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GEMINI LAUNCH VEHICLE

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Introduction

This paper presents a broad technical description of the changes made to the Titan II ICBM to enable it to perform the Gemini mission. In effect these changes created an essentially new product, the Gemini Launch Vehicle.

The data presented in this paper has been collected from numerous program documents.

Program Objective

The purpose of this program is to develop launch vehicles which will place the Gemini Spacecraft in trajectories designed to meet the following operational objectives:

(1) Perform a 14-day earth orbital flight.

(2) Demonstrate that the spacecraft can rendezvous and dock with a target vehicle in orbit.

(3) Develop simplified spacecraft and launch vehicle countdown techniques in order to optimize the rendezvous mission.

(4) Develop a fully reliable man-rated launch vehicle system.

Mission and Performance

Mission

The objective of the basic launch vehicle is to inject the spacecraft into orbit at an altitude of 87 nautical miles with sufficient overspeed to maintain a perigee of 87 nautical miles and an apogee of 161 nautical miles.

The general trajectory mechanization for the Gemini Launch Vehicle is similar to that used on the basic Titan ICBM, except for inclusion of a variable launch azimuth capability which has been added to meet the conditions imposed by the rendezvous missions.

Sequentially, the Gemini launch is characterized by an engine start signal, followed by a 1.08-second span in which engine thrust is built up to 77%. At that point, the Thrust Chamber Pressure Switch (TCPS) activates a two-second timer and, at the end of that period, the launch bolts are blown and liftoff begins. Then follows a vertical rise of approximately 20 seconds. During the vertical rise, the roll program is inserted to obtain the desired launch azimuth. The first of three open loop pitch commands is initiated approximately 20 seconds after liftoff in order to approach a zero lift trajectory during the Stage I flight regime. Figures 1 and 2 show the results of this type of trajectory on a few of the basic nominal design parameters. As in Titan II, a fire-in-the-hole technique is used to separate the first and second stages.

Sustainer flight is guided by a closed loop Radio Guidance System (RGS) which employs an explicit guidance law similar to that used during the Mercury-Atlas program. Figures 1 and 2 show the characteristics of this portion of the trajectory. Injection conditions are supplied by a velocity cutoff signal which is activated through the guidance system at the required attitude and altitude.

Performance

The performance capability of the Gemini Launch Vehicle is shown as a function of altitude and velocity in Fig. 3. For the mission objectives just described, the vehicle is capable of launching a payload weight greater than the combined weight of the Gemini Spacecraft with the adapter.

Fundamentally, the injection altitude chosen for the launch vehicle is governed by the design premise that minimum modifications will be made to the basic Titan II structure. Such parameters as aerodynamic heating, first-stage dynamic pressure, staging dynamic pressure and minimum elevation angle required for guidance were considered in determining this injection altitude (Fig. 4). A concession was made to the flight loads criteria in that the wind environment used for the Gemini Launch Vehicle is reduced in comparison to that normally used on the SM68B vehicles. Explicitly, Avidyne winds are used in this design application as representative of the environment experienced at the Atlantic Missilé Range. Dynamic pressure in the first-stage regime is in excess of that used in SM68B vehicle design. Aerodynamic heating limits, which are derived from SM68B performance, and the minimum angle required for guidance provide the constraints which limit the injection altitude to approximately 87 nautical miles.

Description of Changes From Titan II

As has been mentioned, the Gemini Launch Vehicle is a version of the Titan II. The differences between the two vehicles can be categorized into three classes:

(1) Changes needed to physically adapt the launch vehicle for the spacecraft.

(2) Changes required to accomplish the mission of accurately injecting a spacecraft into an 87-nautical mile orbit with enough overspeed to achieve a 161-nautical mile apogee.

(3) Changes or additions made because men are part of the payload.

In Class 1, the diameter of the top of the vehicle has been increased to 10 feet. No other

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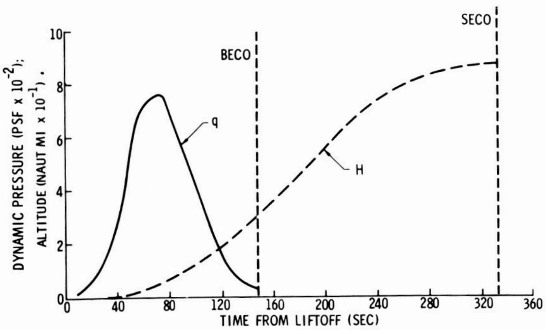

FIG. 1. ESTIMATED TRAJECTORY CHARACTERISTICS

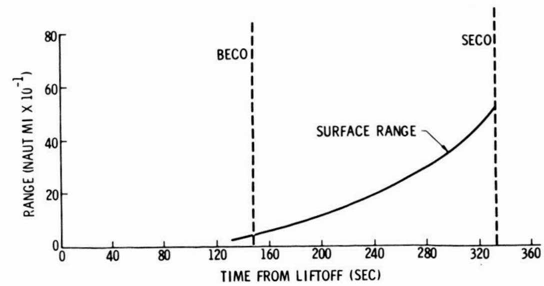

FIG. 2. ESTIMATED TRAJECTORY CHARACTERISTICS

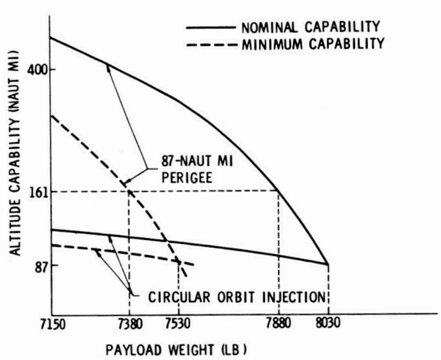

FIG. 3. PERFORMANCE CAPABILITY

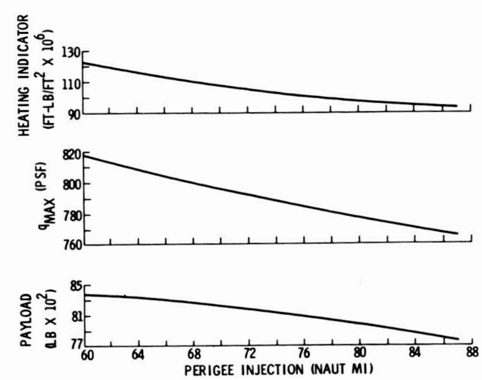

FIG. 4. INJECTION ALTITUDE PARAMETERS

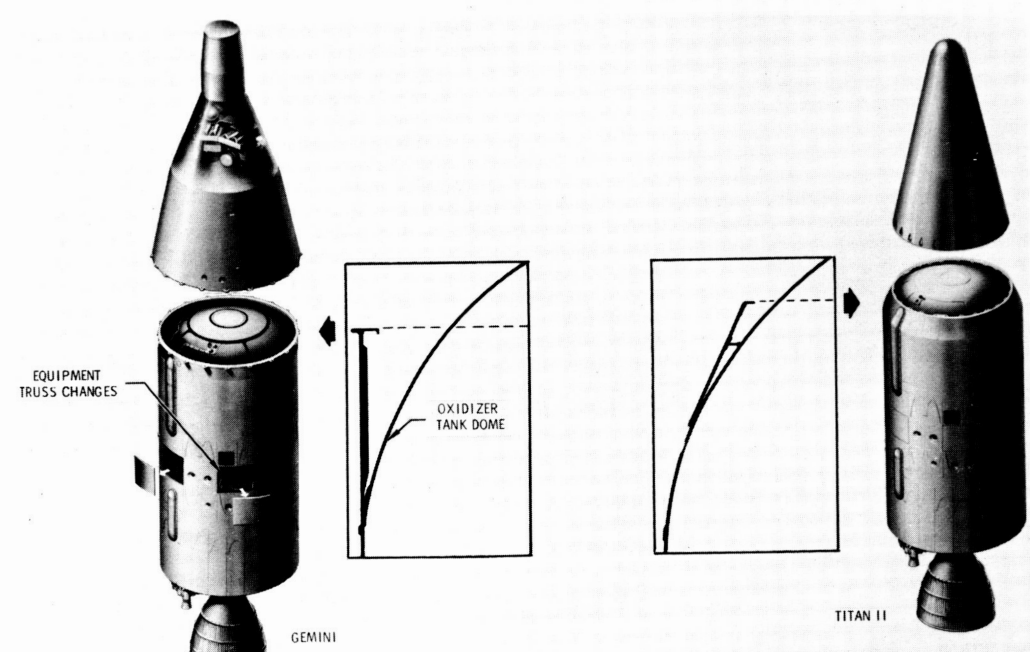

FIG. 5. CLASS I CHANGES

basic changes are required (Fig. 5) because the weight of the spacecraft is less than the maximum warhead weight carried by Titan II, and the trajectory flown will not impose loads which exceed those for which the ICBM was designed.

While some refinements were required, the environment and criteria used for the structural design of the Gemini Launch Vehicle are essentially those of Titan II. Figure 6 shows four major trajectory parameters which directly affect

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the vehicle structural design. Dynamic pressure (q) and axial acceleration are essential to loads calculations, while structural heating is dependent upon the altitude-velocity relationship. The flight path shown in Fig. 6 is one of the numerous trajectories studied in defining the Gemini Launch Vehicle performance requirements. This trajectory is based upon nominal conditions for a 7400-pound payload injected at an orbital altitude of 87 nautical miles at perigee.

