs may be summarized as follows:

(1) It is desirable to specify an overall numerical reliability goal to insure that adequate attention is directed to reliability in the design stage. This goal should be apportioned or budgeted through a mathematical model down to the various subsystems and their components. The subsystem designer should be required to show that his subsystem is capable of absorbing the expected number of random or statistical type failures of parts without serious consequences or without exceeding his reliability budget.

(2) The logic flow diagrams which show functionally the systems sequence of action were especially useful since they represented primary and critical abort paths, crew inputs, and principal events. They reflected the basic ground rules relative to choice of alternate modes of operation and aborts. From these diagrams the effect of a component failure could readily be determined.

(3) Beyond this point the usefulness of formal quantitative reliability assessment procedures is debatable: the most effective approach from here on is to concentrate on establishing a testing program and quality assurance program that will assure detection and correction on all the unproven design and induced sources of system failure before flight.

Fabrication

Fabrication of the spacecraft was generally in accordance with the accepted aircraft production practices for small lots on the order of twenty articles. Air-conditioned clean room procedures were introduced in an effort to eliminate the introduction of contaminants or debris into components.

The results of operational experiences were fed back into the fabrication process by holding frequent Development Design Engineering Inspections (DEI). The purpose of the DEI was to assure that the Mercury spacecraft as engineered and manufactured was safe for manned flight. Emphasis was placed on attaining reliability and flight safety with existing Mercury hardware. To accomplish this objective, the DEI team was responsible for conducting suitable inspections for deficiences and initiating necessary corrective action. The DEI board was authorized to make final decisions on the acceptability of the spacecraft.

Preparatory to the DEI, the inspection team reviewed in detail engineering design, fabrication, and assembly, as well as component, system, and composite testing.

Testing and Checkout

Ground testing.—In addition to the standard type of qualification and acceptance tests, the following types of tests were conducted.

Demonstration tests: Demonstration tests were made to determine reliability, wherein several samples of each major subsystem were tested under simulated operational environments and duty cycles for a total operating time considerably longer than that of a single mission. The scope of these tests is shown in figure 6–5.

Major subsystems	Typical test time or firings
1. Environmental control system	1500 hrs
2. Automatic stabilization and control system	2000 hrs
3. Reaction control system - automatic	290 hrs
4. Reaction control system - manual	112 hrs
5. Horizon scanner	720 hrs
6. Landing and recovery	38 firings
7. Rockets	27-37 firings (ea. type)
8. Sequential system	400 cycles
9. Communications (tranceivers, audio center, transponders, beacons, etc.)	1000 hrs (ea. type)
10. Satellite clock	3000 hrs
11. Bolt, expl. clamp release	108-155 firings (ea. type)
12. Bolt, retrorocket release	106 firings
13. Battery (3000 w, 1500 w)	20 discharge cycles (ea. type)
14. Ejector, antenna firing	145 firings
15. Explosive egress hatch	67 firings
16. Inverter, static	4000 hrs. (ea. type)

FIGURE 6–5.—Spacecraft subsystems reliability tests.

The results of these tests were questionable since the equipment being tested did not always represent production-quality hardware. In addition, actual flight hardware was subject to conditions not contemplated in the reliability testing such as handling and shipping environments, installations in high density and crowded areas within the spacecraft adjacent to unrelated heat generating equipment, and contamination external to the subsystem as well as within the subsystem.

Safety margin tests: Safety margin tests were made wherein a number of component units were tested under progressively severe environments to determine the safety margin provided.

It was necessary for such tests as Project Orbit and subsystems tests at contractor's plant, followed by the intensive subsystems checkout at the Cape, to uncover weaknesses. These tests are discussed in the following paragraphs.

Ground test program: A continuous ground test program, using a complete spacecraft and identified as Project Orbit, was instituted at the contractor's plant about midway through Project Mercury. It became apparent early in the Mercury Project that malfunctions occurring at Cape Canaveral and in the flight made it imperative that design and fabrication weakness be disclosed as early as possible. A comprehensive test program was started in which, to the greatest degree possible, the mission was simulated in real time and included orbital heating and near-vacuum effects. Obviously zero g effects, launch time and vibration, explosive devices, launch escape rocket, tower and spacecraft separation, exposure of the ablation shield to reentry temperatures, parachute deployment, and landing could not be duplicated. However, cabin environment and operation of time dependent subsystems under normal and emergency cabin environment were closely simulated. The continuous aspect of this program conducted in an altitude chamber with all systems operating as they would in a mission not only disclosed the weaknesses but validated equipment revised as a result of the malfunctions. Consequently, the test demonstrated the performance of up-to-date configurations.

The tests were very effective in disclosing design weaknesses associated with interface problems, time dependent failures, and thermal balances involving heat sinks and heat removal. A typical example of the usefulness of Project Orbit is discussed.

A revision in the gyro design resulted when, during the operation of the autopilot under an emergency mode (decompressed cabin), a failure in the gyros caused by decreased heat dissipation under vacuum conditions was disclosed. The lubricant vaporized, and there was a breakdown in insulation windings. The problem was resolved by changing the lubricant to one having a lower vapor pressure, and by using an insulation that maintained its dielectric characteristic when subjected to high temperatures.

Spacecraft subsystems tests: Spacecraft subsystems tests at the contractor's plant were followed by extensive tests at Cape Canaveral. Altitude sensitive systems were tested in an altitude chamber at the Cape since such tests were not made at the contractor's plant for each spacecraft.

Flight testing.—Contributing much to the success of Project Mercury was the flight test program. Each flight of this test program was designed to qualify equipment and procedures for succeeding flights as well as ultimately for the manned orbital flights. Any malfunctions that occurred in a flight were analyzed, and corrected prior to the next flight. These early flights included (1) Beach Abort for qualifying the launch escape and landing system; (2) the Little Joe flights; (3) the Mercury–Atlas unmanned ballistic flights for qualifying the structure and ablation shield under severe reentry conditions, (4) the ballistic Mercury–Redstone unmanned, primate, and manned flights, and (5) the Mercury–Atlas unmanned and primate orbital flights.

The manned orbital flights progressed in a logical manner from a 3-orbit mission to a 22-orbit mission.

Technical competence.—A very important feature of the Mercury approach to flight safety was the assignment of personnel with a high level of technical competence to the performance and monitoring of all preflight tests and preparations at the launch site. Senior engineering personnel, in many cases key members of the original design team, moved to the launch site and developed the launch preparation procedures. This high level of competence also extended into the quality control and inspection areas at the launch site.

Quality screening.—The Mercury Project has featured extremely tight quality screening for deficiencies during all preflight checkout operations. This was accomplished by providing a system for effectively reporting unsatisfactory conditions to the contractor and to NASA management, to obtain conclusive corrective action, and to eliminate irregularities and deficiencies which adversely affect the spacecraft program. These anomalies were recorded on forms noted as Unsatisfactory Reports (UR's).

Failure analysis and corrective action.—The effectiveness of the contractor's failure analysis and corrective action program was evolutionary and improved considerably as the project went through its transition period from unmanned to manned flight. Later in the program, it became apparent that a streamline procedure was necessary for failure diagnosis and corrective action to assure effectivity in subsequent spacecraft. In many cases joint contractor-MSC teams analyzed a failure on-the-spot, or hand-carried the failed part to the supplier where a laboratory analysis of the failure was made.

In addition to individual failure reports on all failures, the contractor maintained an up-to-date status of all failures, submitting an IBM tabulation summary to MSC monthly. This tabulation included all unresolved failures, and was used to point out critical and recurrent problems.

