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Testing the Man-Rated Launch Vehicle

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Although manned space flight is still in its infancy, testing of launch vehicles has progressed to a high degree of sophistication.

As of December 1965, the Martin-built Gemini launch vehicle has launched seven Gemini spacecraft successfully out of seven attempts. This remarkable record was made possible by two facts:

1. The basic reliability of the hardware
2. The test program.

This paper briefly describes the Gemini launch vehicle noting the major differences between it and the Titan II and discusses the test program. It is not proposed that this is the only method of testing a man-rated launch vehicle; however, it is a successful one.

The Gemini launch vehicle is a basic Air Force Titan II which has been modified in certain areas to achieve man-rating (see Figure 1).

Launch Vehicle Man-Rating

Man-rating is the philosophy and plan for marshaling the disciplines necessary to achieve a satisfactory probability of mission success and crew survival. This probability may be expressed in terms of reliability of the malfunction detection and escape systems. Mathematically, these terms are linked in the equation:

$$P_{CS} = R_{LV} + (1 - R_{LV}) R_{MDS}$$

where:

P_{CS} = probability of crew survival

R_{LV} = reliability of the launch vehicle

R_{MDS} = reliability of the malfunction detection and escape systems.

From this mathematical relationship, it can be seen that achieving a satisfactory level of probability of crew survival required that careful consideration be given to such launch vehicle items as:

1. Component and/or system redundancy which can improve the reliability of the launch vehicle
2. Analysis of launch vehicle failure modes followed by design of a reliable malfunction detection system or MDS
3. Functional utilization of the crew as part of the malfunction detection system
4. Emphasis in launch vehicle checkout on minimizing the possibility of launching a bad vehicle
5. Test, countdown, and launch procedures that will lead to maximum probability of launching a good vehicle
6. System simplification where possible to achieve reliability.

Man-rating a Titan II is a many-sided process conceived by Martin and the Air Force to improve the reliability of the basic vehicle by modifying existing systems, by using redundant components, by adding special systems for crew safety purposes, by special handling of critical components, by meticulous selection of qualified people, and by developing procedures in the entire design-production-manufacturing-test-launch cycle that establishes as a goal flawless performance from the launch vehicle.

Titan II Modifications for Gemini

The high reliability of the ultimate man-rated Gemini-Titan launch vehicle was the result of two decisions made early in the program by Martin and Air Force engineers--

first, to make as few changes as possible in the basic Titan II and secondly, to make every necessary change count toward overall reliability. Each of the modifications added to reliability and safety and contributed to man-rating.

The major modifications, accepted only after careful review by a top level Air Force-Martin engineering board, were:

1. The addition of a malfunction detection system designed to sense problems in any of the vital booster systems and transmit this information to the astronaut crew

2. A redundant flight control system which could take over should the primary system fail in flight

3. Redundancy in the electrical system with necessary changes to provide power for such added launch vehicle equipment as the MDS

4. Substitution of radio guidance for inertial guidance used in the Titan II ICBM version to provide a weight reduction and also to provide a more responsive system during critical orbital injection

5. Redundancy in the hydraulic systems where desirable for pilot safety, such as hydraulic actuators for engine gimbaling

6. Instrumentation to provide additional data during pre-flight checkout and flight not considered necessary in the ICBM version.

Malfunction Detection System

The malfunction detection system (MDS) (Figure 2) is perhaps the most significant modification made to Titan II to prepare it for the manned Gemini mission. The MDS monitors operation of vital launch vehicle subsystems and signals the spacecraft crew if a malfunction takes place. The Gemini-Titan MDS is comparable in function to the launch vehicle ASIS (Abort Sensing and Implementation System) of Project Mercury with one important exception: ASIS was completely automatic (after a serious malfunction was discovered, ASIS automatically initiated mission abort), while MDS only provides vehicle condition information to the astronauts. In other words, MDS signals to the astronaut the proper or improper functioning of the launch vehicle and allows him to make the decision as to whether to continue the mission in the event of a malfunction or when to abort the mission if this should become necessary. Only in the event of an engine hardover condition does the MDS automatically act and in this case, only to switch over to redundant hydraulic, flight control, and electrical systems, not to abort the mission. As a result, the MDS is a relatively simple system with a high degree of reliability. It takes advantage of one of the lessons of Project Mercury - that man can function in space as a working pilot, not just as a passenger in his spacecraft. The MDS, which is completely redundant, monitors during flight: propellant tank pressures, staging of the launch vehicle, thrust chamber pressures, electrical system voltage, and turning rates which would indicate a need for action if the launch vehicle structural limits are approached.

Hydraulic System

The launch vehicle's hydraulic systems control the position of the Stage I and Stage II rocket engine thrust chambers in response to electrical signals from the flight control system. Changing the thrust direction in this manner, the launch vehicle is steered along the desired line-of-flight by making corrections in pitch or yaw axes. The thrust chambers are gimbal-mounted to allow two degrees of freedom. Stage I roll control is provided also by varying direction of the two thrust chambers. A separate roll control nozzle with one degree of freedom is provided in Stage II which has one thrust chamber.

The Gemini-Titan launch vehicle has three separate hydraulic systems - the Stage I primary system, the Stage I secondary system, and the Stage II system. A schematic of the redundant Stage I hydraulic system is shown in Figure 3.

Stage I Primary. Basic components of the Stage I primary hydraulic system are an engine-driven pump, an accumulator/reservoir, four servo actuators, and an electric motor-driven pump. An axial piston, pressure-compensated, variable volume engine-driven pump pressurizes the system at 3000 psi during Stage I flight. The pump is driven by the accessory drive pad on the turbodrive assembly of the Stage I engine. The servo actuators control the movement of the Stage I thrust chambers. The actuators are built as tandem units consisting of two complete electro-hydraulic servo system sections. One servo section of each actuator is connected to the primary hydraulic system; the other servo actuator section in each actuator comprising a portion of a secondary or redundant hydraulic system. The actuator sections are independent but interconnected by a special "switchover" valve that permits only one of the servo sections to be operable at any given time.

