12. GEMINI LAUNCH VEHICLE DEVELOPMENT

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

Summary

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

Introduction

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

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

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

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

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

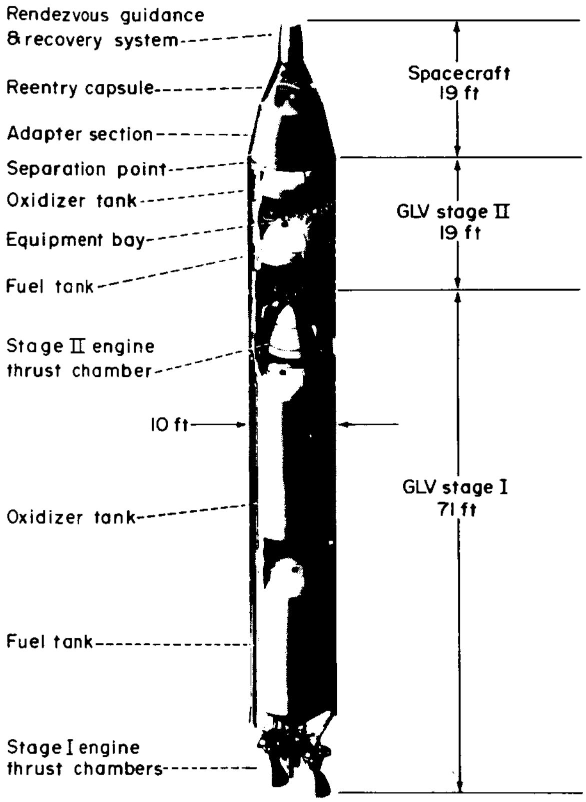

FIGURE 12–1.—Gemini launch vehicle.

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

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

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

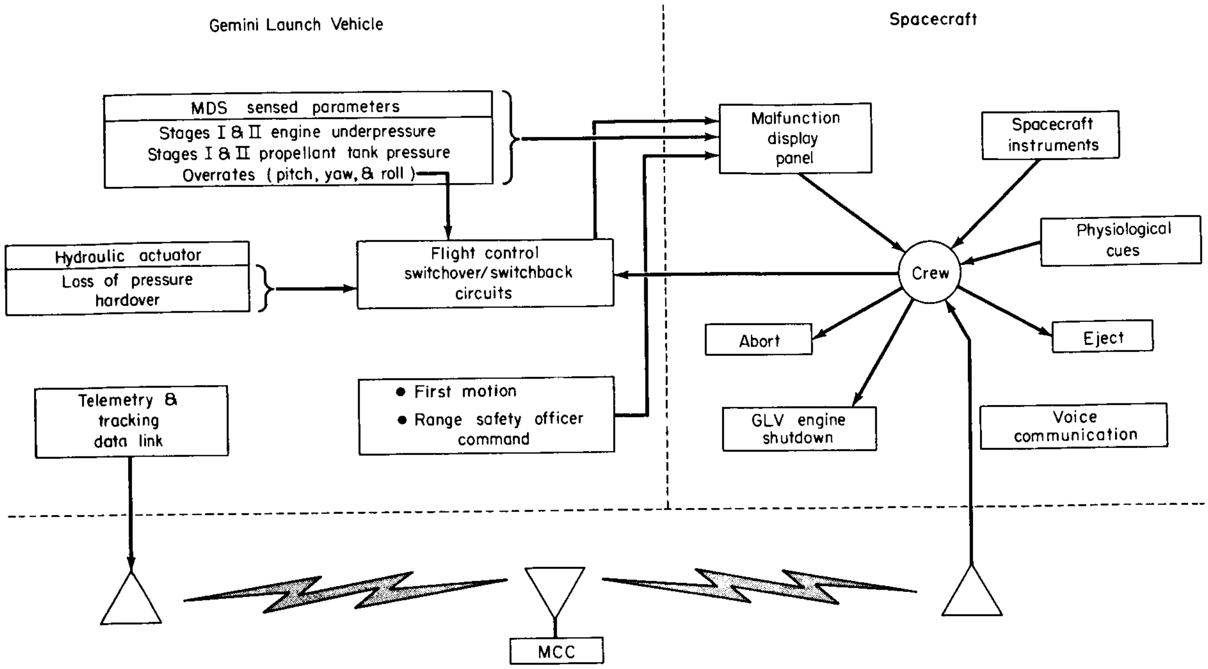

FIGURE 12-2.—Malfunction detection system.