Operations

Simulated Flights

There were several features and practices in the Mercury operation that are worth mentioning in connection with reliability and safety. A great deal of attention was given to rehearsals and simulations of complete missions prior to each flight. These simulations were made extremely realistic. They not only served to verify the feasibility of planned procedures and provide crew practice for the expected flight plan, but also included a wide range of emergencies deliberately introduced to show up areas where improved planning might be needed to eliminate all possibility of confusion or indecision.

Interface Control

With different groups responsible for the launch vehicle and the spacecraft, there was need for very special planning and procedures to insure proper handling of interface problems. It was found necessary in the field to establish a joint inspection team charged with the responsibility for witnessing all mating and other interface activities, measuring and verifying the adequacy of all physical clearances, inspecting all structural joints and electrical connections, and assuring that no debris was left in critical areas. Adequate access ports for field inspection were found to be an absolute requirement.

Special procedures were established for maintaining and periodically distributing one and only one official interface wiring diagram, reflecting the exact current status of the wiring on the vehicle at specified dates.

Flight Safety Reviews

The final item on figure 6–6, Flight Safety Reviews, deals with the problem of determining that the launch vehicle and spacecraft were in fact ready for launch. These activities are covered in figures 6–2 and 6–7. In Mercury, the philosophy was adopted that a launch would not take place with any unresolved difficulty. To insure this, preflight launch vehicle readiness and spacecraft readiness review meetings were set up. In these meetings, representatives from engineering, operations, flight safety, astronauts, and Cape inspection reviewed in detail with the specialists responsible for the checkout of each system, all malfunctions observed in the system, and all changes and corrections made. Two sets of contractor failure records were maintained: first, a segregation of failures from all testing into specific subsystems; second, a file of all failures associated with subsystems of a specific spacecraft. From these records, it was possible to determine any general weaknesses and to review the case histories of critical areas in any specific spacecraft. These data, together with the unsatisfactory reports (UR's) and record of anomalies occurring in the subsystems checkout recorded by MSC personnel at the Cape provided a major input in these meetings.

These detailed meetings on the major pieces of equipment were followed by a Final Mission Review meeting. This meeting provided a final

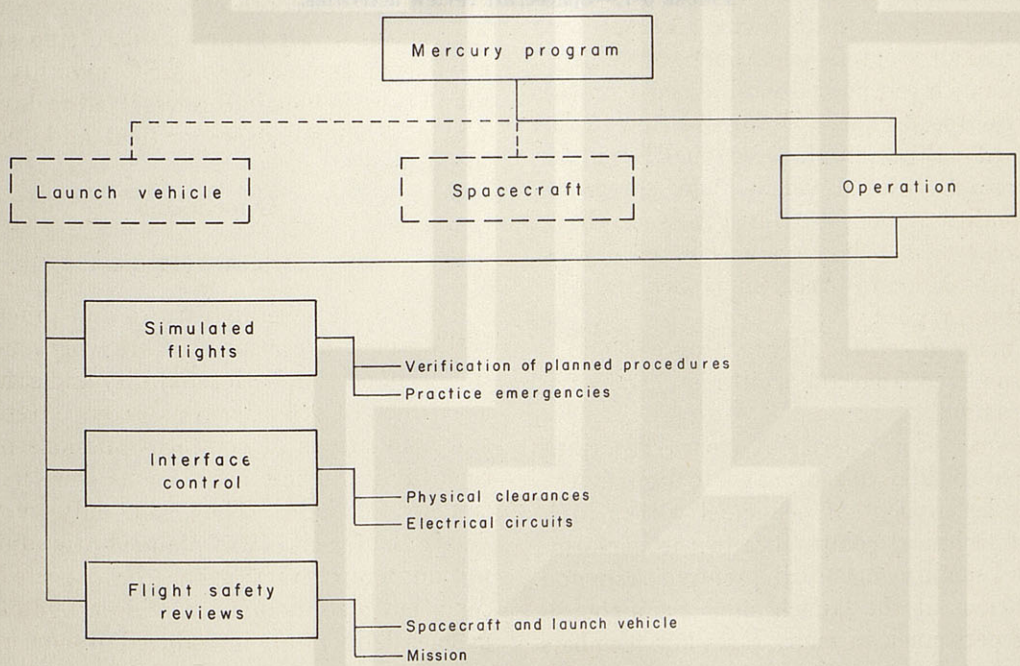

FIGURE 6–6.—Operational reliability and flight safety features.

confirmation of launch-vehicle and spacecraft readiness and established the readiness of the range, recovery, weather, and aeromedical elements.

These operating procedures were very effective in concentrating the attention of the best qualified technical talent available on the detailed engineering problems of each vehicle.

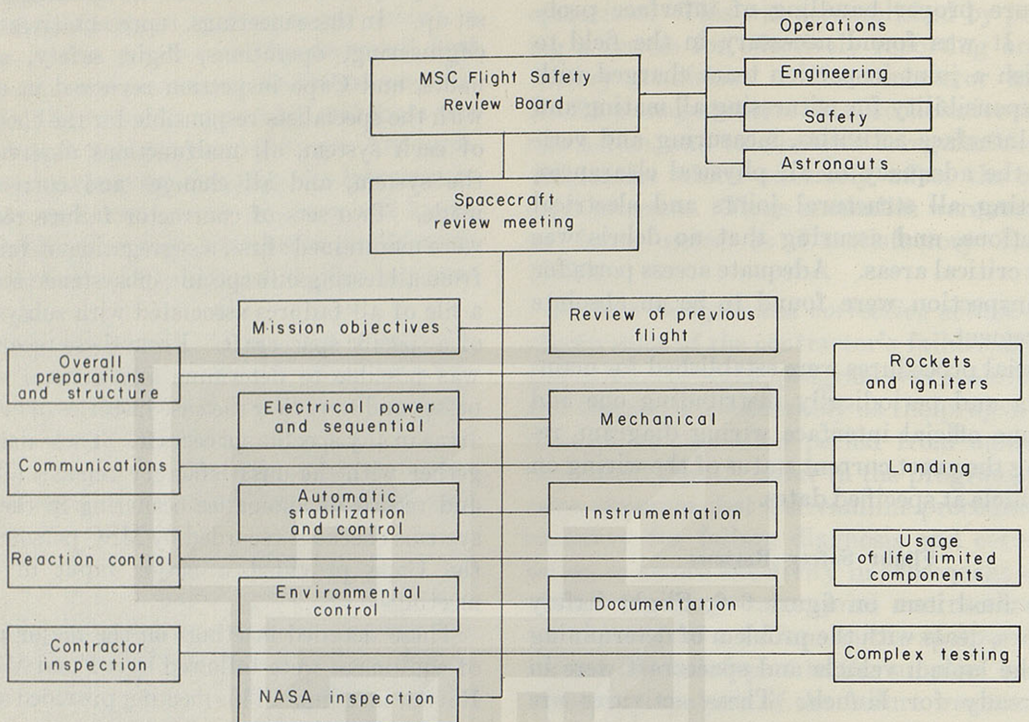

FIGURE 6-7.—Spacecraft review activities.

II

MISSION SUPPORT DEVELOPMENT

7. TRAJECTORY ANALYSIS

By JOHN P. MAYER, *Asst. Chief for Mission Planning, Flight Operations Division, NASA Manned Spacecraft Center;* and CARL R. HUSS, *Flight Operations Division, NASA Manned Spacecraft Center*

Summary

A description of the mission analysis studies conducted for Project Mercury is given along with specific examples for the various mission analysis phases.