Stage I Secondary. The secondary or redundant Stage I hydraulic system is identical to the primary system. It operates in line with the secondary or redundant flight control system. The same electric motor-driven pump is used for both secondary and primary systems during checkout through use of a system test selector valve. The secondary system is pressurized throughout flight, although not in use unless switchover occurs. The actuator switchover valves are designed to sense primary system pressure and initiate switchover to the secondary hydraulic system in case of any failure within the primary system that would result in a loss or degradation of pressure below a predetermined value.

Stage II. The Stage II hydraulic system is not redundant since studies of potential malfunctions did not indicate a necessity for redundancy such as is the case in the event of an engine hardover condition at Max Q in Stage I. The components of the Stage II system are basically the same as for the Stage I systems except that three actuators are used - two on the single thrust chamber for pitch and yaw control, and one on the off-center turbine exhaust nozzle for roll control.

Flight Controls

The Gemini-Titan flight control system constitutes a basic change from the Titan ICBM system. Specifically, the system is designed to withstand any single malfunction (failure) and complete the launch phase successfully. The design achieves this increased reliability by complete flight control system redundancy (Figure 4).

A complete secondary or redundant flight control system is provided in the Gemini-Titan launch vehicle to protect the astronaut crew from this one event. Switchover from the primary to secondary system in the event of an engine hardover condition is automatic and requires only 15 milliseconds. The switchover is final and no provision is made for switching back during Stage I flight. After staging, the pilot may switch back to the primary mode. Switchover to the secondary system also can be effected by three other methods:

1. Astronaut command
2. Vehicle overrate detected by the MDS rate sensors
3. Loss of Stage I primary hydraulic system pressure.

The primary flight control system consists of the following:

1. Three Axis Reference System (TARS). Flight-proven and adapted from the Titan I ICBM program, the TARS is located in the equipment bay between the tanks of Stage II. It provides information concerning angular displacement along the three perpendicular axes of roll, pitch, and yaw by use of gyroscopes. Included in the TARS unit is a programmer which, as a function of time during Stage I flight, changes the angular reference of the launch vehicle in the pitch and roll axes, thereby

initiating changes in the direction of flight along these two axes according to a preplanned flight trajectory. In other words, it serves a guidance function in Stage I flight signaling commands to such other units as hydraulics by way of the adapter package and autopilot. The roll programmer can be continually updated during the prelaunch countdown to set a new azimuth, such as might be necessary in a rendezvous mission to achieve the correct orbital plane. The pitch program is not changed during countdown since it is fixed ahead of time for each individual mission. During Stage II flight, the TARS accepts steering signals or angular reference changes in pitch and yaw from the radio guidance system and executes these commands by again signaling other components of the flight control system. The TARS also actuates auxiliary switches in accordance with a preset time program to provide such functions as arming of the sensors which signal stage separation at the proper point in flight.

2. Adapter Package. The adapter package, also located between the Stage II tanks, conditions the attitude outputs of the TARS for input in the autopilot. At the proper time as programmed in the TARS, the adapter allows the pitch and yaw steering signals from the radio guidance system to be applied to the TARS guidance amplifiers. The adapter package also houses the switchover relays (redundant) which effect the primary-secondary flight control system shift. During checkout of the booster, performance of all fifteen flight control system gyros are monitored through signals amplified by the adapter package.

3. Stage I Rate Gyro. Located in the inter-stage section, the Stage I rate gyro is a Titan II flight control unit. It contains three gyros to measure pitch, roll, and yaw components of the launch vehicle's angular rate during Stage I flight. The rate gyro output signals are supplied to the autopilot.

4. Autopilot. The autopilot is located between the Stage II tanks and is modified from the Titan ICBM autopilot only where necessary to meet specific Gemini mission requirements. Included in the autopilot are: three axes rate gyros for Stage II flight; an 800 cycle static inverter providing magnetic amplifier and rate gyro power supply for both Stage I and Stage II rate gyros; and circuitry necessary to accept signals from Stage I and Stage II rate gyros and to amplify, distribute, and condition attitude reference signals from the adapter package to the hydraulic servo actuators.

5. Mod III G Radio Guidance. The radio guidance system, discussed in detail in another section, provides pitch and yaw guidance signals to the TARS through the adapter package during Stage II flight.

6. The secondary or redundant flight control system consists of the following components: (1) Stage I rate gyro: a duplicate of the primary system unit (2) Autopilot: primary system duplicate.

7. Spacecraft Inertial Guidance System. The spacecraft IGS provides those attitude stabilization signals to the secondary system autopilot which would have been provided during Stage I flight in the primary system by the TARS and during Stage II flight by the radio guidance system. The inertial guidance system is pre-programmed to perform this function in the event of switchover to the secondary system.

Gemini Test Program

The test program on the Gemini launch vehicle is a repetitive series of detailed, quantitative tests starting at the Vertical Test Fixture in the Martin Company's manufacturing plant at Baltimore, Maryland and culminating in the launch from Complex 19 at Cape Kennedy, Florida.

In developing this program, great care was taken to have the tests and procedures at both facilities as nearly identical as possible. The aerospace ground equipment, with the exception of the propellant loading system, at both facilities are identical.

The data gathered in the Vertical Test Fixture is plotted and analyzed by the system designer to verify initial performance. Each time a test is repeated, the data is compared with the original. All this test data accompanies the vehicle to Martin's Canaveral Division at Cape Kennedy where the process is continued.

Factory Test and Vehicle Acceptance

The sequence of testing and acceptance at the Vertical Test Fixture is shown in Figure 5. A brief description of these activities follows.

Launch Vehicle Erection and Alignment. The vehicle is erected and checked for vertical alignment structural twist and levelness of the gyro mounting pads.