All load and structural heating calculations were obtained by using the atmospheric properties given by the 1959 ARDC model atmosphere (NASA Technical Note D595). Figure 7 presents the ground and flight wind profiles used in the loads calculations; as shown, both ground and flight winds represent 1% risk values. The ground wind profile, which is used for prelaunch and launch loads development, is based upon climatic data for Patrick Air Force Base as interpreted by Geophysical Research Directorate, Hamson Field, Bedford, Massachusetts. The first two-thirds of the wind profile is applied as a steady wind condition, while the final one-third is applied as a gust. The flight winds used are those developed by Avidyne for the winter months at Cape Canaveral. A 1-cosine, 20-fps, true gust is added to the Avidyne profiles at any given altitude. In the example shown, the predominant wind is from the west.

Figure 8 shows the net effect for the critical air load condition. The Gemini Spacecraft-Launch Vehicle configuration creates a different air load distribution at the forward end, and this different distribution causes higher internal structural

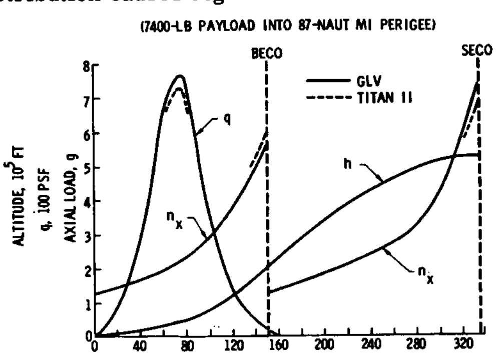


FIG. 6. STRUCTURAL DESIGN TRAJECTORY PARAMETERS

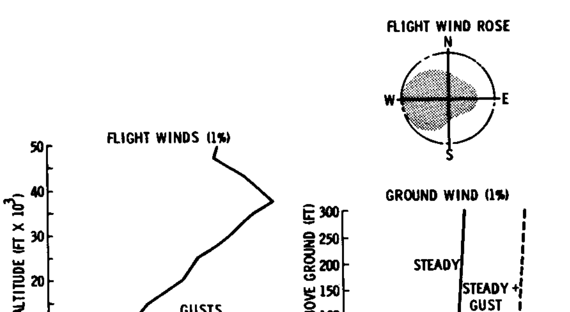
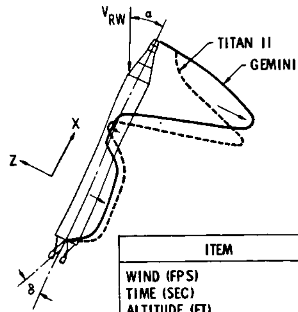


FIG. 7. DESIGN WIND ENVIRONMENT

stresses. These differences are offset by using a lower engine gimbal angle, 3.5 degrees instead of 5 degrees (Fig. 9). The substitution is justified because the control requirements for the most dispersed cases are less than 3 degrees.



ITEM	GEMINI	TITAN II
WIND (FPS)	AVIDYNE (259)	SISSENWINE (300)
TIME (SEC)	69	63.5
ALTITUDE (FT)	36,000	31,000
MACH NO.	1.44	1.27
DYNAMIC PRESSURE (PSF)	685	705
WEIGHT (LB)	216,000	224,000
ANGLE OF ATTACK (α) (DEG)	-10.2	-11.5
ENGINE GIMBAL ANGLE (δ) (DEG)	-3.5	-5.0
AXIAL LOAD FACTOR (η_X) (g)	2.03	1.88
LATERAL LOAD FACTOR (η_Z) (g)	-0.31	0.365

FIG. 8. MAXIMUM AIRLOAD CONDITION

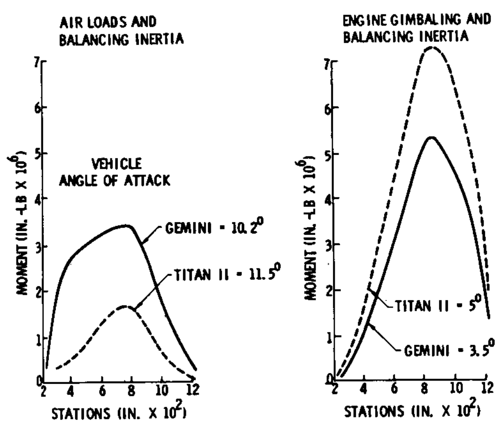


FIG. 9. VEHICLE BENDING MOMENT

Class 2 modifications (Fig. 10) deal with those changes needed to increase the payload capability for the required orbit. The following steps were taken to meet these new requirements.

(1) Delete the Titan II Inertial Guidance System. The Gemini Launch Vehicle system uses a Three-Axis Reference System during the first stage flight and a Radio Guidance System during the second stage. Since the GE Mod III-F is used as a tracking and impact predictor for Titan II, a complete Radio Guidance System (GE Mod III-G) was developed by simply adding a decoder.

(2) Use MISTRAM only on the Gemini Launch Vehicle. Titan II uses both MISTRAM and Azusa tracking equipment.

(3) Remove the Titan II retro and vernier rockets.

(4) Change the instrumentation system from a 0- to 40-millivolt system, to a 0- to 5-volt system.

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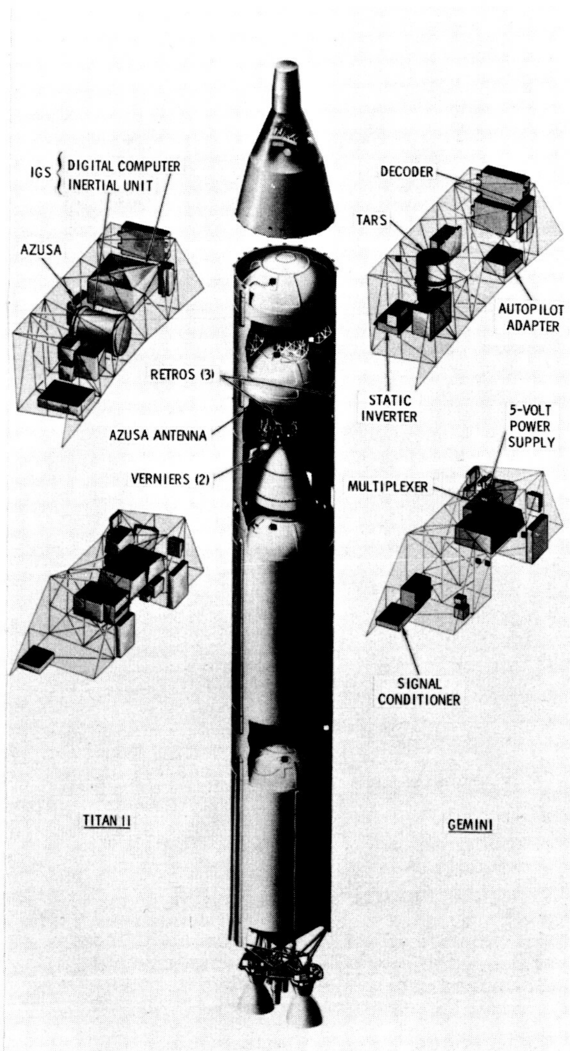

FIG. 10. CLASS 2 CHANGES

Figures 11 and 12 show the modifications made to the guidance and instrumentation trusses in order to adopt the Titan for the Gemini mission.

Table 1 shows three Stage II configurations which have the necessary equipment to perform

TABLE 1

COMPARISON OF THREE STAGE II CONFIGURATIONS FOR THE GEMINI MISSION

Vehicle Part	Titan II N-11 (lb)	GLV No. 5 with RGS & MISTRAM (lb)	GLV with IGS & MISTRAM (lb)
Body	2,262(1)	2,262	2,262
Separation and destruct	66	82	82
Propulsion	1,328	1,332(2)	1,332(2)
Power generation	100	104	104
Static inverter	0	68	0
Orientation controls	338(3)	138	138
Mod 3-F	32	33	0
Decoder	0	14	0
TARS	0	72	0
Autopilot No. 1	38	38	38
Autopilot No. 2	38	38	38
Adapter	0	17	0
IGS	282	0	282
MISTRAM	30	30	30
Azusa	29	0	0
Command receivers	50	50	50
Strobe light	72	0	0
Wire and bracketry	528	254	440
Environmental control	24(4)	14	14
Instrumentation and telemetry	1,074	684	753(5)
MDS	96(9)	96	96
Unaccountable variation	-58	0	0
Translation system	120(6)	0	0
Total Weight Empty	6,449	5,326	5,659
Residual Propellant	445	400(7)	400
Burnout weight	6,894	5,726	6,059
Disposable propellants	58,896	60,083(8)	60,083
Engine bleed	13	11	11
Solid propellants	145	0	0
Starter grain	3	3	3
Gross Weight	65,951	65,823	66,156

NOTES:

(1) Normalized to remove N-11 warhead adapter.
(2) Revised Gemini engine specification weight.
(3) Stated with vernier system weight included (200 pounds).
(4) Reflects ducting in equipment compartment for air conditioning while the vehicle is on pad.
(5) Includes AC-Spark Plug (IGS) telemetry packages.
(6) Used to rotate the burned out Stage 2 out of the flight path of the payload after separation.
(7) Based on propellant loading statement issued 20 February 1963. These values are nominal and include mean outage.
(8) Based on cold propellant loading statement issued 20 February 1963.
(9) Included to normalize comparison basis.
(10) All weights include malfunction detection and redundancy provisions.