Aborted mission studies constituted about 90 percent of all mission-analysis studies conducted. These studies were necessary from a flight-safety standpoint and are considered equally applicable to future manned spacecraft projects. It was found that the basic mission design must be chosen in a flexible manner so that consideration can be given to the changes in mission constraints. Real-time computing has proved extremely valuable in Project Mercury; however, consideration must be given to changes in mission operational plans which cannot be effectively included in the Real Time Computer Complex.

Introduction

The mission-analysis effort in Project Mercury was conducted in several phases leading up to the flight missions. These phases include the mission analysis supporting the systems design of the spacecraft, the basic operational design of the Mercury missions based on mission requirements and objectives, detailed operational mission analysis for each specific flight, and the formulation of the mission logic to be included in the computer used for inflight real-time control of the missions.

Mission Phases

In figure 7–1 are shown the important phases of mission-analysis studies. In the early mission-analysis phase, the analysis was specifically for use in spacecraft system design. For example, the maximum loads and heating conditions were determined for structural design, and the spacecraft propulsion performance requirements were determined leading to the design of the retrorocket system. After the spacecraft systems were essentially designed, the mission-analysis effort shifted to the operational phase. In this phase the system design was reasonably fixed and the detailed mission design was then accomplished by taking into account all of the constraints, including spacecraft, launch-vehicle, and operational constraints. The objective in this phase is to design a mission within the capabilities of the actual spacecraft system developed. In this phase of the mission some feedback into system design was made, although these were small changes since the early design proved to be sound.

The next mission analysis phase was in the design of specific missions. In this case the mission analysis was specialized to handle the aspects of a particular mission by using the actual performance characteristics of the launch vehicle and spacecraft being used. This phase also included the analysis for the particular operational mission objectives and ground rules developed for these missions.

The next phase was the real-time mission-analysis phase, which started at the beginning of the launch countdown and lasted until the

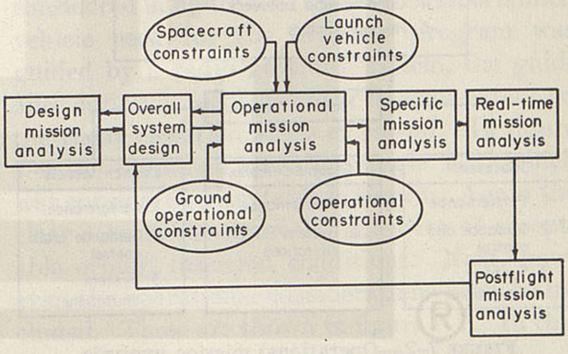

FIGURE 7–1.—Mission analysis sequence diagram.

vehicle was recovered after the mission. In this, calculations were accomplished in real time by a computer; however, the logic and equations used in this computer were developed in the preceding operational mission-analysis phase. Although every effort was made to anticipate all the possibilities that could affect the flight and include them in the real-time computer program, these possibilities were never fully established. Therefore, mission-analysis experts were used as flight controllers and also performed auxiliary computing using off-line computers other than those used in the real-time computing complex during the missions.

The next mission-analysis phase was a post-flight analysis phase in which the information obtained from actual flights was fed back into the plans for future flights and, in some cases resulted in system modifications to the spacecraft, the launch vehicle, and the ground support system.

Some specific examples of mission constraints affecting the analysis are shown in figure 7–2. Some of the spacecraft constraints that must be considered are the performance of the spacecraft propulsion system, the spacecraft control system accuracies, and other system limitations. Some of the ground complex constraints to be considered are performance (which includes the effects of the locations of command stations and command ranges) and system limitations. Constraints involving the launch vehicle which had to be considered were performance, guidance accuracies, and systems limitations. In Project

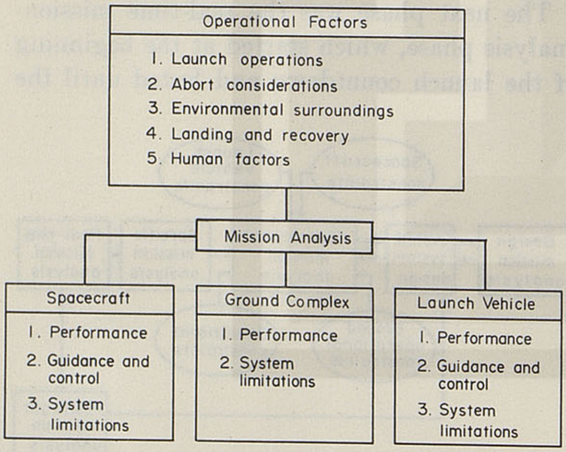

FIGURE 7–2.—Operational mission analysis.

Mercury the systems limitations of the launch vehicle included heating and load restraints and the guidance radar look angle constraint.

The operational constraints to be considered in the area of launch operations are range safety limits, abort considerations, environmental considerations, landing and recovery considerations, and human factors. Some of the environmental factors that were considered were the effect of atmospheric and geophysic constraints and winds. Consideration had to be given to recovery and landing constraints for both normal and aborted missions and, in all cases, the human tolerances to acceleration loads and motions were considered.

Abort considerations resulted in about 90 percent of the mission-analysis studies. Studies were made to provide flight controllers with the information as to when to initiate aborts for maximum pilot safety. Studies were also made to determine allowable tolerances in order to obtain safe miss distances between the launch vehicle and the spacecraft and acceptable lateral loads. Also of importance were the studies to determine the abort recovery areas for all phases of the flight.

In order to illustrate some of the techniques used and the results accomplished in the mission-analysis area, a few specific examples from each phase will be discussed.

Design Mission Analysis

One example of the work performed in the advanced mission analysis phase is illustrated by a study of the immediate post-abort conditions. The selection of the escape-rocket offset involved a compromise between high lateral loads and low miss distances between the space craft and the launch vehicle in the high-dynamic-pressure abort phase of launch. For low offset values the probability of exceeding high lateral loads was low; however, the probability of obtaining low miss distances was high. For high values of the offset the opposite is true. Thus, the selection of the offset was made on the basis of minimum combined probability of occurrence of either events. In figure 7–3 the combined probability of exceeding either a dangerous lateral load or an unacceptable miss distance is shown plotted against the escape-rocket offset.

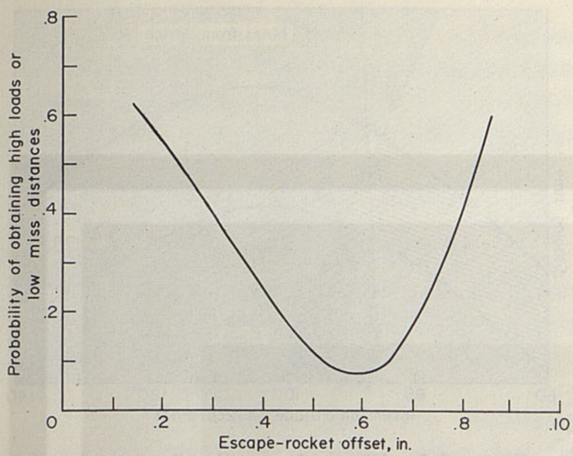

FIGURE 7-3.—Selection of escape-rocket thrust offset.

Operational Mission Analysis

A typical example of the operational mission analysis was in the selection of the Mercury orbital elements. The orbital inclination which governed the ground track for Project Mercury was selected because the network facilities established prior to Mercury could be used to good advantage, reentries for the first three orbital passes and the 16th to the 18th passes occurred over the United States, and the orbital ground track fell within the temperate region of the world. In addition, the specific Mercury inclination was affected by launch-abort recovery considerations.