Post-Erection Inspection. Approximately one week is allotted for a complete physical inspection of the vehicle structure, cabling, and black box installations prior to initiating the subsystem test phase.

Subsystem Functional Verification. Verification of the launch vehicle subsystems commences with voltage standing wave ratio and attenuation tests of the waveguide and radio frequency cabling and resistance checks of the auxiliary and instrumentation busses. Power is then applied to the individual busses and systems sequentially.

The launch vehicle subsystems, i. e., electrical, flight controls, tracking and radio frequency, propulsion, malfunction detection system, and instrumentation are now subjected to the first of the series of meticulous tests which finally culminate in the launch from Cape Kennedy. All data is reviewed by the responsible design engineers for specification compliance and entered on the trend charts (see Figure 6).

When the individual subsystems have been verified, each of the 235 airborne telemetry measurements is calibrated - usually four to six points - and data reduction curves are drawn.

The next step in the subsystem test sequence is the telemetry ambient test. Here, the individual subsystems are exercised in a tightly-controlled sequence while the airborne telemetry system is on and radiating and recordings of this data are made. The data is reduced and entered in the launch vehicle Data Book for future reference and use as a data reduction tool.

Systems Test. The first major system test which the launch vehicle undergoes is the Acceptance Mode Verification Test. This test is quite similar to the Combined System Acceptance Test and the combined system tests at Cape Kennedy in that a short minus time countdown is conducted, followed by a simulated flight. This data is again reviewed by the design groups and used as a baseline for the launch vehicle.

Combined System Acceptance Test. The Combined System Acceptance Test specifies in detail the steps necessary to bring the launch vehicle and aerospace ground equipment from a static power-off condition to a countdown configuration. In addition, it gives direction for performing an abbreviated count and two simulated flights (for details, see section on Cape Kennedy Tests).

Launch Vehicle Acceptance. The data from the Combined System Acceptance Test and all manufacturing and test data are prepared and presented to the Vehicle Acceptance Team by Martin management and engineering personnel for their review. The Vehicle Acceptance Team is chaired by a senior officer of the Air Force Space Systems Division Gemini Program Office in Los Angeles and is composed of management and technical specialists from the Air Force, NASA, and Aerospace Corporation. When the Vehicle Acceptance Team determines, through its own analysis of the launch vehicle history and test data, that the launch vehicle meets the stringent requirements of the Gemini program, it is accepted by the Air Force and prepared for shipment to Cape Kennedy.

Cape Kennedy Tests

The pre-launch testing at Cape Kennedy and the Air Force Eastern Test Range can be divided into two phases:

1. Launch vehicle
2. Integrated launch vehicle and spacecraft.

Launch Vehicle

Subsystem. The first portion of the testing at the Cape is quite similar to the testing at the Vertical Test Fixture in Baltimore (see Figure 7). However, in place of the subsystem functionals, an abbreviated subsystem retest has been substituted. This abridged testing in no way detracts from our confidence in launch vehicle performance. Rather, it is made possible by our test philosophy which dictated that all test data remained with the vehicle until launch. It is therefore possible for the data recorded at the Cape to be compared with the acceptance data and analyzed for trends indicative of an incipient failure. This analysis is performed concurrently by the Martin Company Project Engineering Section at the Canaveral Division and by the various design groups at the Baltimore Division. It should be noted that this data comparison and trend analysis do not cease when the individual subsystems have been reverified. The subsystem retest data is added to the vehicle history and compared with the data gathered in all the combined tests which follow (Figure 8).

System. The launch vehicle is declared ready to accept the spacecraft upon the successful completion of the Pre-mate Combined System Test. The Premate Combined System Test consists of a countdown from T-45 minutes through T-0; a simulated liftoff; switchover to the secondary guidance and flight control system initiated by the malfunction detection system; and a simulated flight terminating at sustainer engine cutoff. An electronic spacecraft simulator is used to provide proper electrical loads and simulated guidance signals to the launch vehicle. All systems are then recycled to T-3 minutes and another countdown ensues. During the plus time run of this second simulated flight, the launch vehicle remains in the primary guidance and flight control mode. Verification of correct response to the pitch and roll program from the Three Axis Reference System as well as steering commands from the radio guidance system and system response to discrete commands, i. e., gain change, staging arm, booster engine cutoff, and sustainer engine cutoff is made. The launch vehicle systems are again recycled to a T-3 minute prelaunch configuration for a third and final simulated flight. During the plus time run for this test, all launch vehicle umbilicals are removed in their normal sequence. At the normal staging time, booster engine cutoff, the electrical connectors at the staging interface plane are disconnected. The launch vehicle is evaluated not only for response to proper commands but also to ensure that there is no improper subsystem interaction.

Combined Launch Vehicle and Spacecraft

Most of the combined launch vehicle and spacecraft testing is done on a system basis. However, since the inertial guidance system is used as the secondary launch vehicle guidance system, subsystem testing, i. e., detailed quantitative verification of system gain and response is performed. Likewise, since the primary purpose of the malfunction detection system is to inform the pilots of vehicle performance, this subsystem also undergoes more detail verification of the interface.

Electrical Interface Integrated Validation. All interface wiring across the launch vehicle/spacecraft interface is redundant. To ensure that this is true, each circuit is subjected to a power-off resistance test with first one and then the other electrical interface connector connected. When verification of the proper resistance on all circuits is complete, both connectors are mated and the complete launch vehicle/spacecraft is ready for functional test. Some of the functional tests which the space system is subjected to and a brief description of each follows.

Liftoff and Test Conductor Abort. Verifies that when the launch vehicle lifts off the spacecraft receives the proper indication. Verifies that the test conductor abort command is received by the spacecraft by both radio frequency and landline methods.

Ascent Simulation. The Ascent Simulation is quite similar to the first run of the Pre-mate Combined System Test. However, the spacecraft inertial guidance system now furnishes the pitch and roll programs, certain steering commands, and discrete commands to the launch vehicle. Switchover to the secondary guidance and control system is initiated by the pilot.