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

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

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

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

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

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

Pilot Safety

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

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

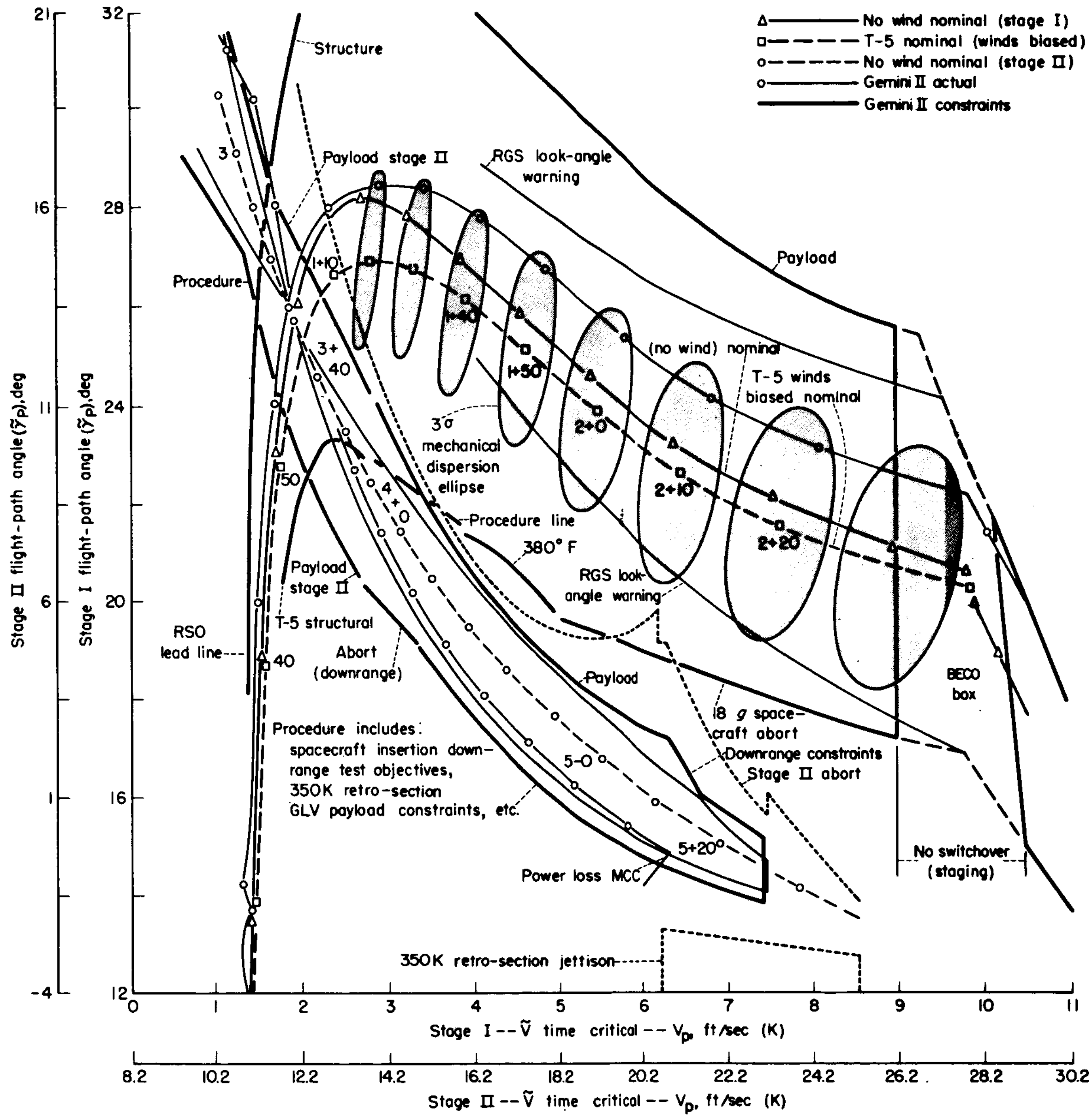

FIGURE 12–3.—Detailed failure-mode analysis.