The orbital altitude and shape of the Mercury orbit were selected based on launch-vehicle performance, accuracy, and abort operational considerations. These considerations are illustrated in figures 7-4 to 7-7. In figure 7-4 the orbital lifetime is shown plotted against apogee altitude for given perigee altitudes. For Project Mercury it was desired to have minimum lifetime of 36 hours for a 24-hour mission. Since the atmospheric densities at orbital altitudes were not well-defined at the time Project Mercury was initiated, it was believed that a conservative value for density must be used for estimating lifetime. The density used in this figure is considered to be a 3σ, or very conservative, dense atmosphere. From figure 7-4 it can be noted that for an adequate lifetime a circular orbit at an altitude of 105 nautical miles could have been selected, or an elliptical orbit having the same lifetime could have been selected, for example, an orbit having an 80-mile perigee and a 170-mile apogee.

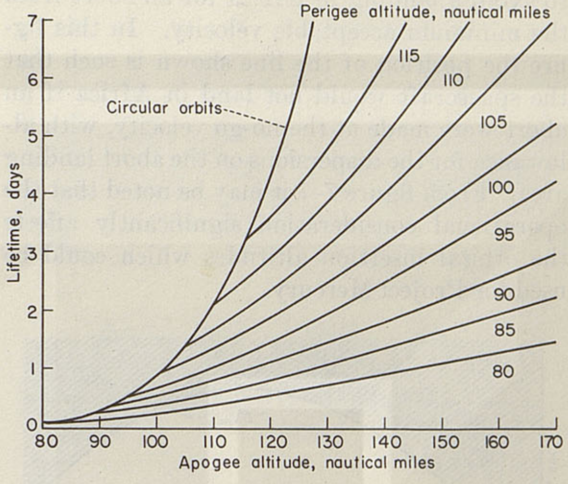

FIGURE 7-4.—Minimum lifetimes for elliptical orbits.

The next constraint to be considered is that of launch-vehicle performance. In figure 7-5 the staging time is shown plotted against the insertion or perigee altitude. The curves shown are given for a constant orbital lifetime; that is, the apogee altitude decreases as the insertion altitude increases. For a constant insertion altitude the performance, or excess velocity available above that required (ΔV_{min}), increases with staging time until it reaches a peak value. For greater staging times the performance decreases. The minimum acceptable performance curves are shown in figure 7-5. The increment of velocity ΔV that defines the acceptable performance is the difference between the velocity at fuel depletion and the planned velocity. Therefore, all of the clear area in the figure would represent acceptable orbital insertion altitudes.

The launch-vehicle guidance accuracies are considered in figure 7-6. Since the Atlas launch vehicle used for the Mercury program was guided by a radio guidance system, the guidance accuracy was dependent to some extent on the radar elevation angle at cut-off. In figure 7-6 the minimum elevation angle E_{min} which was considered acceptable is shown. Again the clear area in the figure is indicative of acceptable orbital insertion conditions. Next, however, the operational considerations must be included. These are shown in figure 7-7. In this case the operational consideration which affected the orbital conditions was the requirement to avoid a landing in Africa for an abort from the minimum acceptable velocity. In this figure the position of the line shown is such that the spacecraft would not land in Africa if an abort were made at the no-go velocity, with allowance for the dispersions on the abort landing area. From figure 7–7 it may be noted that the operational consideration significantly affects the orbital insertion altitudes which could be used for Project Mercury.

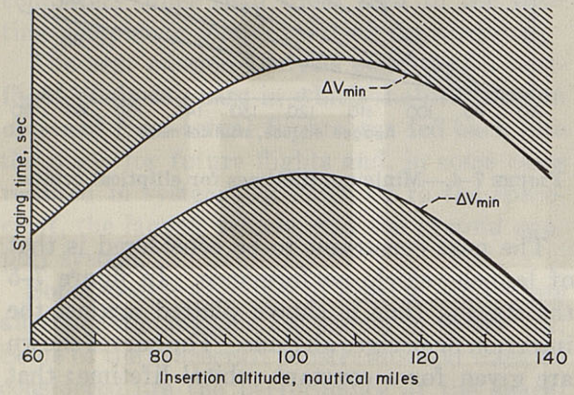

FIGURE 7–5.—Effect of launch-vehicle performance.

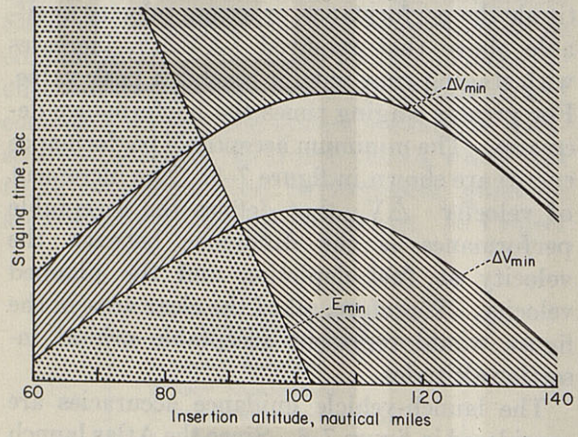

FIGURE 7–6.—Effect of launch-vehicle guidance.

As operational experience was gained in Project Mercury flights, confidence and knowledge in the systems made it possible to reduce to some extent, the original guidance and performance constraints. For example, the minimum elevation angle was reduced after obtaining a better understanding of the effects on guidance accuracy from operational experience with the guidance system.

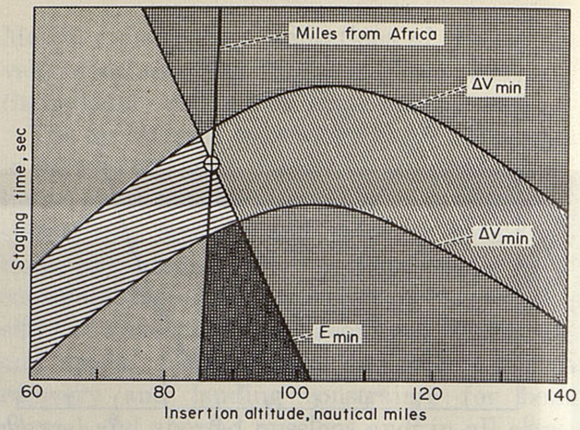

FIGURE 7–7.—Effect of operational constraints.

Specific Mission Analysis

A considerable mission-analysis effort is made in the design of each specific Mercury flight. Included in this effort are detailed trajectory calculations for the mission, dispersion calculations, calculations concerning aborts during all phases of the mission, and calculations of retrograde time to be used in the mission. When the flight day arrived, special mission-analysis studies were performed to support the flight. These studies included evaluating the wind effects on the loads on the launch vehicle and determining the landing areas of the spacecraft in aborted missions based on actual wind profiles. In figure 7–8 the effects of the actual winds on the abort landing areas at various times of the flight are shown for the MA–9

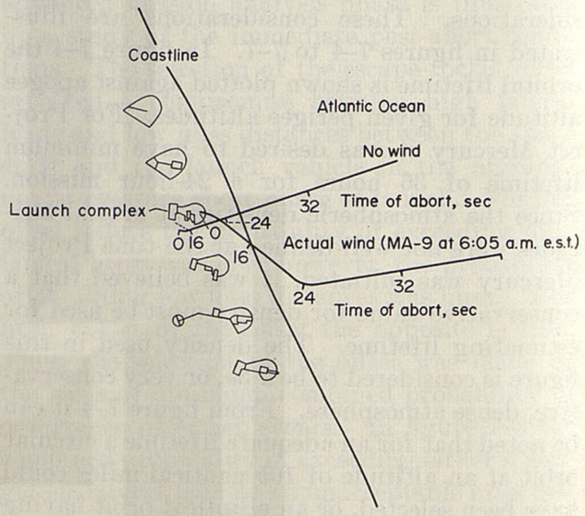

FIGURE 7–8.—Effects of actual winds on MA–9 abort landing.

mission. These calculations were made to enable the recovery forces to be positioned prior to the launch such that they could most easily make an emergency recovery should abort occur.