TABLE 2

INCREASED PROPELLANT AND PAYLOAD

Items	Stage I (lb)	Stage II (lb)	Total (lb)
Cold propellant	2090	900	2990
Tank volume considerations	1260	200	1460
Total Loaded	3350	1100	4450
Nonusables, transients and bias	650	60	710
Total Steady-State	4000	1160	5160

The increase in payload capability which results can be stated as follows:

Stage	W (lb)	Payload (lb)
Stage I		
△ Propellant weight	4000	133
Stage II		
△ Propellant weight	1100	73
△ Empty weight		1168
△ Total Payload Gain		1374

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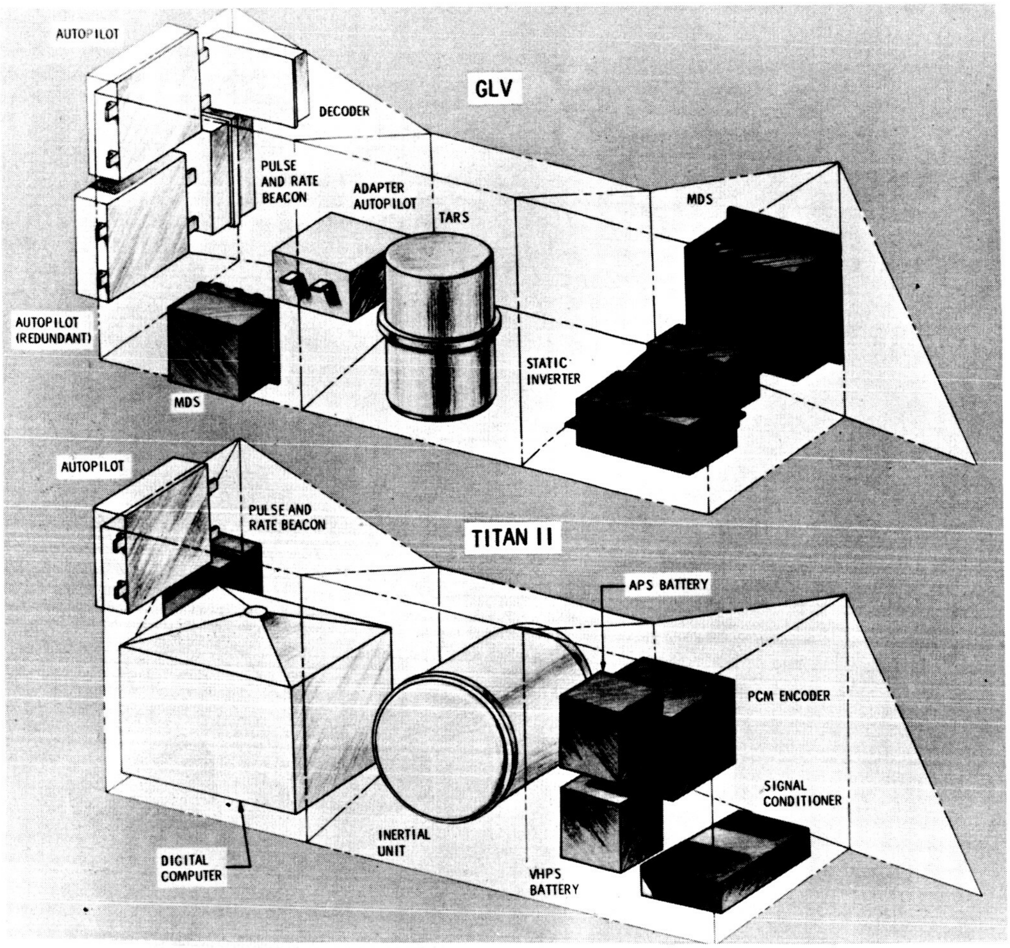

FIG. 11. EQUIPMENT TRUSS NO. 1 (GUIDANCE)

a Gemini mission. The tabulation indicates that a payload increase of 1168 pounds was realized because of the differences between the Titan II research and development ship No. 11, which served as the base for the Gemini Launch Vehicle, and the Gemini configuration finally chosen. In addition, it is shown that there is a payload differential of 264 pounds between a stripped Titan II with inertial guidance and the final Gemini Launch Vehicle configuration.

Table 2 shows the increased payload and propellant that the Gemini Launch Vehicle is capable of handling. There are four reasons why the Gemini Launch Vehicle can carry this additional propellant: (1) calibrated tanks with nominal rather than minimum values are used; (2) The area between the prevalves and thrust chamber valve can be used for propellant storage; (3) a more accurate loading system is provided; and (4) lower propellant temperatures are maintained. Table 2 shows how the additional 5160 pounds of propellant which can be loaded on the Gemini is distributed.

The preceding tabulation explains the payload gains realized to date; it does not include additional gains that could be effected through:

(1) Reducing ullage requirement and loading more propellant.

(2) Using selective injectors to bring about I_{sp} gains.

(3) Using chambers selected to optimize burning mixture ratios.

(4) Devising additional means of reducing weight.

The changes in instrumentation hardware, some of which resulted in the weight savings just discussed, are summarized in Table 3 and are schematically indicated in Fig. 13. The summary of all the Class 2 changes is shown in Fig. 10.

Class 3 modifications (Fig. 15) deal with those changes which have been introduced to ensure the safety of the two astronauts who will be aboard the spacecraft. The Man-Rating and Pilot Safety Program which was developed to do the job involves many considerations. These are summarized in Fig. 14.

Gemini changes related to hardware are considered under the category of system design. Specifically, the major considerations made in this category can be delineated as:

(1) Addition of a Malfunction Detection System (MDS).

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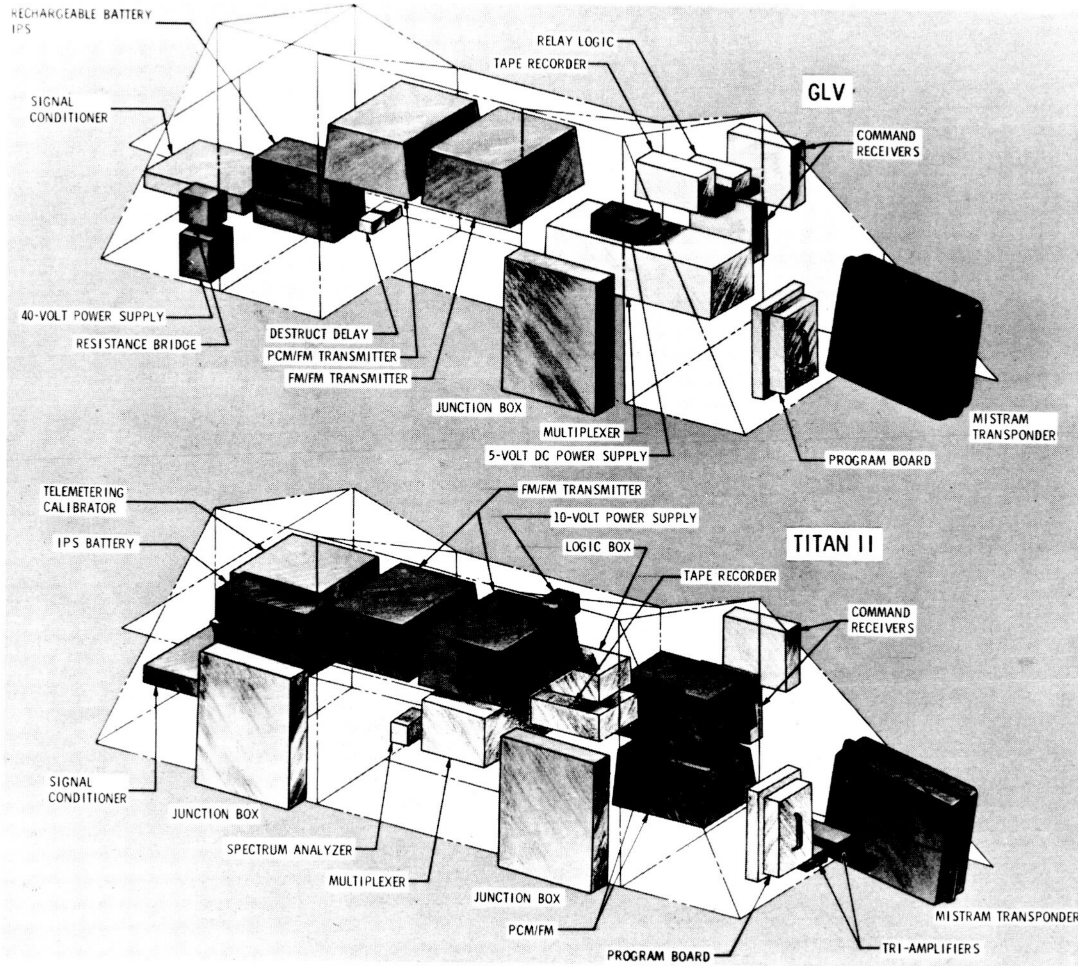

FIG. 12. EQUIPMENT TRUSS NO. 2 (INSTRUMENTATION)