Fade-in Demonstration. Verifies that when the space system switches from primary to secondary guidance and control the inertial guidance system computer recognizes the amount of attitude error and corrects the flight path over a period of time rather than applying a step function.

Fuel and Oxidizer Tank Pressure Meter Calibration. A three-point calibration and comparison of the malfunction detection system propellant tank pressure transducers and spacecraft analog meters.

Primary to Secondary Switchover Test. Verifies that each of the hydraulic inputs to the malfunction detection system will initiate switchover independently.

Staging Interface. Verifies the spacecraft receives proper indication of launch vehicle staging.

Joint Guidance and Controls Test

The Joint Guidance and Controls Test assures that the launch vehicle/spacecraft secondary guidance and control system performs properly by making detailed, quantitative measurements of the attitude and rate gains. Proper system phasing is determined by placing the inertial platform in the inertial mode and utilizing the earth's rotation to generate an output to the flight control system.

Joint Combined Systems Test

The Joint Combined Systems Test is the first major exercise of the complete space system and is designed to verify compatibility of the launch vehicle and spacecraft in a simulated flight configuration. This test consists of:

1. An abbreviated countdown and liftoff followed by a simulated abort initiated by the pilot

2. The launch vehicle and spacecraft are then recycled to T-45 minutes for a countdown and simulated flight utilizing the primary guidance and flight control system.

Simulated Flight Test

The Simulated Flight Test is the final exercise of the complete launch vehicle/spacecraft system prior to initiating the launch countdown. This test consists of three simulated countdowns and flights which are identical to tests previously described, i. e., abort, secondary guidance and control, and primary guidance and control. Immediately following the Simulated Flight Test, the launch vehicle is declared ready for flight and special quality and security procedures are put in effect to ensure system integrity is maintained.

Launch Countdown

Since the launch vehicle and spacecraft are both relatively complex systems and independent during normal flight, a split countdown was developed for Gemini (see Figure 9). The split count allows for maximum flexibility and minimum interdependence between the two systems during the countdown.

The launch count commences on F-1 day

approximately nineteen hours prior to launch. In a four-hour period, all of the launch vehicle/spacecraft interface is reverified. The count is then held until T-360 minutes on launch day. It is during this hold that the launch vehicle is loaded with propellants and final topping of some of the spacecraft cryogenics is accomplished.

The Range count is initiated at T-240 minutes with the spacecraft and launch vehicle counting independently. It should be noted that one of the major precepts of the Gemini test philosophy dictated that the launch vehicle be completely verified as ready for flight prior to crew ingress (T-95 minutes). Therefore, there are no tests performed after ingress that have not been performed earlier in the count.

After crew ingress, the countdown - with the exception of the mechanical functions necessary to lower the erector and prepare the stand for launch - is an exact replica of the count performed for the Pre-mate Combined Systems Test, Joint Combined Systems Test, and Simulated Flight Test.

For a closer insight into the problems associated with planning and conducting a countdown, see "Rendezvous Launch Operations Planning."[1]

Pilot Safety Program

A launch vehicle pilot safety program has been established by the Air Force Space Systems Division to ensure that a concern for safety is manifested in plans, reflected in appropriate activity, adequately documented, and thoroughly assessed prior to launch. The program is implemented in two ways. First, the program ensures a continuous monitoring effort commencing with the preliminary design and continuing through launch. Second, the program concentrates considerable effort at key focal points when major problems arise. Assurance that nothing has been neglected is provided by following a pattern of rigorous technical monitoring of associate contractors' activity; rigid control of all phases of design, development, engineering changes, production, inspection, testing, handling, acceptance and launch; emphasis on configuration documentation and verification control; and extensive data and procedural reviews.

As part of the pilot safety program, the Air Force imposes stringent requirements during the acceptance phase. Hardware is not accepted until the Air Force is convinced that the hardware and documentation comply with appropriate specifications and other contractual requirements and meet the requirements for the Gemini mission. Acceptance is characterized by a methodical approach and an uncompromising attitude.

In discussing the pilot safety program, this paper is confined to its influence and impact on the pre-launch test program.

Test Procedure Control

The Gemini test procedures used at Cape Kennedy are written by the Canaveral Division of the Martin Company. These basic procedures are prepared by the engineers assigned to the Launch Operations Section and are governed by contractual and engineering specifications. The Gemini test procedures are written in considerable detail, leaving nothing to the discretion of the operator. The specification revision designation is noted on the draft.

The procedure is then forwarded to Configuration Management where it is verified that the 'written to' level of the referenced specifications is the latest released engineering. This is performed using a computer tab run which is updated daily.

The procedure next is routed to the Gemini Project Engineering and Quality Engineering sections for reverification of technical adequacy, specification compliance, and adherence to Martin quality standards.

The Pilot Safety engineer reviews the procedure for compliance with the total pilot safety program.

The procedure is then forwarded to the

customer, i. e., the 6555th Aerospace Test Wing, for review and coordination.

After all approvals are secured and all agencies satisfied, the procedure is published and released for use (Figure 10).

Now, the typical procedure starts through the test cycle. The Official Test Copy (OTC) is issued to the responsible system engineer in accordance with the test schedule. Attached to the Official Test Copy are several Procedure History Sheets for documenting all pertinent facts, deviations, and anomalies concerning the test. The complete package is bound and sealed by the Quality Section to assure that all pages remain in the Official Test Copy.

After the completion of the procedure, a post-test critique meeting is held during which all recorded data and Procedure History Sheet items are reviewed and accepted or rejected. Present at this meeting and responsible for verifying that the data is satisfactory and the system is performing properly, are the Operations engineer and the appropriate system engineers from the Project Engineering and Quality Engineering sections.

The test procedure and all associated data, i. e., strip charts or other recordings, are then forwarded to the Quality data storage area for review by the Pilot Safety Working Team.