Real-Time Mission Analysis

General Computing Requirements

Real-time computing has proved very valuable in Project Mercury for use in flight control and monitoring. The basic computing requirements in real time are as follows:

(*1*) *Powered flight.*—Pertinent trajectory parameters were computed in order that the status of the launch could be monitored for any indication of an impending abort. The cut-off velocity was used to determine the acceptability of the orbital parameters based on preplanned criteria. In addition, landing points for possible aborts and radar-acquisition data were computed.

(*2*) *Aborted missions.*—For aborted missions the computer must be programed to select a target recovery area and if necessary compute the time for retrofire to land within this area.

(*3*) *Orbit.*—In this phase the orbital parameters were predicted with sufficient accuracy to establish the minimum lifetime of the orbit, to predict the retrofire time to land in normal and contingency recovery areas, to determine spacecraft orbital position, to determine acquisition data for all radar sites, and to predict the time of landing for use by recovery forces.

(*4*) *Reentry.*—During reentry the computer program recalculates and updates the landing point and time of landing, based on conditions at retrofire, in addition to predicting acquisition data for reentry radar stations.

Example of Go—No–Go Computation

The computation of the go—no–go parameters was probably the most important of the real-time computations. The selection of the Mercury go—no–go criteria which were used in the real-time computing program is shown in figures 7–9 to 7–11. In figure 7–9 the minimum energy for an acceptable Mercury mission is illustrated. The flight-path angle at insertion is plotted against the insertion velocity. The minimum acceptable orbit was defined as that orbit in which the spacecraft could safely complete one orbital pass and land. Because of the critical flight safety nature of the problem, the minimum orbit was selected on the basis of a very conservative drag coefficient C_D and atmospheric density ρ. The symbol, $(C_D\rho)_n$, shown in figure 7–12 has been normalized and

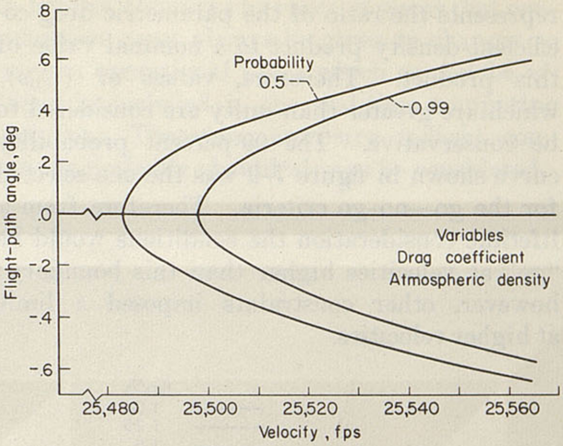

FIGURE 7–9.—Determination of minimum acceptable orbit.

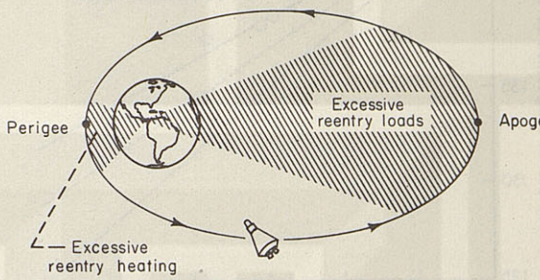

FIGURE 7–10.—Determination of maximum acceptable orbit.

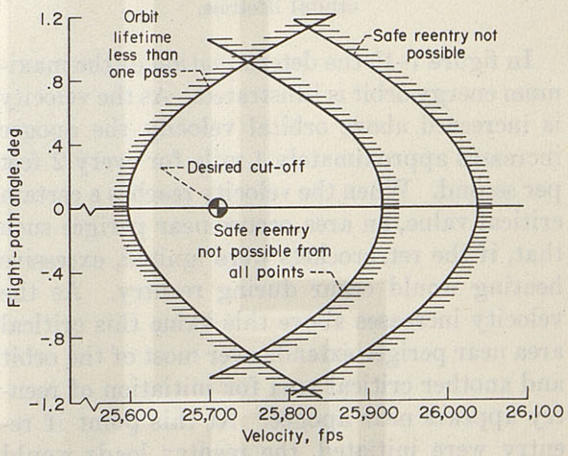

FIGURE 7–11.—Operational go—no–go orbital-insertion criteria.

represents the ratio of the parametric drag coefficient-density product to a nominal value of this product. Therefore, values of $(C_D\rho)_n$ which are greater than unity are considered to be conservative. The 99-percent probability curve shown in figure 7–9 was the one selected for the go—no–go criteria. Therefore from a lifetime consideration the conditions would be "go" at velocities higher than this boundary; however, other constraints imposed a limit at higher velocities.

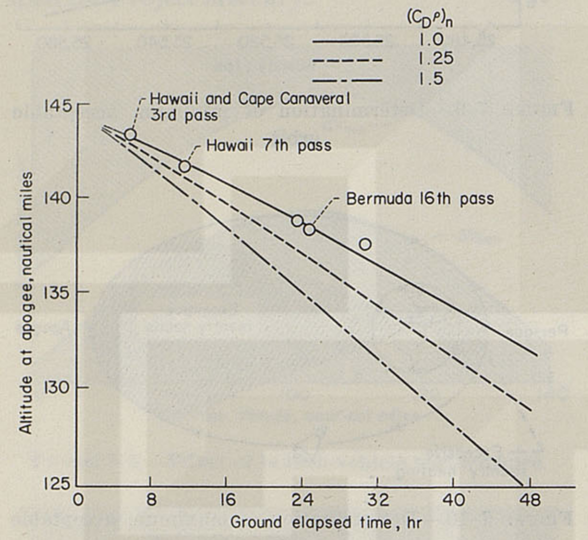

FIGURE 7–12.—Effects of actual atmosphere on MA–9 orbital lifetime.

In figure 7–10 the determination of the maximum energy orbit is illustrated. As the velocity is increased above orbital velocity the apogee increases approximately 1 mile for every 2 feet per second. When the velocity reaches a certain critical value, an area occurs near perigee such that, if the retrorockets were ignited, excessive heating would occur during reentry. As the velocity increases above this value this critical area near perigee extends over most of the orbit and another critical area for initiation of reentry appears near apogee. At this point if reentry were initiated, the reentry loads would become excessive. As the velocity is further increased, a velocity is reached in which these critical areas cover the entire orbital range and a safe reentry would not be possible from any point in the orbit. The operational go—no–go criteria that resulted from these constraints are shown in figure 7–11 where the flight-path angle at cut-off is plotted against the insertion velocity. The region for a minimum acceptable orbit lies within the boundaries shown. For all Project Mercury missions the cut-off velocities were well within the safe boundaries. For the MA–9 mission, for example, the cut-off occurred within the boundary of the symbol shown in this figure.