TABLE 3

INSTRUMENTATION SYSTEM

Components	Same as Titan II	Extent of Changes: Remarks
PCM/FM transmitter	Yes	None: Denver supplied
FM/FM telemetry	Essentially (98%)	Five FM low level oscillators changed to high level: Denver supplied
PCM multiplexer	Essentially (90%)	Channel capacity, format, and sampling rates are same as Titan II. Changed input section for Gemini Launch Vehicle to high level, 5 volts. Weight saving on Gemini Launch Vehicle is 25 pounds
Power divider, 5 port	Yes	None
Antenna telemetry, 4 required	Yes	None
Diplexer	No	Repackaged to cover 2 RF links
Program board	Yes	None
Signal conditioner types of modules		
(1) 400-cps phase demodulator	No	Same as Titan II, except that the Gemini Launch Vehicle has a TARS package, while Titan II doesn't
(2) 800-cps phase demodulator	No	Titan I 400-cps phase demodulator has been modified for 800 cps
(3) 115-vac discriminator	No	Same as Titan II; required for Gemini Launch Vehicle because of 400-cps static inverter and TARS
(4) 26-vac discriminator	No	Titan II modified to give 5-volt instead of 40-mv output
(5) 400-cps frequency deviation	No	Same as Titan II; required for Gemini Launch Vehicle because of 400-cps static inverter and TARS
(6) DC amplifier	No	Same as Titan II; high level for current monitoring of IPS and APS
Transducers		
(1) Temperature sensor system	No	Same as Titan II; has a 5-volt output without signal conditioning, replaces thermocouples used on Titan II
(2) Static accelerometer	No	Unit has high level output; similar units on Titan II are low level
(3) Pressure transducer	No	Unit has high level output; Titan II uses low level; sensing element is solid state bridge
Airborne tape recorder	Yes	None
Connectors	Yes	None
Wire	No	Titan II uses twisted pair shielded for each measurement, while Gemini Launch Vehicle uses single conductor shielded. Weight saving 142 pounds

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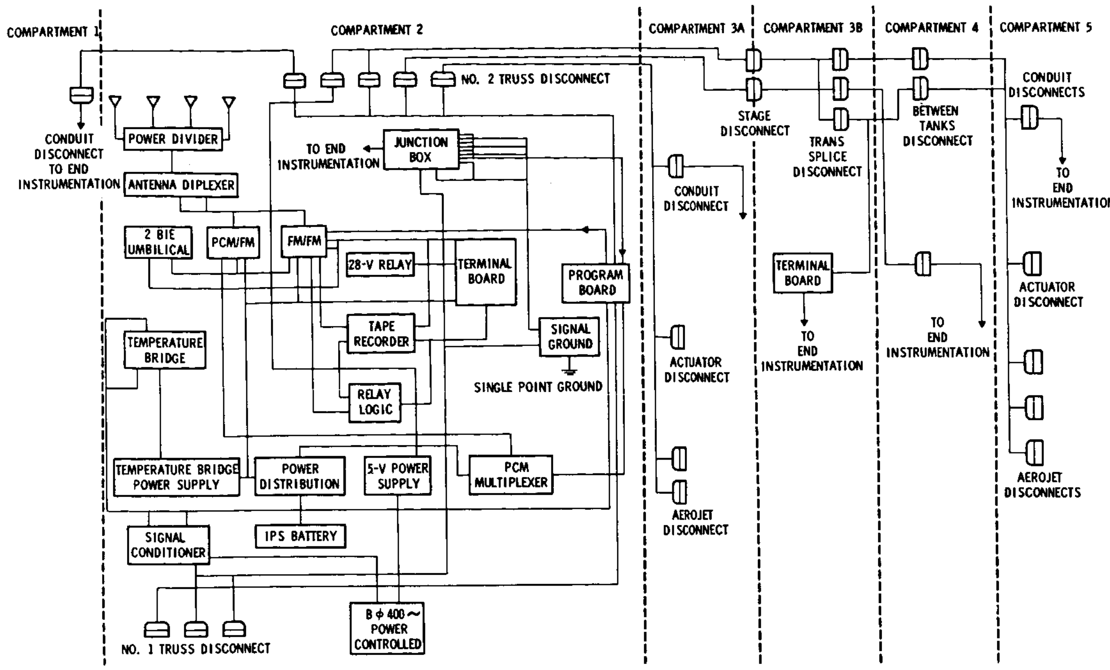

FIG. 13. INSTRUMENTATION SYSTEM

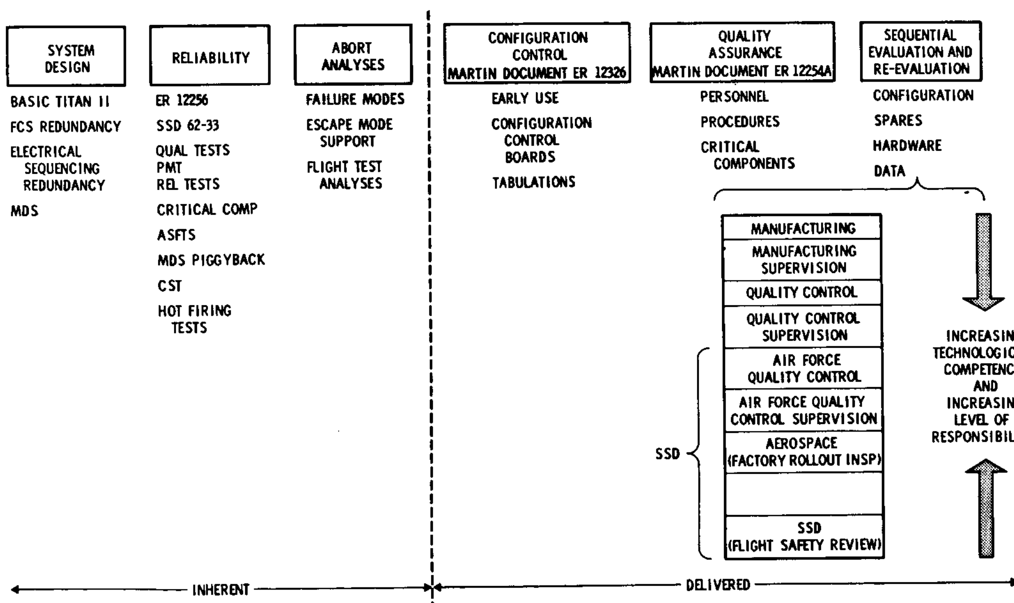

FIG. 14. MAN-RATING AND PILOT SAFETY

(2) Addition of those features required to produce flight control system redundancy.

(3) Addition of time delays in the flight termination system.

(4) Addition of redundancy provisions in the electrical circuits of the flight sequencing system.

Malfunction Detection System (MDS)

Effective implementation of a Man-Rating and Pilot Safety Program, like the one shown in Fig. 14, will ensure a launch vehicle which will perform more reliably. Even though the goal is perfection, realistically, there is always some possibility of hardware failures. In order to minimize losses due to this possibility to the lowest attainable level, a highly sensitive Malfunction

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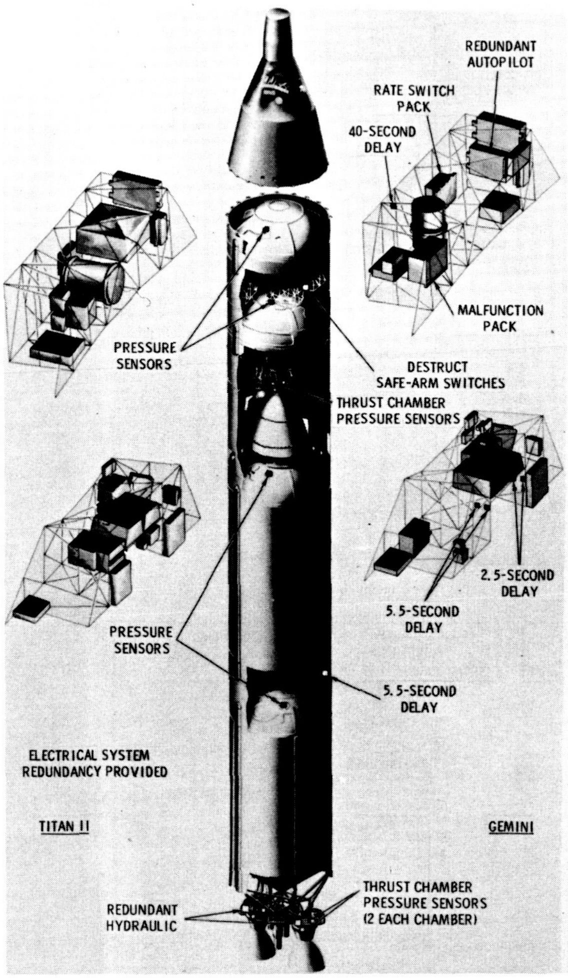

FIG. 15. CLASS 3 CHANGES

Detection System has been incorporated in the Gemini Launch Vehicle. This system (Fig. 16) provides information on those parameters which most significantly affect the safety of the astronauts and the success of the mission.

The fundamental question which must be answered in developing a Malfunction Detection System is, "How will the sensed information be used?" Stated simply, the question can be reduced to determining the degree of automatic action which should result; that is, should the sensed information cause automatic ejection or should the information be displayed to the pilots who would then decide what to do. Before a valid decision can be made, the following factors must be considered.