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

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

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

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

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

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

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

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

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

System Description

Structures

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

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

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

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

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

Propulsion

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

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

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

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

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

Electrical

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

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

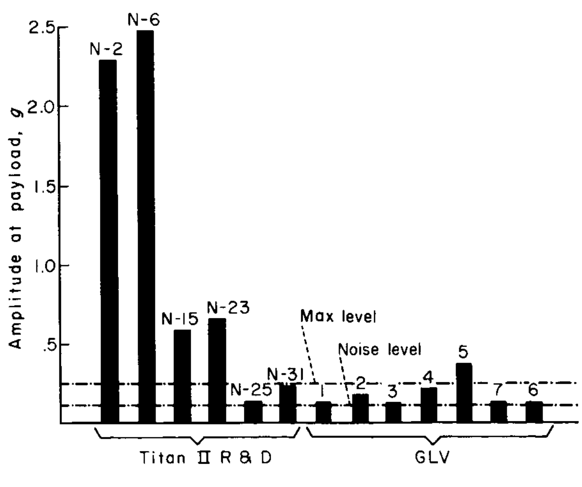

FIGURE 12–4.—History of POGO reduction.

plete set of functions wired through each connector.

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

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

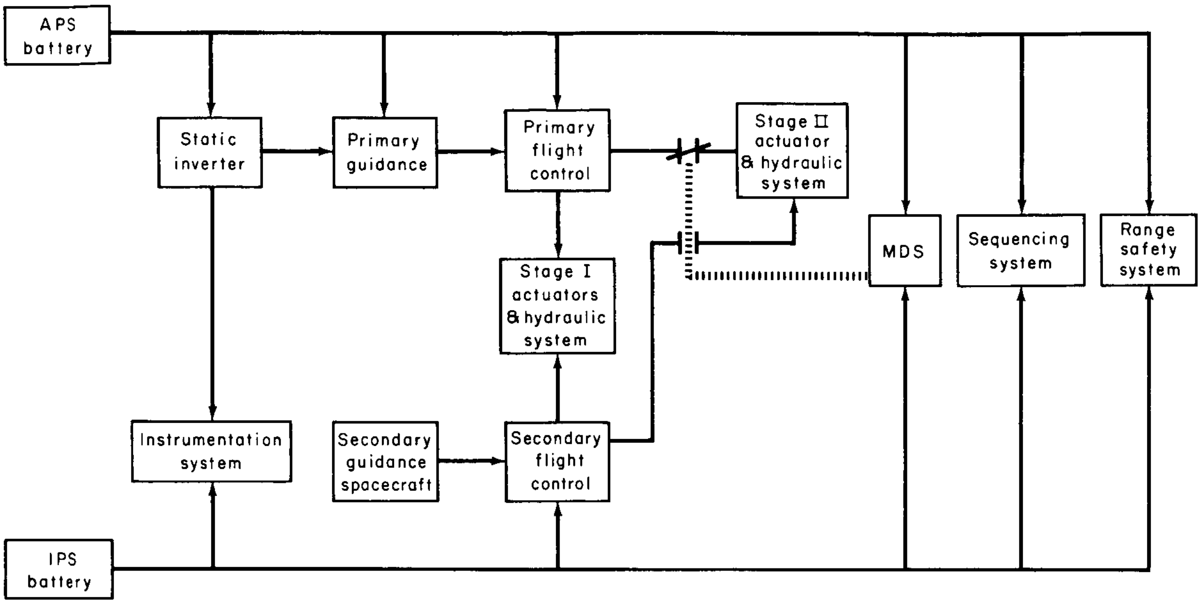

FIGURE 12–5.—Electrical power subsystem.