As was previously stated, some auxiliary computing was performed during each mission outside of the real-time computers. An example of this auxiliary computing is shown in figure 7–12 where the effects of the actual atmosphere on the orbital lifetime of the MA–9 mission are shown. In figure 7–12 apogee altitude is plotted against time. Because of the length of the MA–9 mission and the uncertainty of the density of the actual atmosphere on the day of this flight, it was thought necessary to attempt to determine the variation of the actual atmosphere from that used in preflight computations. This calculation was necessary in order to commit the mission to completing 22 passes at a predetermined time during the flight. The lines shown in the figure are for precalculated atmospheric densities which varied from that of the assumed atmosphere. The symbols in this figure indicate the actual apogee obtained during the flight and also that the actual atmosphere was very close to that used in the preflight computations. The actual orbital lifetime for the MA–9 mission would have been about 4.7 days if a reentry were not initiated using the retrorockets.

Concluding Remarks

The operational experience obtained in mission-analysis studies for Project Mercury has proved valuable for application to other manned space-flight programs. Aborted mission studies constituted about 90 percent of all the mission-analysis studies conducted for Mercury. Although the results of these studies were not required operationally, the amount of effort spent on abort studies is necessary from a flight-safety standpoint and will be equally applicable to future manned space projects. It is also evident that the basic mission design must be chosen in a flexible and manner so that consideration can be given to changes in the spacecraft launch vehicle or operational constraints. Real-time computing has proved extremely valuable in Project Mercury; however, it seems that consideration must always be given to changes to mission operational plans which cannot be effectively included in the real-time computing complex. Therefore, auxiliary inflight computing probably should always be considered.

8. WORLDWIDE NETWORK SUPPORT

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Summary

Because the Mercury orbital flight program required effective ground control during the unmanned and manned phases, a worldwide tracking and telemetry network was developed. Early in the project, the requirements for the network in terms of systems, installation, site locations, testing, and training for network personnel were established. Maximum utilization was made of existing facilities, but additional stations had to be implemented because of a strategic need at certain points along the orbital ground track. In addition to the telemetry and tracking facilities, two important centers were established, those of the Mercury Control Center, which was the focal point for all flight control activities, and the Computing and Communications Center. System reliability and provision for ease of maintenance were primary guidelines during the network implementation. Because of the unique spacecraft tracking task, an acquisition aid device was developed to assist in the location and tracking at first contact with the spacecraft. As the telemetry, tracking, and computation functions of the network were being installed, the network was staffed to support even the early ballistic flight program. As the scope and complexity of the missions increased, the network was expanded and modified to accept the changing and more demanding flight control and monitoring requirements. In addition to the tracking and data reception capabilities of the network, a multi-frequency air-to-ground reception and remoting provision was necessary during the manned flight program. A requirement had been established to provide continuous tracking and communications during the launch phase, as well as voice communications with the astronaut within maximum prescribed time intervals. Throughout the Mercury-Atlas orbital flight program, the Mercury Worldwide Network provided adequate and timely support in each of its charged responsibilities. Voice communications, telemetry, and tracking were satisfactory for effective flight control and monitoring, and the computation and data handling facilities provided timely support during the critical retrofire and reentry phases of each of the manned orbital flights.

Introduction

Meeting mission objectives required that a worldwide tracking and ground instrumentation system be developed to provide a continuous flow of information to be used for mission control. The intent in this paper is to describe the evolution of the network in support of the various Mercury missions. Specifically, the paper discusses the development of network requirements and systems; installation, test, and training; the network configuration and later changes made in response to mission requirements, operations, and performance.

Development of Network Requirements

Approach

The task of implementing a tracking and ground instrumentation system was given to the NASA Langley Research Center (LRC). LRC formed the Tracking and Ground Instrumentation Unit to manage and direct this effort. This unit in turn utilized industrial firms to assist in determining the approach to be taken in meeting the requirements in certain critical areas and to augment the NASA team.

Basic Requirements

Basically, network systems were required to provide all functions necessary for ground control and monitoring of a Mercury mission from launch to landing. The function of the network was to end when the spacecraft had landed and the best possible information on the location of the landing point had been supplied to the recovery teams.

At the outset, the following functional requirements were established:

(1) Provision of adequate tracking and computing to determine launch and orbital parameters and spacecraft location for both normal and aborted missions.

(2) Voice and telemetry communications with the spacecraft with periods of interruption not to exceed 10 minutes during the early orbits, contact at least once per hour thereafter, and communications to be available for at least 4 minutes over each station.

(3) Command capability to allow ground-initiated reentry for landing in preferred recovery areas and to initiate abort during critical phases of launch and insertion.

(4) Ground communications between the ground stations and the control center.

Safety of the Mercury spacecraft and its occupant was made a dominant consideration. Speed and efficiency of installation were essential to meet the planned operational dates. Although no compromises with safety were made, economy was an important consideration in the overall plan.

Selection of Stations

Stations were selected on considerations of the flight plan and on the character of the spacecraft electronic systems consistent with the basic requirements. Because of factors relating to the earth's rotation and the lack of suitable geographic locations, certain compromises had to be made in selecting the total number and locations of the stations required for a three-orbit mission. These compromises resulted in gaps, primarily on the third pass, greater than the desired 10 minutes. For stations selected, see figure 8–1.

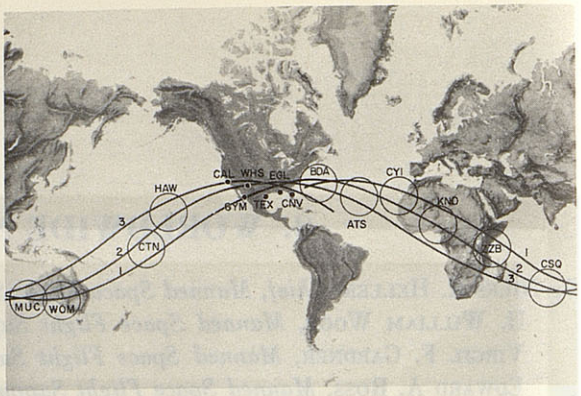


FIGURE 8–1.—Map showing the locations of the selected Mercury stations.

Two Centers were also required:

The Mercury Control Center (MCC), to be located at Cape Canaveral, was to provide equipment necessary to allow control and coordination of all activities associated with the Project Mercury operation.

The Computing and Communications Center, to be located at the Goddard Space Flight Center (GSFC), Greenbelt, Maryland, was to provide for communications control, switching and distribution; also, it was to provide all computations necessary to monitor and control the mission from launch to landing.

Such an arrangement of stations, supported by appropriate instrumentation, would provide for tracking, command, and monitoring capabilities in the highest probable abort phase of launch through insertion and for the critical reentry phase after orbital flight. It also allowed the maximum use of facilities at the National Ranges and of equipment at the Australian Department of Supply facilities at Woomera, Australia. The participating countries and ranges were as follows:

The U.S. Department of Defense provided use of facilities at the Atlantic Missile Range, Pacific Missile Range, White Sands Missile Range, and the Eglin Gulf Coast Test Range.

Australia allowed the use of certain existing facilities and construction, installation, and operations of the required new facilities. These arrangements were made through the Australian Department of Supply and were implemented by the Weapons Research Establishment.

United Kingdom permitted the construction of stations in Canton Island and Bermuda.

Nigeria agreed to the lease of land and permission to construct a station in Tungu and Chawaka.

Spain agreed to provide the land for the Canary Island station.

Development of Equipment Systems

Criteria and Equipment Functions

Basic equipment design and implementation criteria for this program were the result of several major considerations. One of these was economics: existing facilities were to be used wherever they met the Mercury location requirements. Thus, at six locations, a major part of the equipment, including most of the network's tracking radars, was already available. Another major consideration was time. Maximum use of existing, proven equipment was dictated by the necessity to avoid the long-lead times required for research and development. But the primary consideration, overriding all others, was the safety of the astronaut. Some of the design requirements stemming from this consideration follow:

(1) *Reliability* of components and units was required to be designed and engineered into every element of the equipment configuration, and adequate testing was required to prove this reliability.