The Pilot Safety Working Team consists of representatives of Martin Company Engineering Section; Customer Quality; Aerospace Corporation; and the 6555th Aerospace Test Wing. They conduct a separate review to assure completeness of the documentation and procedure.

It is readily recognizable that these additional reviews and reverification of the documentation add considerable cost to a program over and above what would normally be expected in a weapon system development program. From a launch operations standpoint, it can be estimated this cost as approximately thirty per cent over a normal flight development effort.

Crew Training and Motivation

Another aspect of the pilot safety program - one which cannot be over-emphasized - is the motivation and skill of the launch crew. Each and every technician and engineer working on the program must be aware of his responsibility for pilot safety. For in the final analysis, all of the management effort, data reviews, trend analysis, and basic vehicle reliability can be negated by an undetected human error.

The average man assigned to the Gemini launch team has over seven years of Martin launch experience. Most of this experience was on the earlier Titan I and Titan II vehicles. Nevertheless, upon transfer to the Gemini program, they were again subjected to rigorous classroom training, written and oral exams, and performance demonstrations before being certified as team members. This training has continued throughout the program.

Methods of motivating people to assure maximum quality vary from the classical poster "Uncle Sam Wants You" to pep talks and lectures. The ones which seem to have worked on Gemini are the sense of identification with the program and the pride in a job well done. These have been greatly enhanced by the almost daily contact with the flight crews, the establishment of many space records for the United States, and the many words and letters of commendation from the Air Force, NASA, and the'guys who drive them'.

Launch Confidence versus Program Needs

The confidence level necessary to commit to a manned launch is achieved incrementally during the test program for each launch. Affecting this 'confidence' are the results of the previous flights, the Vertical Test Fixture tests, and Eastern Test Range tests, the data and trend analysis, and the performance of the launch crew.

Assuming that this is true, the question naturally arises as to how the Gemini 7 and 6 mission was possible. The answer is simple. The Gemini 7 and 6 mission was possible because of the Gemini test philosophy.

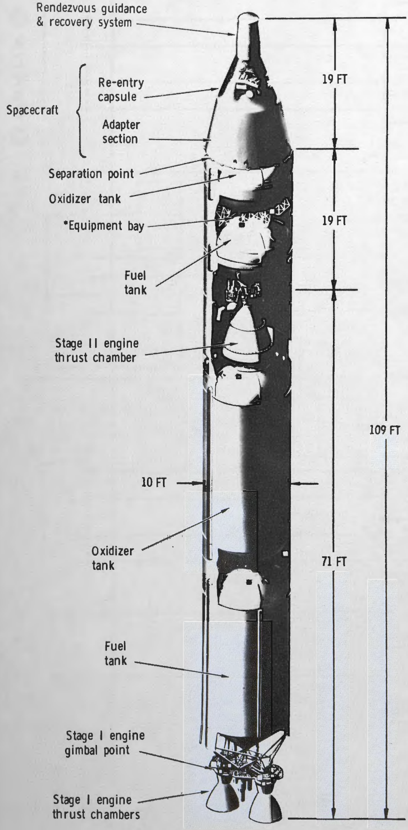
GT-6 was completely checked out according to the normal Gemini schedule, including a launch countdown which terminated prematurely due to a target vehicle problem. GT-6 had performed perfectly during all phases of the ground test and analysis of the test data showed no trends indicative of a potential problem. Extreme care was exercised in removing it from the launch complex. Only those electrical connectors at the staging and spacecraft interface planes were disconnected, making the retest minimal.

After re-erection, the individual subsystems were reverified, using somewhat abbreviated procedures prior to the Simulated Flight Test. From the Simulated Flight Test through to launch, normal procedures were followed. The data from the Simulated Flight Test was compared by use of a computer with the Simulated Flight Test performed prior to the first launch attempt and found to be an overlay. The launch vehicle was again performing perfectly according to the engineering criteria and the emotional criteria of confidence was satisfied.

Although the Gemini test program is controlled by very rigid application of the engineering and quality disciplines as outlined in the procedural control, data and trend analysis, and pilot safety program, it has maintained the flexibility to respond to changing program needs.

References:

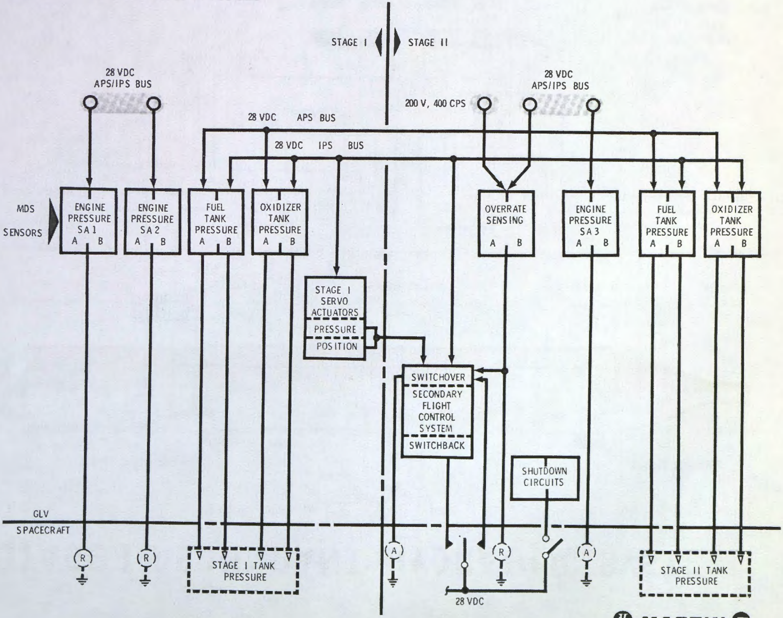
1. Rendezvous Launch Operations Planning by Mark Goodkind, Proceedings of Third Space Congress



*Equipment bay contains:

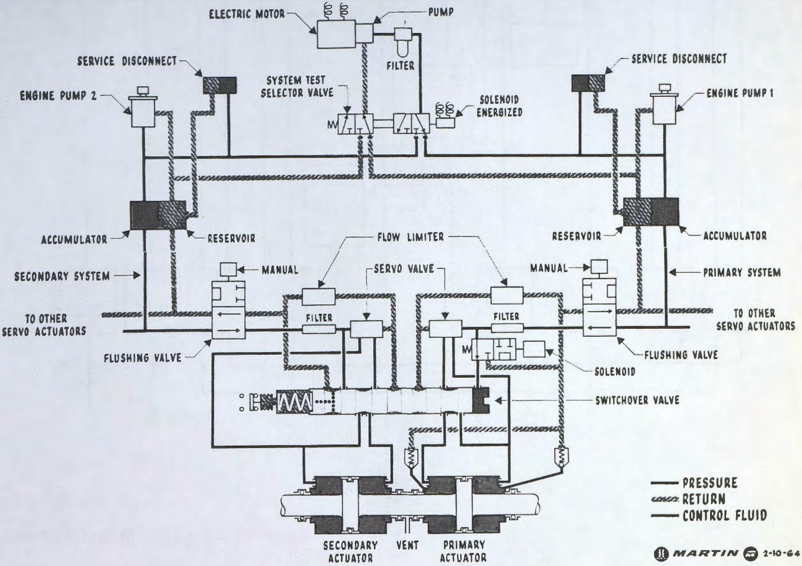
- Batteries
- Malfunction detection system (MDS) units
- Range safety command control system
- Programmer
- Three-axis reference system (TARS)
- Radio guidance system (RGS)
- Autopilot
- Instrumentation and telemetry system

MALFUNCTION DETECTION SYSTEM

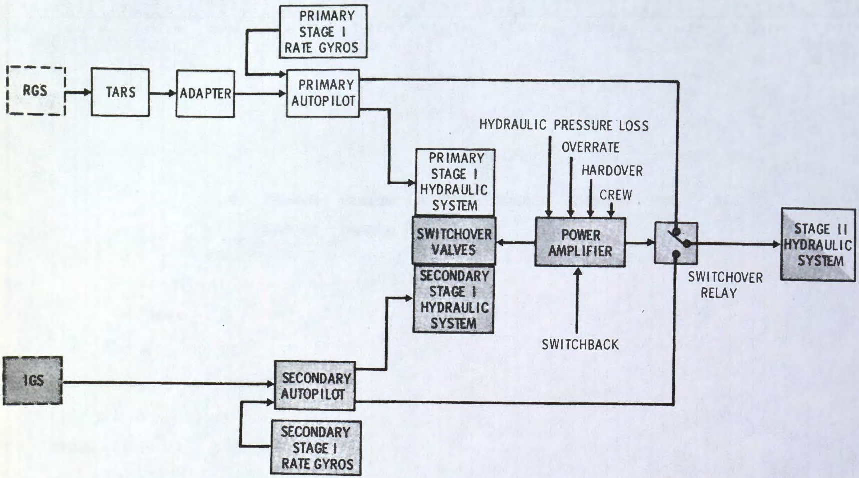


MARTIN

STAGE I REDUNDANT HYDRAULIC SYSTEM



REDUNDANT FLIGHT CONTROL SYSTEM



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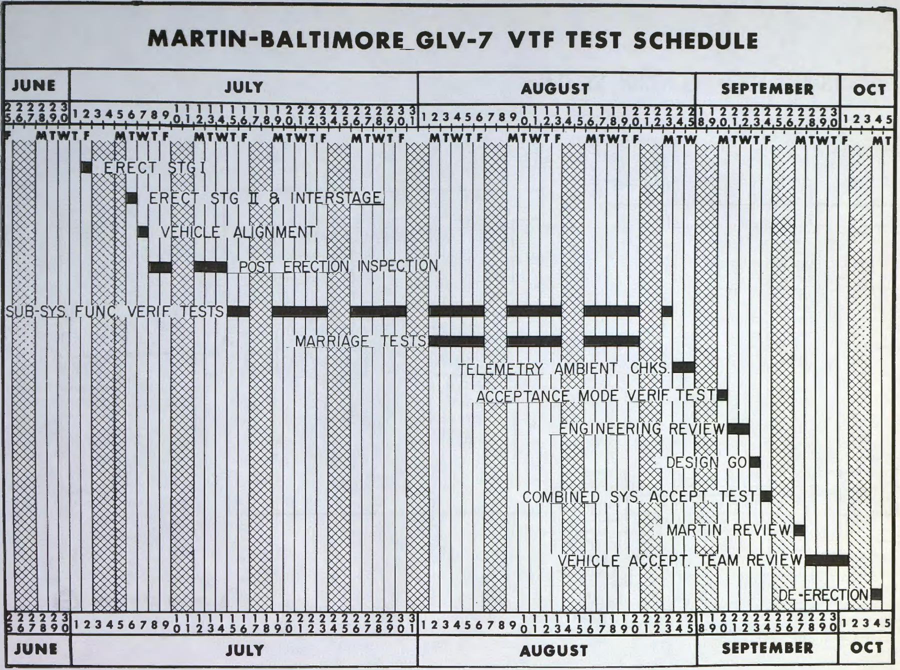

MARTIN-BALTIMORE GLV-7 VTF TEST SCHEDULE
JUNE
JULY
AUGUST
SEPTEMBER
OCT
ERECT STG I
ERECT STG II & INTERSTAGE
VEHICLE ALIGNMENT
POST ERECTION INSPECTION
SUB-SYS. FUNC. VERIF. TESTS
MARRIAGE TESTS
TELEMETRY AMBIENT CHKS.
ACCEPTANCE MODE VERIF. TEST
ENGINEERING REVIEW
DESIGN GO
COMBINED SYS. ACCEPT. TEST
MARTIN REVIEW
VEHICLE ACCEPT. TEAM REVIEW
DE-ERECTION