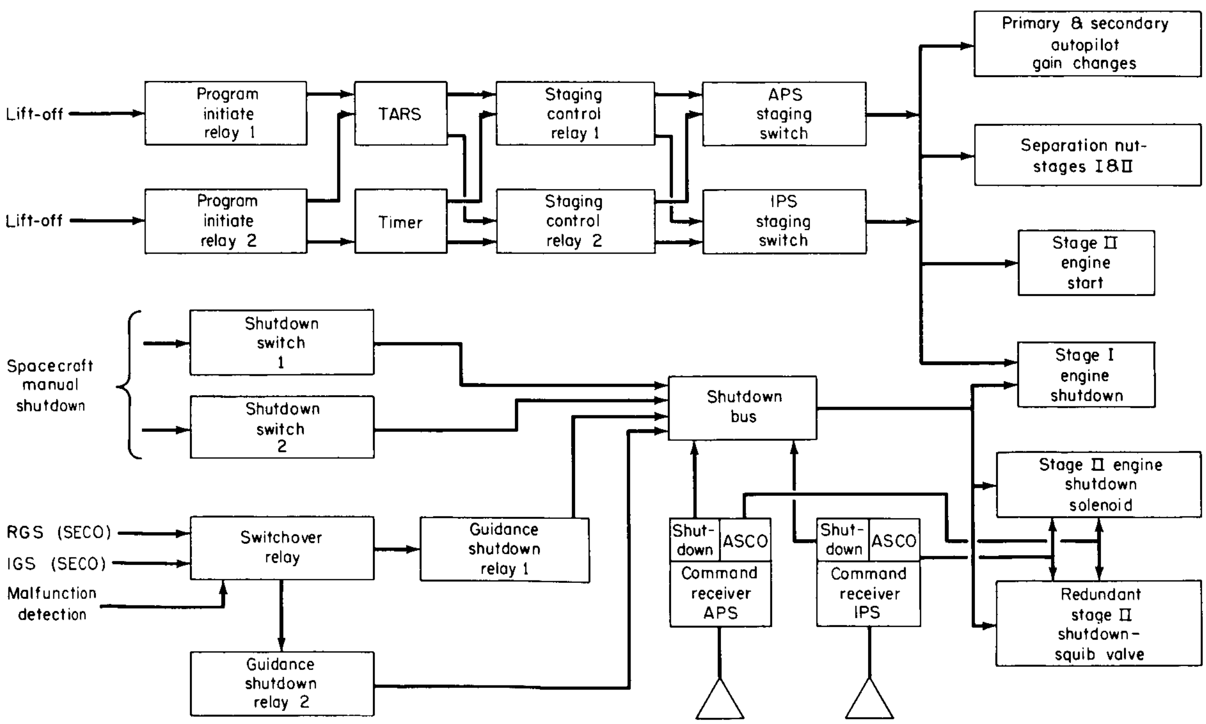

FIGURE 12–6.—Sequencing subsystem.