(1) Time histories of launch vehicle action following anomalies.

(2) The time in which anomalies may be sensed and displayed.

(3) The extent to which "cues" other than hardware sensing will be available and useful.

(4) The relative complexity and reliability of an automatic versus a manual system.

(5) The astronaut's role: the role which is desired and the contribution which can be made.

(6) The mission requirements effect.

(7) The escape system concept.

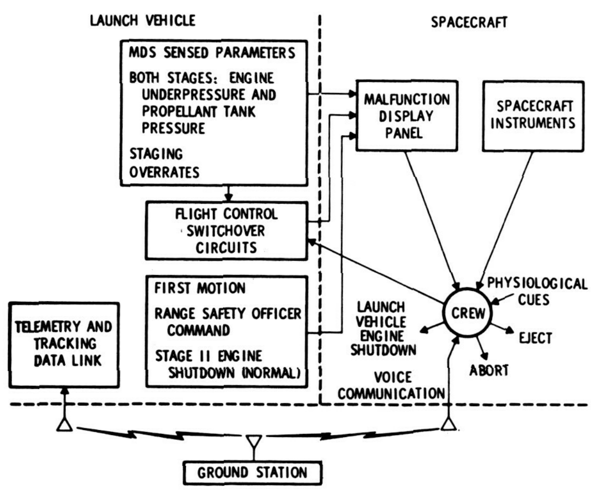

FIG. 16. MALFUNCTION DETECTION SYSTEM

Although these factors can be evaluated independently, many of them are necessarily interrelated. For example, in the case of the Gemini Launch Vehicle, Items 4, 5, 6 and 7 were intermeshed and basic decisions in these areas indicated a need for a manual rather than an automatic abort system. However, this meant that Items 1, 2 and 3 had to be evaluated in order to determine whether a safe manual system could be developed. Once it was proven that such a system could be provided, the Gemini Malfunction

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Detection System was implemented to provide information to the astronauts who must ultimately decide what action is to be taken.

Project Gemini's design philosophy is summarized effectively in a February 1963 article in "Astronautics and Aerospace Engineering" by Chamberlain and Meyer. An analysis of a few quotes from this article enables one to understand the need for a manual abort system.

> The Atlas is so instrumented that it will automatically abort the Mercury Spacecraft if any one of a number of malfunctions is sensed in the launch vehicle. The automatic abort modes in Mercury are very complicated and have caused the loss of complete spacecraft in the early development unmanned flights. In each instance, had a man been on board, he could have manually salvaged the situation.
>
> In Gemini, a launch vehicle malfunction activates lights and gages on the instrument panel and the astronauts exercise judgment as to the seriousness of the situation and the best procedure to follow during any special circumstances. With this sort of system, more than one cue can be used to verify an abort situation. Simulations reveal that in many cases, much reliance is placed on the audio-kinesthetic cues for this purpose. These cues are not only very reliable, but instill confidence in the pilots in the validity of the systems when they are checked by this means.

A further quote from this article shows that one of six primary objectives of the program is:

> To perfect methods for returning and landing the spacecraft on a small preselected landsite. This objective involves re-entry control and a paraglider for spacecraft recovery. The ejection seats not only provide a substitute for a reserve parachute, but also provide an escape mode both early in flight and on landing.

This latter quote is offered to indicate some of the background that led to the choice of ejection seats as one of the escape modes. Their use and speed of reaction is one of the factors that was considered in deciding whether a manual abort system was feasible.

The factors just evaluated cover Items 4, 5, 6 and 7 of the characteristics which had to be considered in evaluating the desirability of a manual versus an automatic abort system. Logically, the next step in such an evaluation was to examine all possible malfunctions in order to determine the more critical malfunction times.

The first step in such an analysis was to determine the frequency of failures by systems. Primarily, this information was gathered by reviewing Atlas, Titan I and Titan II histories. During these analyses, the following information was particularly sought:

(1) Probability of occurrence

(2) Mode of failure.

(3) Time until critical limits are exceeded.

From these studies, a summary of what might be expected on the Gemini Launch Vehicle was prepared; the summary indicated the probabilities of malfunction by systems (Fig. 17). Each system was then considered independently, and the consequences of a failure at different times during the flight on better than 1000 analog simulations of this kind were made for the Gemini Launch Vehicle Program. Typical results of these studies are shown in Figs. 18, 19, 20 and 21. From these data, the time required to reach a critical limit

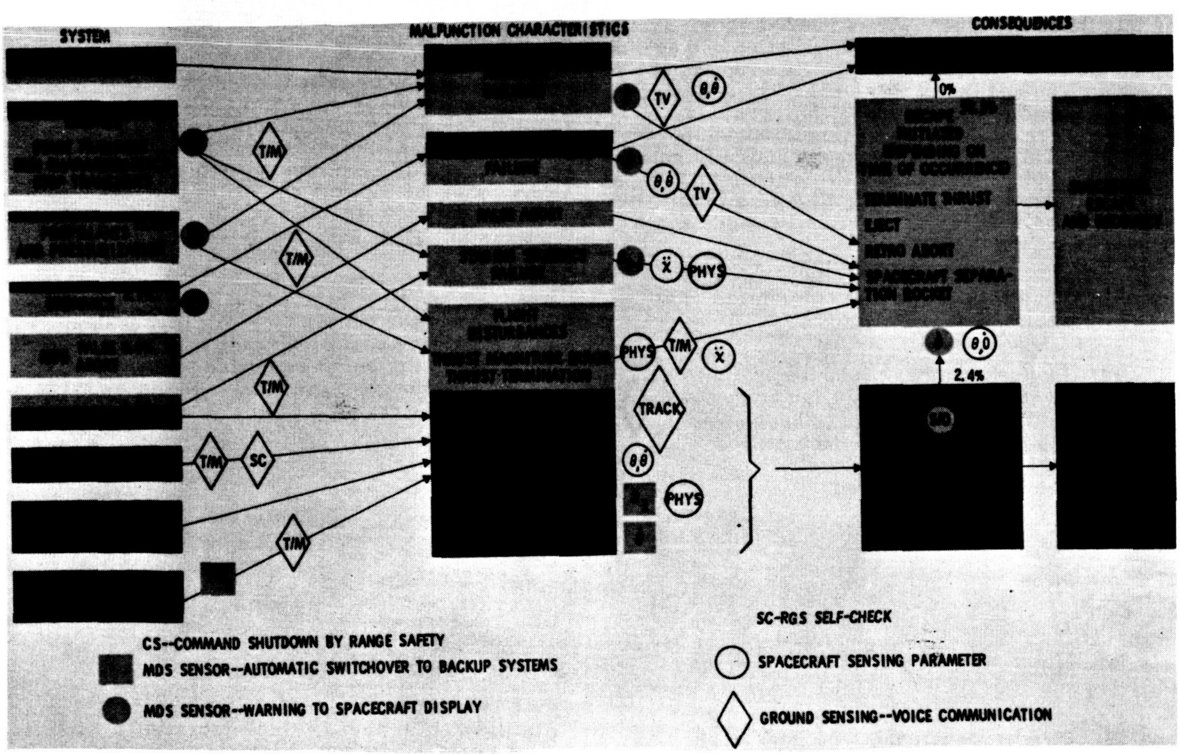

FIG. 17. ORGANIZATION OF MALFUNCTION PROBLEM

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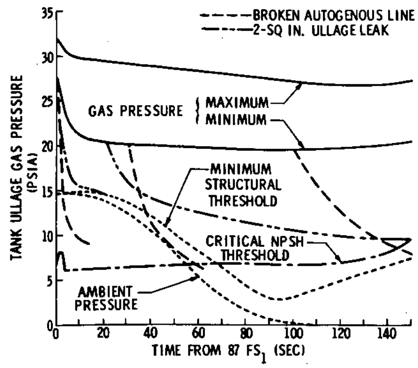

FIG. 18. STAGE I FUEL TANK MDS REQUIREMENTS

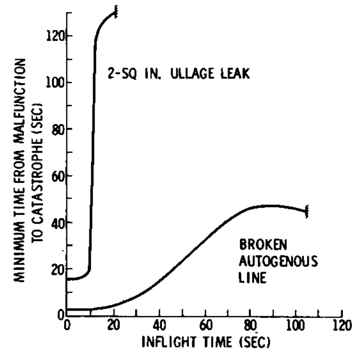

FIG. 19. STAGE I FUEL TANK TIME TO CATASTROPHE

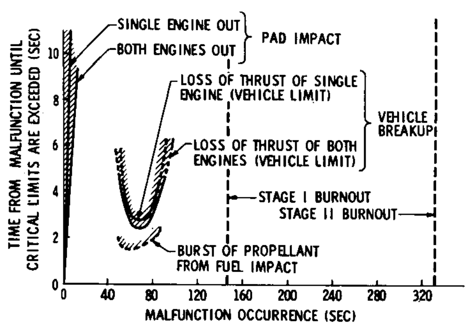

FIG. 20. INADVERTENT THRUST TERMINATION

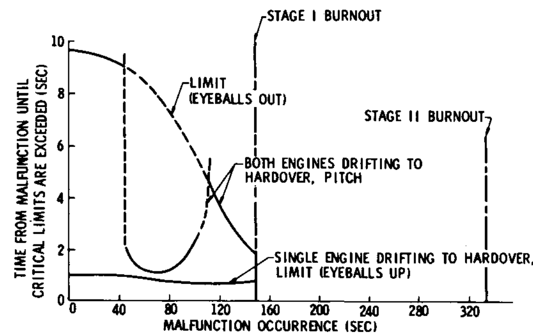

FIG. 21. VIOLENT CONTROL MALFUNCTION

was determined. For example, Fig. 20 shows that if an engine failure occurs at approximately 70 seconds, the vehicle would break up in approximately three seconds. With a manual abort system, the sensing, indication, reaction and escape actions would all have to occur within three seconds. The results of these analyses indicated that it is possible to react to all failures in a timely manner, with the exception of engine hardover cases which will be discussed under Flight Control System Redundancy. From these analyses, it was determined that the following parameters must be monitored while the Gemini Launch Vehicle is in flight:

(1) Four tank pressures (structural limit or minimum NPSH).