GLV-6 TREND DATA MONITORING

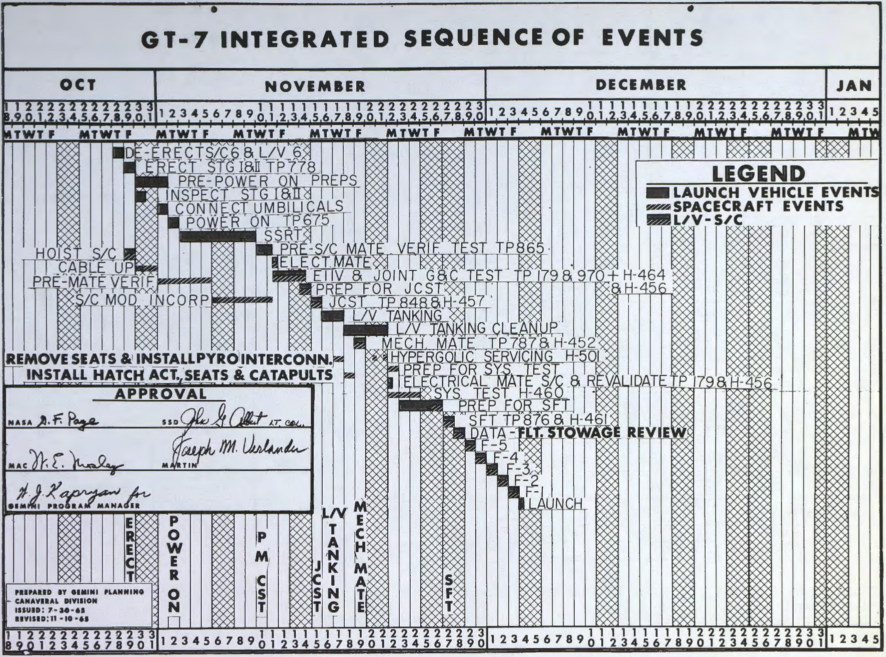
Subsystem Flight Controls

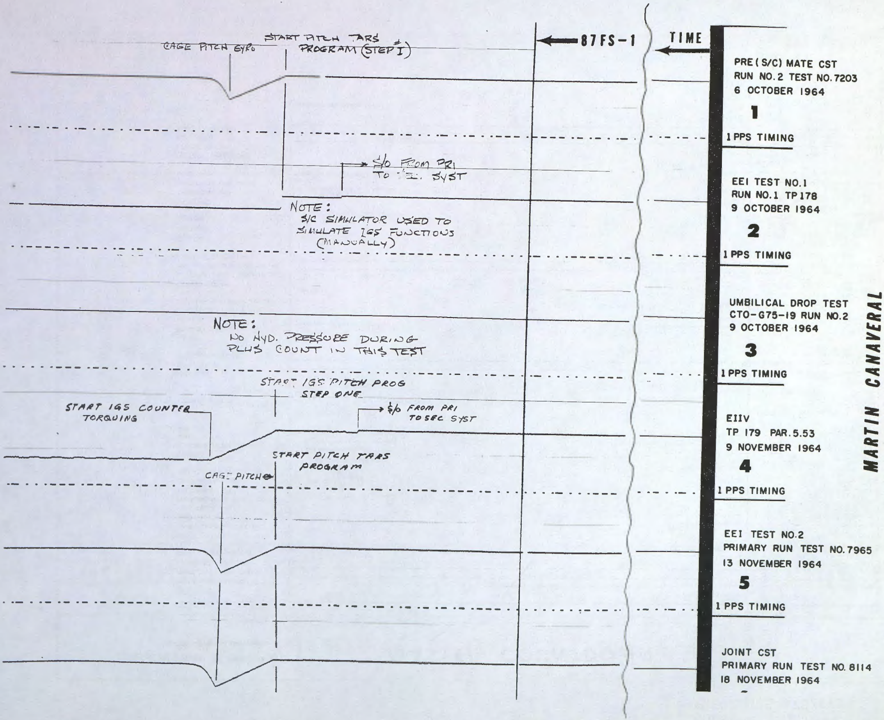
Line No.	Meas No.	Parameter	Spec or Nominal Value and Tolerance	Time Ref. T±, LO±	VTF Tests: CSAT	VTF Tests: EIIV (ETR)	ETR Tests: Pre-SC Mate	ETR Tests: JCST	ETR Tests: FCMT	ETR Tests: WMSL	ETR Tests: SFT	
					Date 6-25-65 Test No. 011/012	Date 9-20-65 Test No. 5750	Date 9-16-65 Test No. 5547	Date 9-23-65 Test No. 5751	Date 10-1-65 Test No. 5901	Date 10-7-65 Test No. 6000	Date 10-20-65 Test No. 6260	Date Test No.
1.	0727	800 CPS Inverter Voltage	24.83to 27.17 Volts AC		25.66	25.67	25.67	25.67	25.67	N/A	25.80	
2.	0728	TARS Discreet Gain Change #1	LO + 104.96 ± 1.07 sec.		104.88	109.65 (See Note 1)	109.8	109.90	110.63	N/A	109.90	
3.	0735	TARS Discreet Arm Stage I Shutdown	LO + 144.64 ± 1.47 sec.		144.53	144.05	144.4	144.46	145.17	N/A	144.50	
4.	0739	TARS Discreet Arm StageII Shutdown	LO + 317.44 ± 3.19 sec.		317.03	(Discontinued)						
5.	0740	TARS Discreet Guidance Initiate	LO + 162.56 ± 1.66 sec.		162.33	162.25	162.3	162.40	163.06	N/A	162.40	
6.	0151 0152	Roll Program	-.042 to -.086 °/s		-0.060 -0.062	N/A N/A	-0.063 -0.063	-0.0619 -0.0649		N/A	-0.062 -0.062	
7.	0150 0153	Pitch Program First Step	∓.095 to ∓.133 °/s		-0.116 +0.117	N/A N/A	-0.116 +0.117	-0.114 +0.115		N/A	-0.114 +0.115	
8.	0150 0153	Pitch Program Second Step BGC	∓.059 to ∓.107 °/s		-0.085 +0.088	N/A N/A	-0.081 +0.080	-0.083 +0.086		N/A	-0.081 +0.082	
9.	0150 0153	Pitch Program Second Step AGC #1	∓.007 to ∓.047 °/s		-0.025 +0.025	N/A N/A	-0.025 +0.025	-0.025 +0.029		N/A	-0.027 +0.025	
10.	0150 0153	Pitch Program Third Step	∓.004 to ∓.020 °/s		-0.010 +0.015	N/A N/A	-0.011 +0.014	-0.010 +0.014		N/A	-0.011 +0.011	
		Configuration			-089 4	-089 5	-089 5	-089 6		N/A 7	-089 8	