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

Guidance and Control

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

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

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

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

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

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

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

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

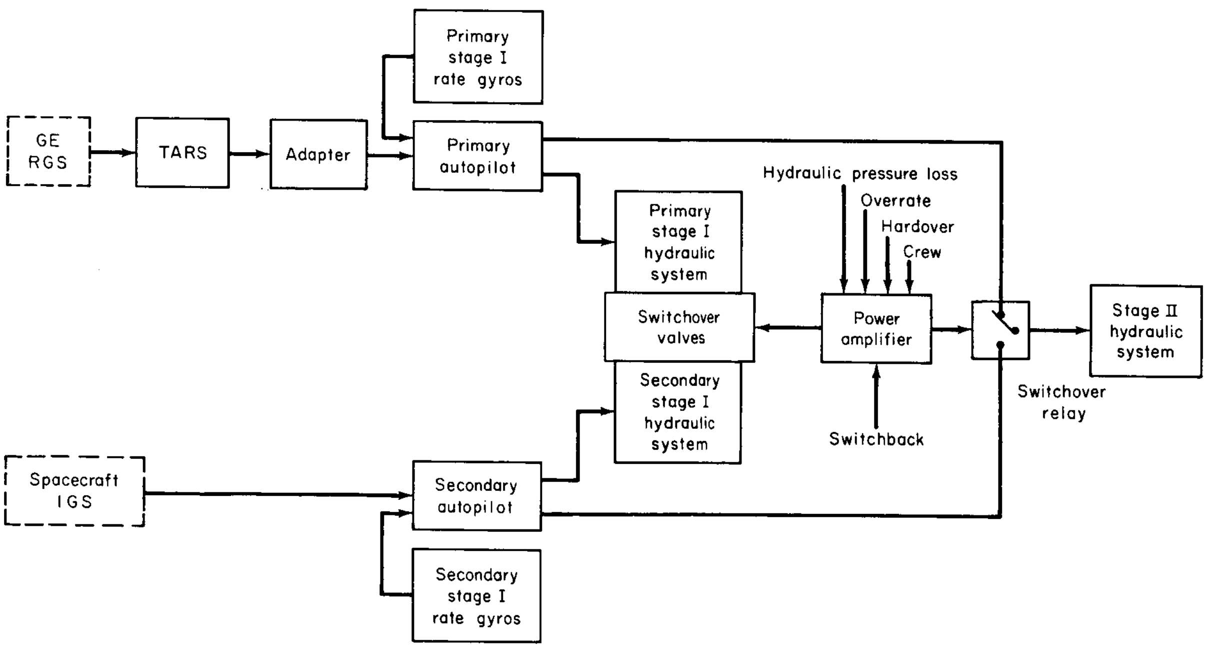

FIGURE 12–7.—Guidance and control subsystems.

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

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

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

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

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

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

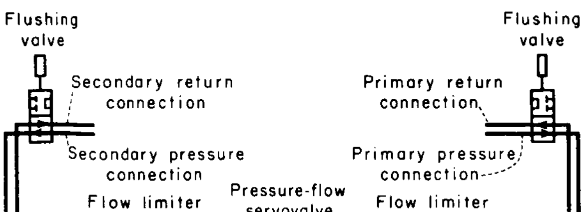

FIGURE 12–8.—Tandem actuator.

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

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

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

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

Malfunction Detection System

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

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

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

(3) Stage II engine fuel-injector underpressure.

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

(5) Secondary guidance and control system switchover.