(2) Engine chamber pressure switches set at 68% of rated thrust for Stage I and 65% for Stage II; this is equivalent to 550 psia ±30 psi for both stages.

(3) Vehicle attitude rates.

	Stage I (deg/sec)	Stage II (deg/sec)
Pitch	+3.5, -4	10
Yaw	±3.5	10
Roll	20	20

(4) Staging signal: the light goes on at staging signal (87 FS_2, 91 FS_1) and goes off at separation approximately 87 FS_2 + 0.6 second.

The tank pressure sensors provide analog signals to the spacecraft indicators. Redundant sensors, which are connected in independent, parallel circuits individually routed to the spacecraft, are supplied for each tank. All other sensors are bilevel. They are also redundant for each parameter, but, in this case, they are connected in series. Consequently, the contact of both sensors in the redundant pair must be closed before a signal is initiated (Fig. 22).

In addition to the parameters measured in flight, sensors have been added in those lines which contain the propellant tank pressurants. These sensors measure whether gas for the tank pressurization is being generated to a value which will be high enough to pressurize the tanks. The values sensed are:

Values	Stage I	Stage II
Fuel	50 ± 4 psi	None
Oxygen	385 ± 25 psia	None

If the sensed values are not high enough, an engine kill is initiated prior to liftoff.

In addition to the flight considerations, there are ground abort conditions which also had to be evaluated. These conditions are shown in Fig. 23.

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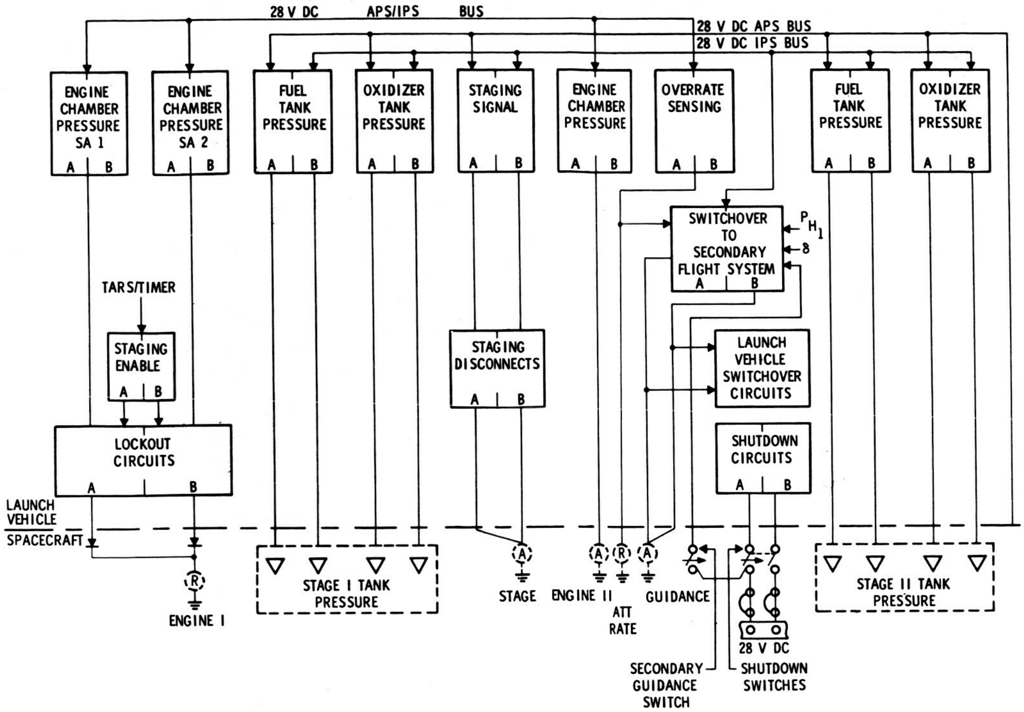

FIG. 22. DETAILED MALFUNCTION DETECTION SYSTEM

The figure shows that the Gemini recovery area is being cleared and leveled for recovery of the two Gemini pilots in the event of a pad abort. The legs of this triangular-shaped area are each 1000 feet long and the angle between them is 54 degrees. All elevated obstacles are being removed; even pad illumination lights will be installed flush in the ground. The highlighted area (dashed line) will be deluged with water in the case of booster explosion. In present Gemini capsule design, the pilot's seats are angled 9 degrees above horizontal and 12 degrees apart. The ejection motor on each seat will develop 2500 pounds of thrust and burn for 1 second; pilot should be clear of capsule 0.4 second after motor ignition. Barostats will activate seat-mounted chutes 3 seconds later when the pilots are about 300 feet above the ground. Pilots will have a maximum 5.5 seconds in which to initiate escape procedures after notification from Range Safety Officer of his intention to destroy a malfunctioning booster.

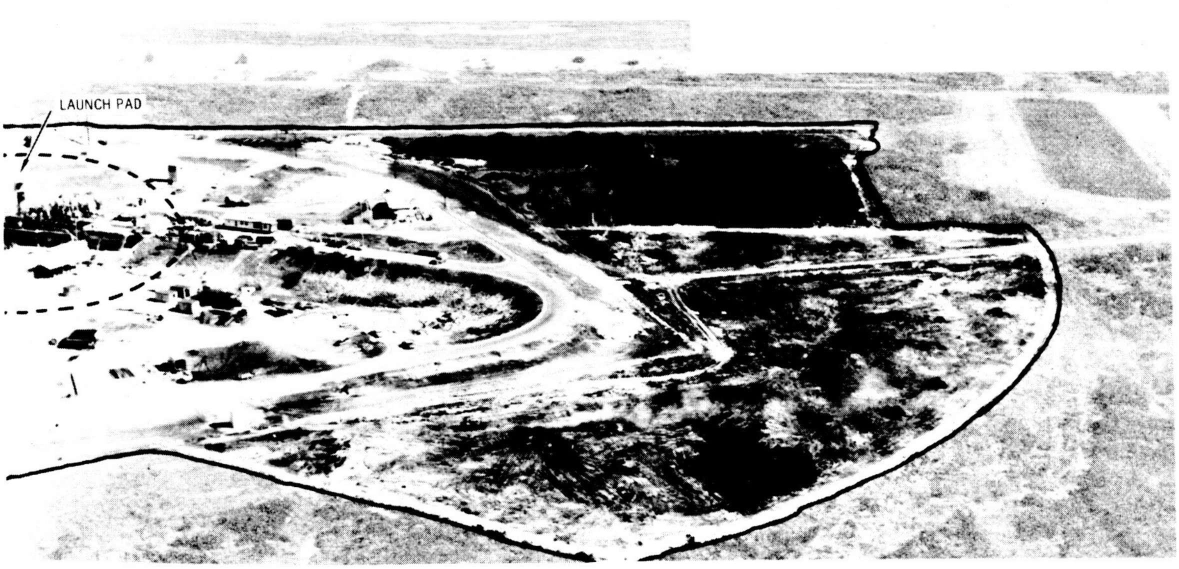

FIG. 23. COMPLEX 19 RECOVERY AREA

[redacted]

One switch will eject both seats. Ejection seats will be the primary escape mode up to 70,000 feet. After that, pilots will escape by firing the spacecraft's solid propellant retrorockets, each developing 2500 pounds, and separating the capsule from the launch vehicle. Pilots would then fly their capsule back to earth by paraglider. NASA, Martin and McDonnell are studying ways of pilot escape from the launch stand before the erector is dropped, preparatory to engine ignition. These include a cherrypicker, high-speed elevator, cork-screw type slide and lifelines.

The times at which the remaining escape modes (use of spacecraft retrorockets or longitudinal spacecraft maneuver rockets) would be used are shown in Fig. 24.