NOTES:

Note 1 : Ecp 517R2 added a time delay relay; New time and tolerance effective all ETR tests is 110.00 ± 1.22 seconds.

2. Launch Attempt on 10/25 65 configuration was number 8.


GT-7 INTEGRATED SEQUENCE OF EVENTS
OCT
NOVEMBER
DECEMBER
JAN
LEGEND
LAUNCH VEHICLE EVENTS
SPACECRAFT EVENTS
L/V - S/C
DE-ERECT S/C 6 & L/V 6
ERECT STG I & II TP 778
PRE-POWER ON PREPS
INSPECT STG I & II
CONNECT UMBILICALS
POWER ON TP 675
SSRT
PRE-S/C MATE VERIF. TEST TP 865
HOIST S/C
ELECT. MATE
CABLE UP
ETIV & JOINT G&C TEST TP 179 & 970 + H-464 & H-456
PRE-MATE VERIF.
PREP FOR JCST
S/C MOD. INCORP.
JCST TP 848 & H-457
L/V TANKING
L/V TANKING CLEANUP
MECH. MATE TP 787 & H-452
REMOVE SEATS & INSTALL PYRO INTERCONN.
HYPERGOLIC SERVICING H-501
INSTALL HATCH ACT., SEATS & CATAPULTS
PREP FOR SYS. TEST
ELECTRICAL MATE S/C & REVALIDATE TP 179 & H-456
SYS. TEST H-460
PREP FOR SFT
SFT TP 876 & H-461
DATA - FLT. STOWAGE REVIEW
F-5
F-4
F-3
F-2
F-1
LAUNCH
APPROVAL
NASA D. F. Page
SSD John G. Albert LT. COL.
MAC W. E. Mosley
Joseph M. Verlander
MARTIN
H. J. Kapryan for
GEMINI PROGRAM MANAGER
ERECT
POWER ON
PM CST
JCST
L/V TANKING
MECH MATE
SFT
PREPARED BY GEMINI PLANNING
CANAVERAL DIVISION
ISSUED: 7-30-65
REVISED: 11-10-65



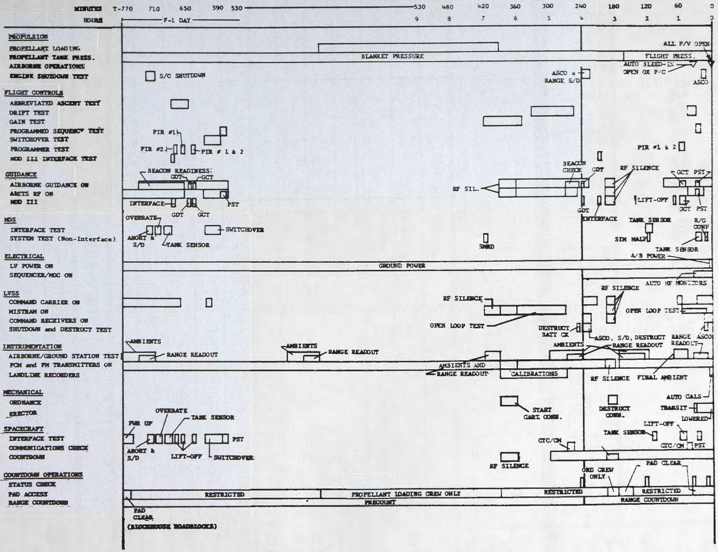
MARTIN CANAVERAL
DATA ENGINEERING

MEASUREMENT NO. 0150 RANGE + 1.25 INCHES

DESCRIPTION TRAVEL ACT. NO. 1 PITCH

MARTIN COMPANY
CANAVERAL DIVISION

MASTER COUNTDOWN



PCN 20
22 Nov. 65

BOOK 20

NUMBER 424 970/ETR

MARTIN **GEMINI**

TEST PROCEDURE

TITLE

JOINT GUIDANCE AND CONTROL TEST

	APPROVAL	DATE
PREPARED BY:	D. STEWART by J.R.B.	6/14/65
PROJECT ENGINEERING:	[signature]	6/14/65
SYSTEM LEAD:	[signature]	6/14/65
TEST CONDUCTOR:	[signature]	6/14/65
QUALITY:	Chas. Dwyer	6/14/65
SAFETY:	D. Gilbert	6/14/65
PILOT SAFETY:	[signature]	6/14/65
CONFIGURATION MANAGEMENT	C.W. Berkebile	6/14/65
6555 TH ATW COORDINATION	Leland G. Fay Capt	6/30/65
ENGINEERING PUBLICATIONS:	[signature]	7/1/65

REVISIONS

SYM.	DESCRIPTION	DATE	APP'D
D	Incorporates PCNs 9 thru 15	14 June 65	[signature]
	Reprinted on 11 October with PCNs 16		
	thru 21 inserted.		

MODEL	RUN NO.	NUMBER
GLV		424 970/ETR