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

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

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

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

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

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

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

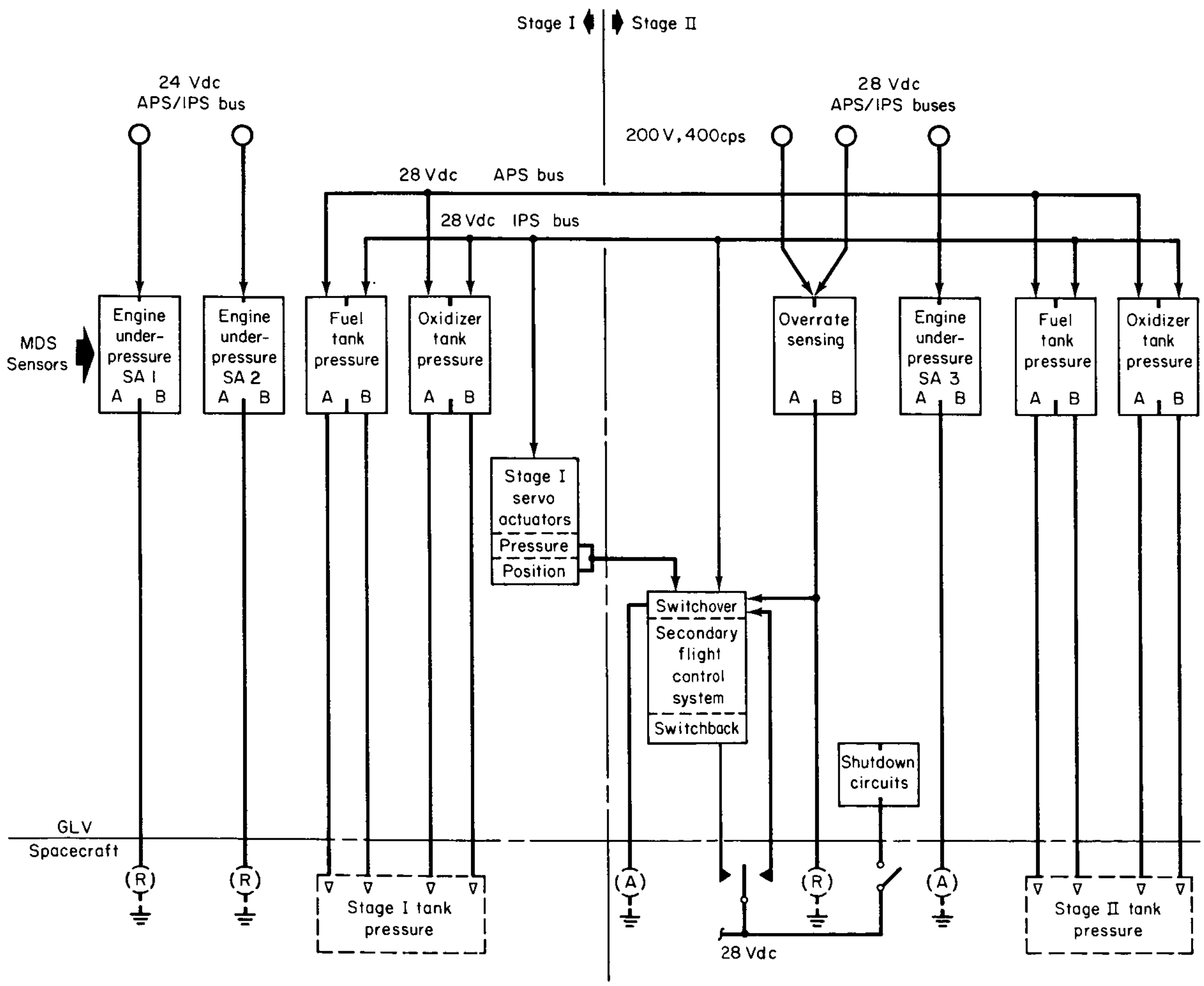

FIGURE 12–9.—Spacecraft monitoring of Gemini launch vehicle malfunction detection.