(2) Despite rigid reliability requirements of units, *redundancy* was to be used extensively throughout each system and always at any critical point. Likewise, *diversity* was to be added to redundancy. Thus, a very reliable system was to be physically duplicated and then to be partially duplicated again by the use of an alternate frequency, location, or some other means of achieving diversity.

(3) Wherever possible, the network system should have the ability to verify its own proper functioning. Suitable monitoring and display devices were thus required.

There were also other requirements resulting from "overlapping" of two or more systems. One of these concerned interference. Determined efforts were made to minimize interference to non-Mercury users of radio frequencies; to reduce mutual interference between Mercury equipment so that there was no degradation of system performance under normal equipment operation; and, to minimize interference from non-Mercury sources by carefully selecting station locations and equipment placement. Interference studies and field measurements were to be undertaken as required. Radiated noise measurements were to be made at all sites.

Particular attention had to be given to system integration problems and to simplifications which might be possible; for example, without compromising reliability, the possibility of reducing the number of antennas at a given site by use of antenna-sharing systems had to be considered.

Finally, all equipment had to be able to withstand the environmental conditions found in such diverse climates as those of the desert at Woomera and the "salt air" of Bermuda.

To provide mission support, the equipment of the network had to provide the following major functions:

(1) Ground radar tracking of the spacecraft and transmission of the radar data to the Goddard computers

(2) Launch, orbital, and reentry computations during the flight with real-time display data being transmitted to Mercury Control Center (MCC)

(3) Real-time telemetry display data at the sites

(4) Command capability at various stations for controlling specific spacecraft functions from the ground

(5) Voice communications between the spacecraft and the ground, and maintenance of a network for voice, teletype, and radar data communications.

Development of the individual systems to meet these requirements is described in the following paragraphs. Some systems have been discussed in earlier publications (refs. 1 and 2); so they are only briefly described here, whereas other systems, especially systems requiring extensive design, are covered in more detail.

Radar

Mission requirements dictated the need for continuous radar tracking during launch and insertion to monitor the launch phase and to establish the initial orbital parameters on which the go—no-go decision would be based. During orbital flight, additional tracking data would be required for a more precise determination of the orbital parameters and time of retrofire for the desired landing point. As nearly continuous tracking as possible was necessary during the less predictable reentry portion of the flight to provide adequate position data on the spacecraft's landing point.

To obtain reliability in providing accurate trajectory data, the Mercury spacecraft was equipped with C-band and S-band cooperative beacons. The ground radar systems had to be compatible with the spacecraft radar beacons.

The FPS–16 radar (fig. 8–2) in use at most national ranges was selected to meet the C-band requirement. Although it originally had a range capability of only 250 nautical miles, most of the FPS–16 radar units selected for the project had been modified for operation up to 500 miles, a NASA requirement, and modification kits were obtained for the remaining systems. In addition to the basic radar system, it was also necessary to provide the required data-handling equipment to allow data to be transmitted from all sites to the computers. Details on data flow and computation are discussed subsequently in the computer section.

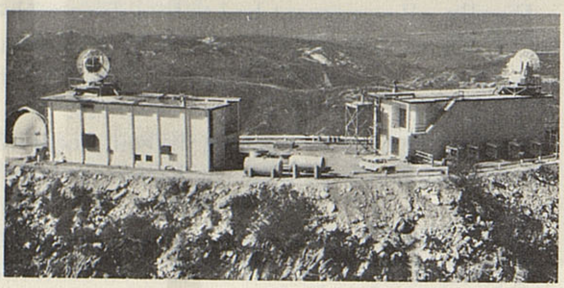


FIGURE 8–2.—FPS–16 radar installation at California.

The FPS–16 system originally planned for the network did not have adequate displays and controls for reliably acquiring the spacecraft in the acquisition time available. Consequently, a contract was negotiated with a manufacturer to provide the instrumentation radar acquisition (IRACQ) modifications. An essential feature of this modification is that it examines all incoming video signals, verifies the target, and automatically establishes angle-only track. Once the spacecraft has been acquired, in angle range, tracking in the automatic mode can be achieved with relative ease. Other features of the IRACQ system included additional angle scan modes and radar phasing controls to permit multiple radar interrogation of the spacecraft beacon. The addition of a beacon local oscillator wave meter permitted the determination of spacecraft-transmitter frequency drift.

Early in the installation program, it was realized that the range of the Bermuda FPS–16 should be increased beyond 500 miles. With the 500-mile-range limitation, it was possible to track the spacecraft for only 30 seconds prior to launch-vehicle sustainer engine cut-off (SECO) during the critical insertion phase. By extending the range capability to 1,000 miles, the spacecraft could be acquired earlier, and additional data could be provided to the Bermuda computer and flight dynamics console. This modification also increased the probability of having valid data available to make a go—no-go decision after SECO.

The Verlort radar (see figs. 8–3(a) and 8–3(b)) fulfilled the S-band requirement with only a few modifications. Significant ones were the addition of specific angle-track capability and additional angular scan modes. At Eglin Air Force Base the MPQ–31 radar was used for S-band tracking by extending its range capability to meet Mercury requirements. The data-handling equipment was essentially the same as for the FPS–16. Coordinate conversion and transmitting equipment was installed at Eglin to allow both the MPQ–31 and the FPS–16 to supply three-coordinate designate data to the AMR radars via Central Analog Data Distributing and Computing (CADDAC).

After implementation these radar systems performed as planned, and only minor modifications were made.

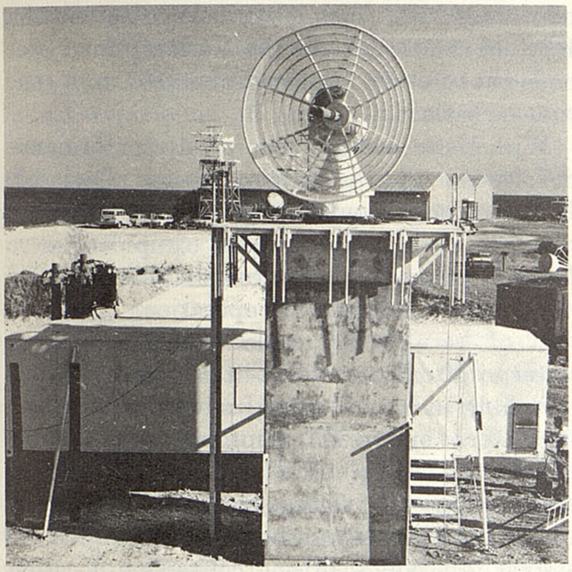
Active Acquisition Aid

Once the types of radars to be used were determined, it became evident that these narrow-beam, precision-tracking units would have difficulty in initially acquiring the small, high-speed spacecraft. Without externally supplied dynamic pointing data, the spacecraft would pass through the radar beam so quickly that the basic radar circuits and/or operators would have very little time in which to recognize the target and switch into automatic tracking.

Two basic types of solution to the radar-acquisition problem were considered. One was the use of an on-site analog computer which would be supplied with predicted spacecraft time and position data by teletype from the Goddard computers. The on-site computer would then generate dynamic-tracking data along the predicted orbit and supply it to the radar during the passage of the spacecraft. This approach was rejected because of the cost and development time necessary to provide suitable analog computers and because it was felt that complete dependence on teletype data for acquisition would not provide sufficient overall tracking system reliability.