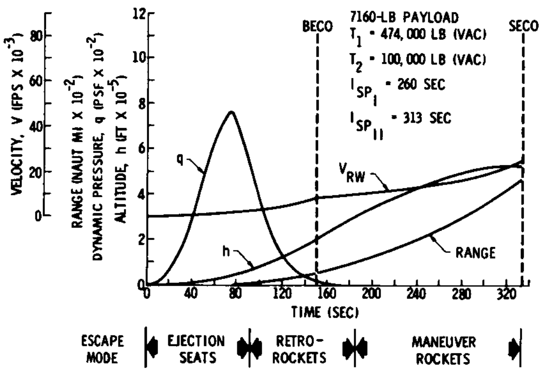

FIG. 24. ESCAPE MODE FLIGHT PARAMETERS

Flight Control System Redundancy

As previously indicated, analyses were made for a number of postulated malfunctions to determine how much time would elapse from the instant when a malfunction was sensed until critical limits were exceeded. These times were then examined to define whether there was sufficient time for pilot warning and reaction. The engine hard-over condition, that is a failure in the flight control system or hydraulics which causes or allows one or two engines of Stage I to drift hard-over, was examined carefully. Figure 21 shows the time histories accumulated during these analyses. As seen, it takes approximately 1.25 seconds to reach vehicle destruction if both engines drift to hard-over in pitch and one second or less to reach a physiological limit should a single engine drift hard-over and cause a yaw-roll buildup.

In order to determine whether there would be enough time for astronaut reaction for this and other cases, NASA decided to conduct a series of experiments. These were conducted at Chance Vought in a simulator where the malfunctions were simulated and response time measured. In all cases, except those for engines hard-over, there was sufficient time for positive astronaut reaction. In no case was the time for engine hard-over met.

These experiments showed that a manual abort system was desirable, possible and practical, except in the case of engine hard-over. The question then remained as to whether an automatic abort should be provided for this condition or whether some compensatory method could be devised. A number of studies were made to determine the effect of various degrees of redundancy. These studies showed that the most effective system was one in which redundancy was provided from guidance through the flight control systems and to the hydraulics of Stage I (Fig. 25). With this system, the probability of an engine hard-over failure is reduced appreciably, while the probability of mission success is increased significantly from 90 to 93.6% (Fig. 26).

The effect of sensing and switchover to maintain the vehicle within structural limits is shown in Fig. 27. Switchover to the secondary system can be effected by four methods:

(1) Command from the pilot.

(2) Detection of vehicle overrate by MDS rate sensors.

(3) Loss of Stage I primary hydraulic system pressure.

(4) Positioning of Stage I hydraulic actuator.

Pilot command is initiated manually by the astronaut. These decisions are based on the pilot's interpretation of the spacecraft display,

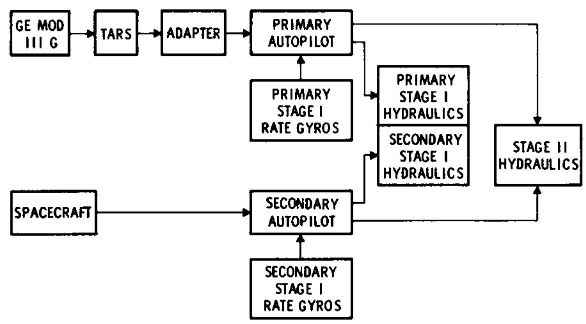

FIG. 25. REDUNDANT GUIDANCE AND FLIGHT CONTROLS SYSTEMS

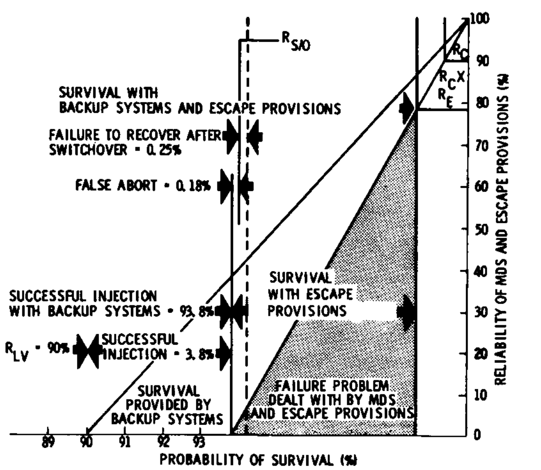

FIG. 26. SURVIVAL CONSIDERATIONS

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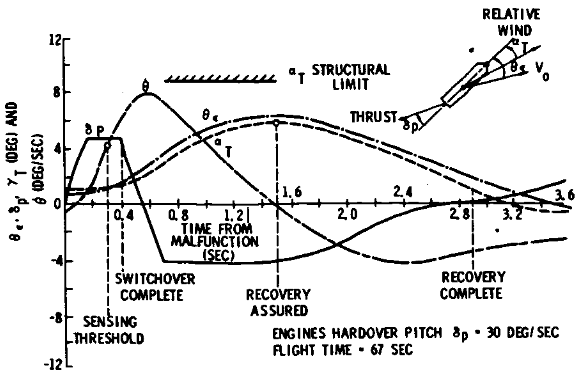


FIG. 27. SIMULATION OF CONTROL MALFUNCTION EVENT

plus information which he receives from the ground station. The MDS overrate sensors will automatically initiate a signal when the vehicle's motion exceeds a predetermined safe limit. In addition, the hydraulic pressure switch automatically initiates switchover when the pressure on the primary side is reduced to a preset value.

Each of these methods produces a signal which simultaneously energizes the hydraulic switchover valve solenoids in the Stage I hydraulic system, and a relay which switches the Stage II hydraulic system input signals from the primary to the secondary autopilot.

Flight Termination System

Except for the following differences, the Gemini Launch Vehicle flight termination and destruct system (Fig. 28) is the same as that used on Titan II (N-I).

(1) Crew safety switches have been added between the airborne 28-v d-c power supply and the destruct switches.

(2) The 28-v d-c power is isolated from the destruct switches until after flight termination system shutdown command has been initiated.

(3) Time delay relays have been added to prevent the flight termination system from giving a destruct command until 5.5 seconds have elapsed after the shutdown command has been initiated.

(4) Time delay relays (5.5 seconds) have been added to the Stage I automatic destruct system; consequently, the system reacts only if there is an inadvertent separation of Stage I from Stage II during the boost phase.

(5) Stage I is shut down and destroyed if it inadvertently separates from Stage II during boost phase.

(6) The Stage I inadvertent separation destruct system is made safe at approximately 10 seconds prior to normal separation by independent signals transmitted from both the Three-Axis Reference System and 140-second timers.

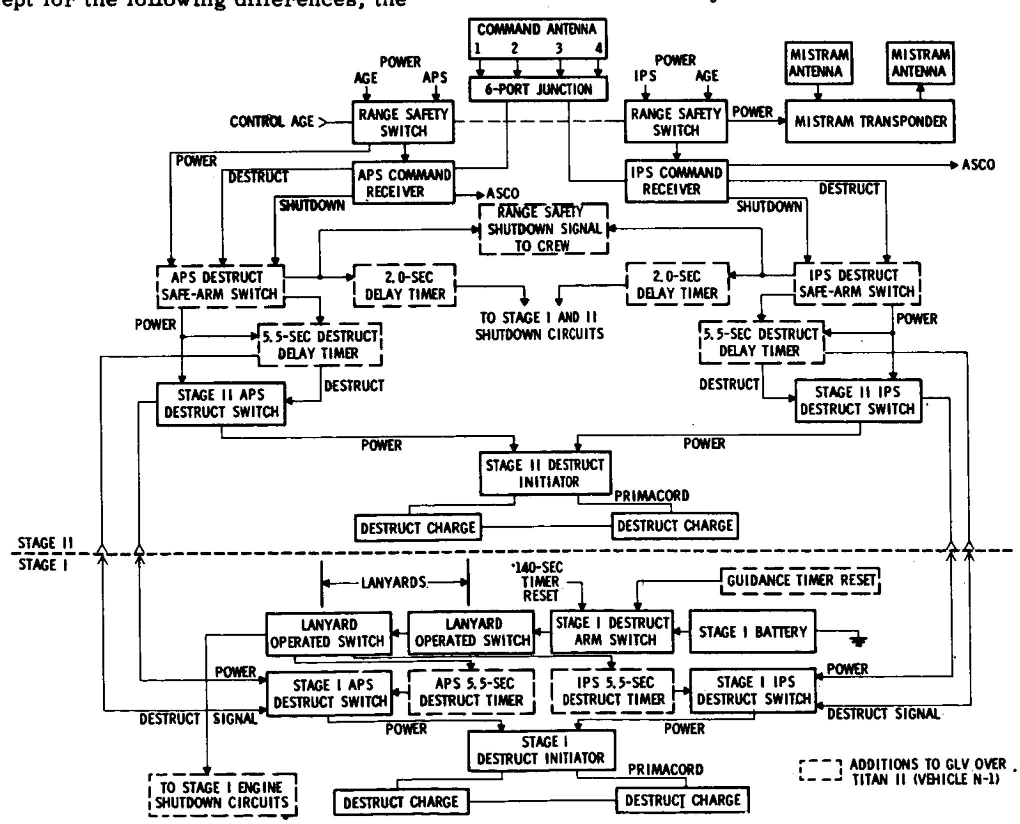


FIG. 28. RANGE SAFETY SYSTEM

[redacted]

Stated simply, these changes, which have been made to protect the men aboard, provide information with respect to Range Safety Officer action and adequate time for independent astronaut action. A summary showing the specific escape mode against the time of flight during which the mode would be employed is shown in Fig. 18. As further evidence of the planning which has been done to provide maximum crew safety, Fig. 29 shows a summary view of tracking, flight termination and destruct systems actions which occur prior to and after launch.

Figure 30 shows the flight termination sequence times during the various modes of escape. Vehicle destruct is accomplished by another independent action and a signal from the Range Safety Officer following destruct enable.