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

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

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

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

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

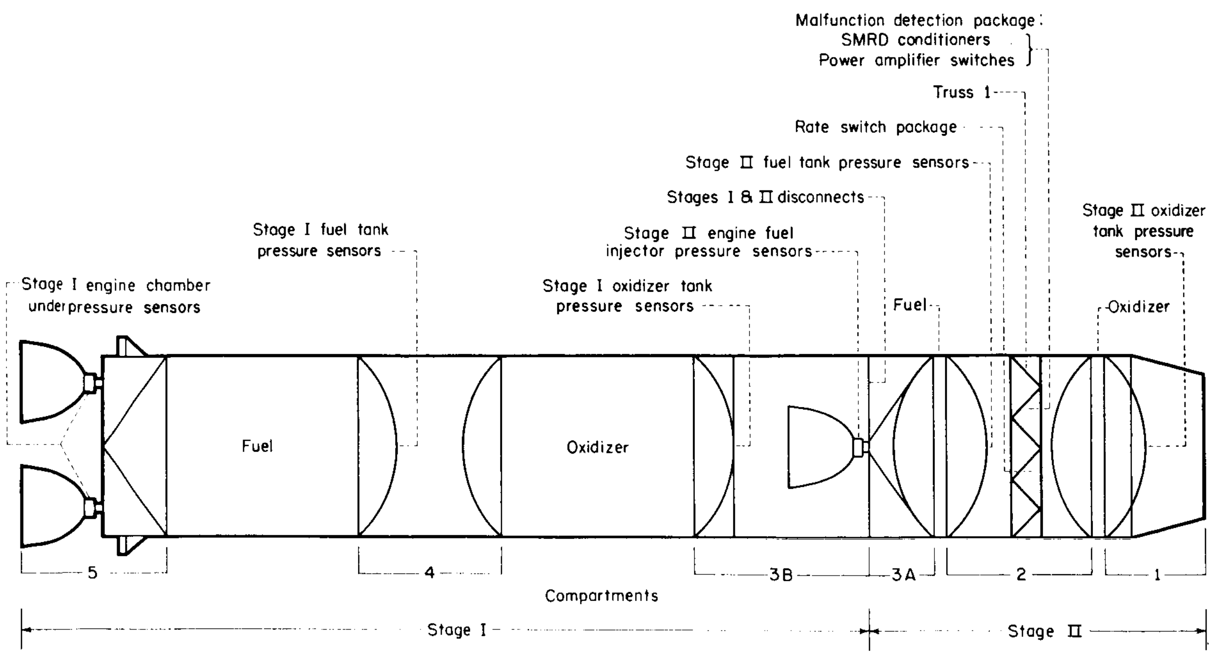

Figure 12–10.—Malfunction detection system components location.

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

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

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

the malfunction detection system has performed as intended.

Test Operations

Airborne Systems Functional Test Stand

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

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

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

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

Vertical Testing at Baltimore

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

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

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

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

Testing at Cape Kennedy

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

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

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

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

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

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

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

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

Test Performance

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

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

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

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

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

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

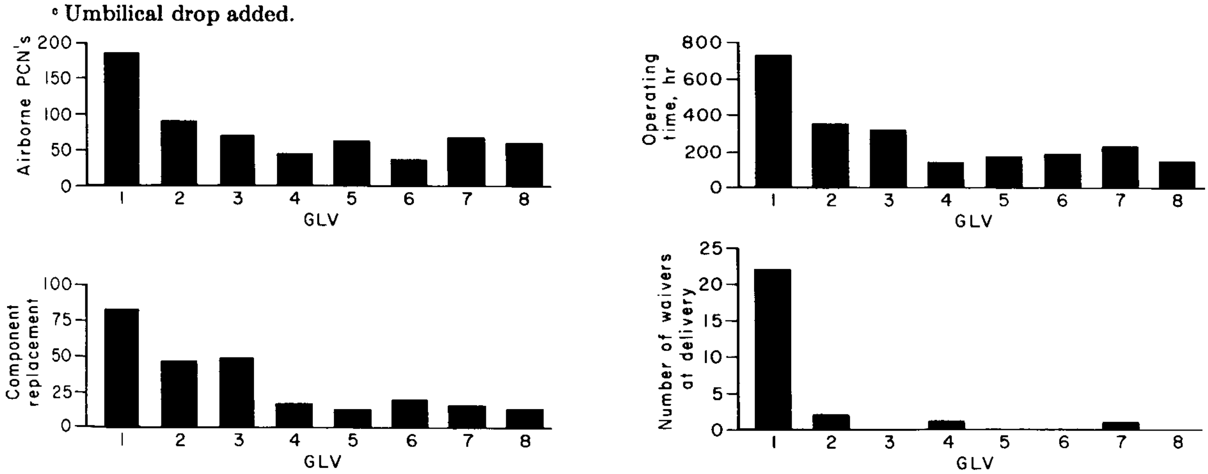

FIGURE 12–11.—Vertical test fixture performance.

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

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

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

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

Data-Trend Monitoring

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

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

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

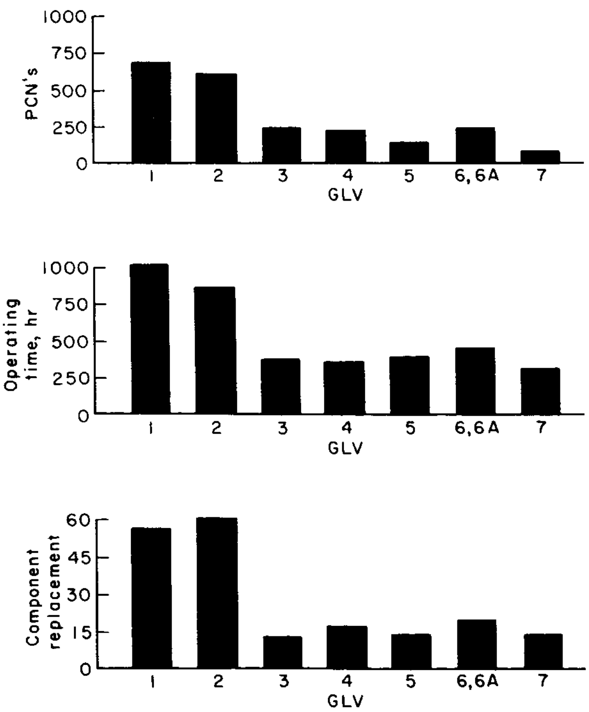

FIGURE 12–12.—Cape Kennedy testing performance.