The second solution to the problem was a new development called the "active acquisition aid." This device was designed to receive the spacecraft telemetering signals and automatically track the spacecraft in angle with sufficient accuracy to provide suitable pointing data to the radar.

The hardware to meet these requirements was developed around refurbished and modified SCR–584 radar pedestals, antenna, and receiver components. The major units of the final configuration used for Mercury are shown in figure 8–4, and figure 8–5 shows the acquisition aid antenna installation at Guaymas, Mexico.



(a).—Verlort installation at Bermuda.

(b).—Interior view of a Verlort radar van.

FIGURE 8–3.—Photographs of Verlort installations.

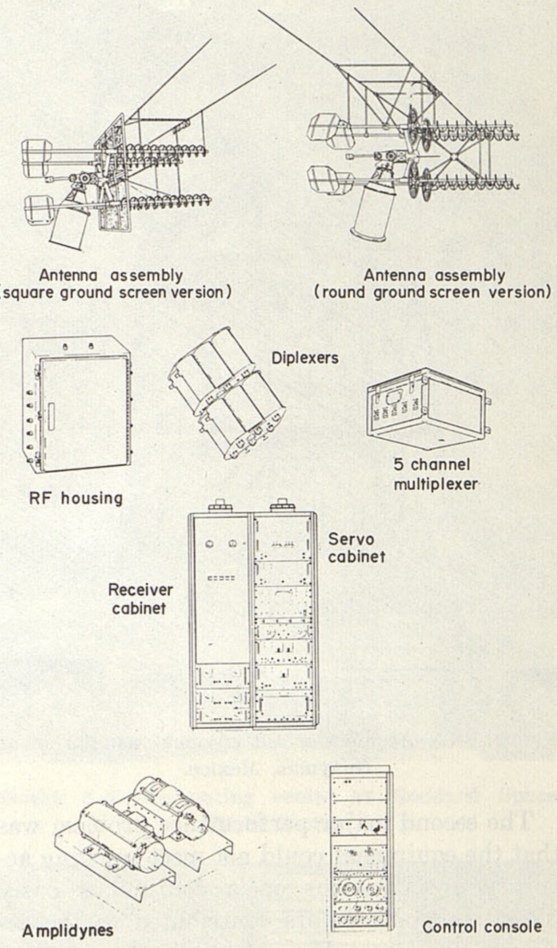


FIGURE 8–4.—Major units of the acquisition system.

Performance analysis.—Tests of the first systems delivered showed two major performance deficiencies. The first of these stemmed from the fact that the spacecraft-telemetering transmitter bandwidth was substantially wider than had been anticipated; the acquisition aid receiver was consequently unable to achieve phase lock. This deficiency was corrected by adapting another existing detector design to the Mercury equipment.

FIGURE 8-5.—Acquisition aid antenna installation at Guaymas, Mexico.

The second major performance problem was that the equipment could not meet tracking accuracy specifications on a continuous basis. Two principal factors contributed to the accuracy problem. The predominant one, especially at low and medium elevation angles, was that of multipath signal reception. The lesser factor was the inherent coarseness of the quadhelix antenna array and other RF components. Redesign of the antenna would have pushed beyond the state of the art and probably would have delayed the program. Use of another, existing antenna with less beamwidth and therefore less multipath susceptibility would, of course, have meant some sacrifice of one of the most desirable advantages of the system: that of being able to cover large areas of space in a short period of time.

Fortunately, early experience with the radars, particularly the FPS-16 which, equipped with the IRACQ modification, can lock on a target very quickly, indicated that the accuracy requirements of the acquisition aid could be relaxed; analysis of tracking requirements showed that with proper alinement, the equipment would provide sufficiently accurate data to the radars. The specified accuracy for the active acquisition aid was thus relaxed to require only tracking within the beamwidth of the particular radar with which it worked ($\pm 0.5°$ for the FPS-16 and $\pm 1.0°$ for the Verlort) for 2 seconds out of every 5 instead of $\pm 0.5°$ on a continuous basis.

With these changes, the initial performance deficiencies of the system were alleviated. However, in the course of the project, a number of other modifications to the equipment were found necessary to improve reliability, ease of maintenance, and ease of operation. Installation of hermetically sealed RF components, waterproof connectors, better antenna limit switching and mechanical limit stops, and bias regulators for the RF amplifiers was made to improve reliability. Test points and grounding switches in the voltage-controlled oscillator (VCO) and a connector board with many of the system test points in one convenient location were installed to improve the ease of maintenance. Changes to the antenna handwheels, relocation of controls, and installation of mode switches were made to increase the ease of operation.

In conclusion, it should be noted that although a number of problems of varying degrees of seriousness were encountered with the acquisition aid—most of them stemming from the necessity of developing a new system in an extremely short time—the equipment successfully fulfilled its intended function. Rarely during the latter Mercury missions did one of them fail to acquire and track the spacecraft shortly after horizon time and thereby aid the radar in acquiring an automatic track.

Computing System

Requirements.—Early in the design of the Mercury system it was considered mandatory to receive information on a real-time basis and to provide for instantaneous computation and display of mission data from lift-off to landing. To meet these requirements, new data transmission equipment and computer peripheral gear were required. A new concept in large-scale, real-time data processing was required to tailor computations to a computer cycle and to manage the priorities of the computations performed automatically.

In all phases of the Mercury mission, it was vital that the many different forms of calculations be performed with exact precision and the data be made available almost instantaneously. For example, in a matter of seconds after launch-vehicle cut-off and spacecraft insertion into orbit, the computers were required to furnish data based on tracking information for evaluating whether or not the mission should be permitted to continue.

Before the Bermuda submarine cable was installed, it was decided to supplement the Goddard-Cape Canaveral complex with a secondary computing station at Bermuda. Installed there was an IBM 709 computer that received the inputs of the Bermuda FPS–16 and Verlort radars. The role of Bermuda was twofold: it served as a backup remote control center during the launch phase and as a tracking site thereafter. Specifically, it performed the following computing tasks:

(1) Provided all the necessary trajectory information to drive the display devices in the Bermuda control center.

(2) Computed an independent go—no-go at insertion based on Bermuda data.

(3) Computed retrofire times to be used in the event of an abort to land the spacecraft in one of the designated recovery areas.

(4) Computed refined landing points for several abort cases.

(5) Computed orbital characteristics.

(6) Sent postinsertion conditions to Goddard.

After the submarine cable was installed in April 1962, the Bermuda computer was removed and all the computations listed above were programed in the Goddard computers.

System description.—Since the computing system was described in a prior publication (ref. 2), only a brief review is presented here.

During a mission, radar data from the network stations are transmitted by way of data circuits (ref. 2) to the communications center (fig. 8–6). Here, real-time equipment places the radar data from each tracking station automatically in the core storage of the computers. Two IBM 7094 computers operating independently, but in parallel, process the data. Should a computer malfunction during the mission, the other computer can be switched on-line to support the mission while the malfunctioning computer is taken off-line and repaired.

The computers provide trajectory information necessary for the flight control of the mission. At MCC, about 18 digital displays, 4 plotboards, and the wall map (fig. 8–7) are driven by the computers. This map shows the present position of the spacecraft and the landing point which would be achieved if the retrorockets were ignited in 30 seconds.

Development of new equipment.—To implement a real-time computing system of the complexity of the one considered for Project

FIGURE 8–6.—Computing center at Goddard Space Flight Center.

FIGURE 8–7.—View of Mercury Control Center showing wall map, plotting boards, and digital displays.