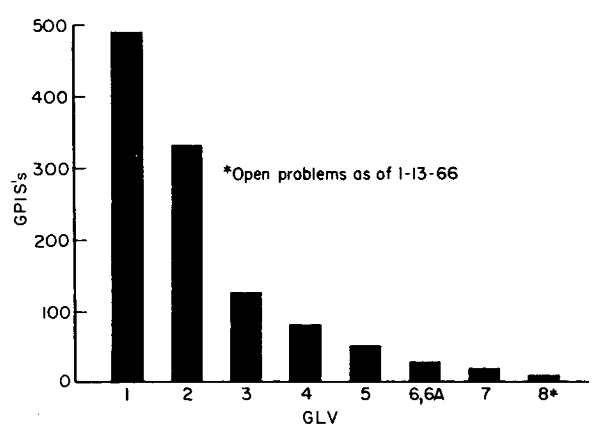

FIGURE 12–13.—Overall measure of test performance.

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

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

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

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

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

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

Personnel Training, Certification, and Motivation

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

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

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

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

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

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

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

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

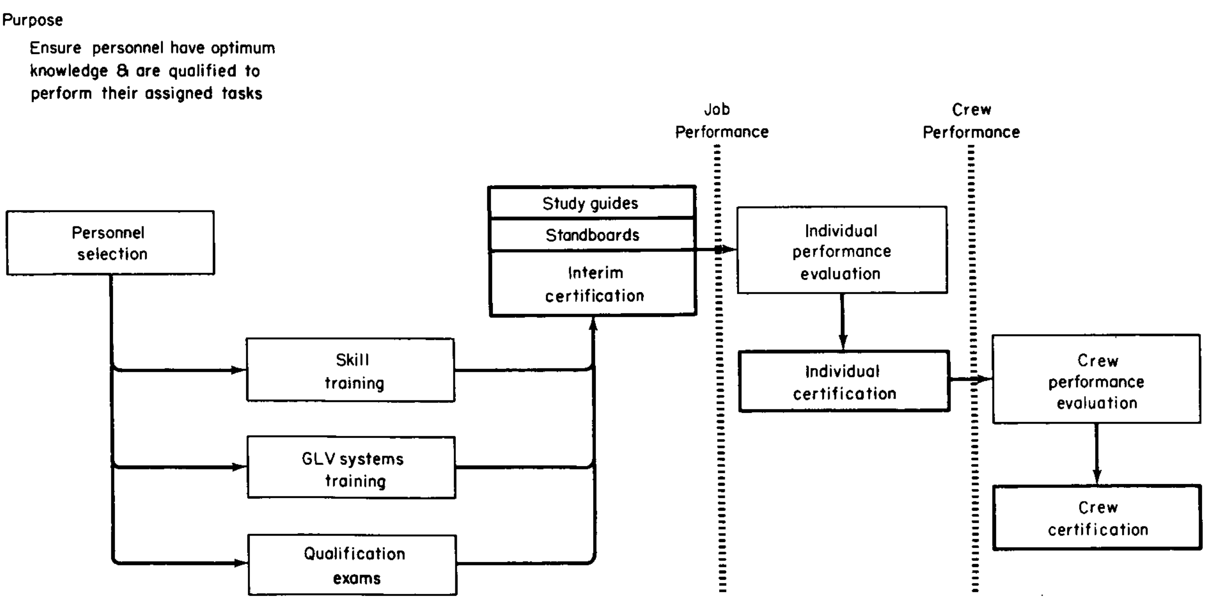

FIGURE 12–14.—Personnel training and certification.

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

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

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

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