Gemini Electrical Sequencing

The addition of the Malfunction Detection System and the modifications made to the guidance system brought about a number of changes in the electrical sequencing circuits. Since the basic design had to be changed, it was decided that the maximum degree of redundancy, within the context of the change, should be provided. Essentially, redundancy was achieved through the circuit wiring design without adding any new components. Table 5 compares the Gemini and Titan II electrical sequencing systems.

The controlling electrical sequencing system for the Gemini Launch Vehicle consists of the motor driven switch and relay logic which is required to perform such functions as:

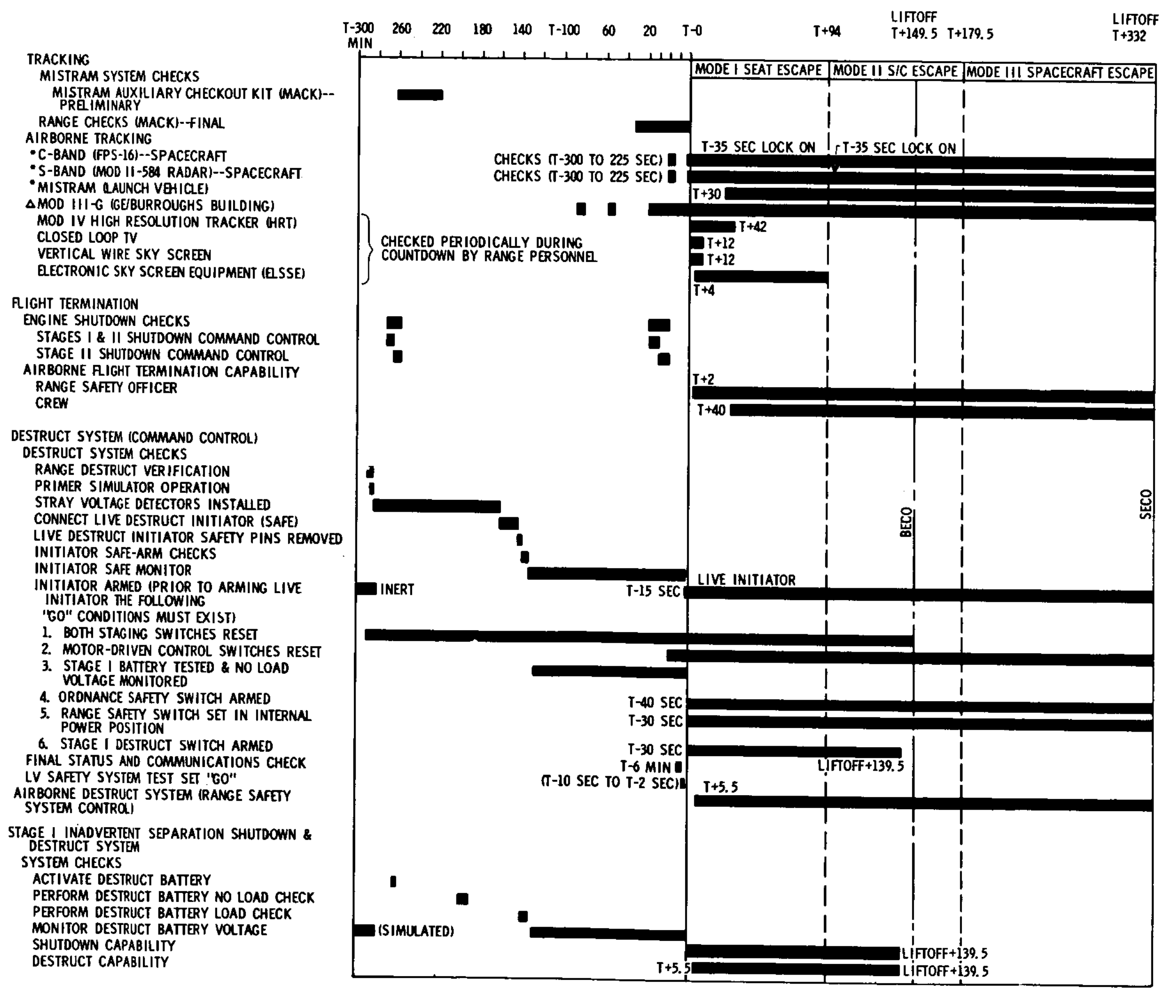
(1) Shut down the Stage I engine.

(2) Fire Stages I and II separation nuts.

(3) Start Stage II engine.

(4) Command autopilot gain changes.

The system is shown in detail in Fig. 31.



NOTE:

• PROCESSED THROUGH DATA SELECTOR BEFORE ENTERING 7090 IMPACT PREDICTOR COMPUTER (AUTOMATIC DATA SELECT PROGRAM)

Δ GE BURROUGHS DATA TO BE TIED IN WITH ADSP

FIG. 29. RANGE SAFETY SYSTEM SEQUENCE

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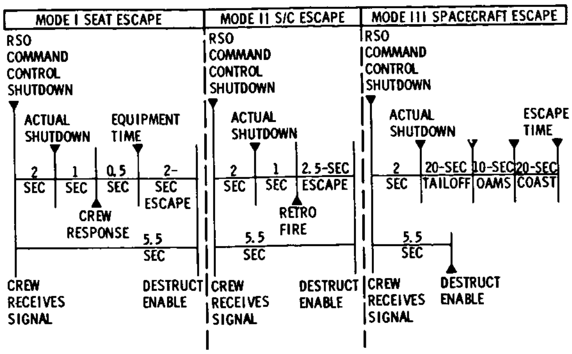

FIG. 30. FLIGHT TERMINATION SEQUENCE

Titan II Electrical Sequencing

While the Gemini and Titan sequencing systems are similar, Gemini has four additional provisions:

(1) The system is redundant.

(2) There is a Stage I fuel shutdown sensor.

(3) There are 40- or 140-second time delay relays. In Titan these arming functions are performed by the Digital Control Unit.

(4) There are two staging switches.

The APS staging switch performs the same function in both the Titan and Gemini Launch Vehicle. However, the Gemini can also call on a backup IGS switch to perform the APS functions. The degree of redundancy which has been added is summarized in Fig. 31.

The sequencing system, which is fully redundant, is set into operation when the launch vehicle actually lifts off from the pad. The following operations occur simultaneously during lift-off:

(1) The 40-second time delay relays (Nos. 1 and 2) start timing.

(2) The Three-Axis Reference System starts timing.

(3) The 140-second time delay relay starts timing.

(4) The spacecraft receives a liftoff signal.

After 40 seconds has elapsed, the 40-second time delay relays are timed out, and the astronaut then has the capability to command a launch vehicle shutdown by operating the appropriate shutdown switches. After 140 seconds has elapsed, the stage separation circuitry is armed by both the Three-Axis Reference System and the 140-second time delay relay.

Normally, at approximately 150 seconds, the oxidizer will be depleted and a low stage I engine chamber pressure will result. The Thrust Chamber Pressure Switches will sense this condition, supply a ground to the staging circuitry, and staging will occur. If the fuel is depleted before the oxidizer, the Stage I fuel shutdown sensors will supply a ground and initiate staging.

TABLE 4

FLIGHT SEQUENCING FUNCTIONS

Function	Gemini Launch Vehicle Implementation	Titan II Implementation
Program initiate	Redundant pad disconnect at liftoff. Redundant Program initiate relays Nos. 1 and 2. Relay No. 1 applies 400-cps power to Three-Axis Reference System and starts 40-second relay No. 1. Relay No. 2 starts 140-second time delay relay and 40-second time delay relay No. 2.	Signal from Master Operations Console at T-3.7 seconds starts Digital Control Unit.
Spacecraft enable for launch vehicle engine shutdown	After 40 seconds has elapsed, the crew can shut down the launch vehicle (redundant relays).	N/A
Stage I fuel and oxidizer shutdown sensing	Thrust Chamber Pressure Switch sensors and fuel shutdown sensors sense depletion of oxidizer or fuel.	Thrust Chamber Pressure Switch only.
Staging arming	Redundant staging control relays Nos. 1 and 2 are armed by the Three-Axis Reference System 139.5 seconds after liftoff, and the 140-second time delay relay arms these relays 140 seconds after liftoff.	One staging control relay is armed by the Digital Control Unit 140 seconds after liftoff.
Staging	APS staging switch (1) Stage I engine shutdown. (2) Stage II engine start. (3) Autopilot gain changes at staging. (4) Fire separation nuts on the Stage II side. IPS staging switch (1) Stage I engine shutdown. (2) Stage II engine start. (3) Autopilot gain changes. (4) Fire separation nuts on Stage I side.	APS staging switch (1) Stage I engine shutdown. (2) Stage II engine start. (3) Autopilot gain change. (4) Fire separation nuts Stages I and II
Stage II low level shutdown	Fuel and oxidizer depletion is sensed by Stage II shutdown sensors. These units are armed by the Stage II low level shutdown control relay, and the relay in turn, is armed by the Three-Axis Reference System at 322.56 seconds after liftoff.	N/A
Stage II guidance shutdown	Shutdown is accomplished by Radio or Inertial Guidance System. Switchover is accomplished by relay No. 2. The output is fed to redundant Stage II shutdown relays Nos. 1 and 2.	Shutdown is accomplished by the Inertial Guidance System through one guidance shutdown relay.

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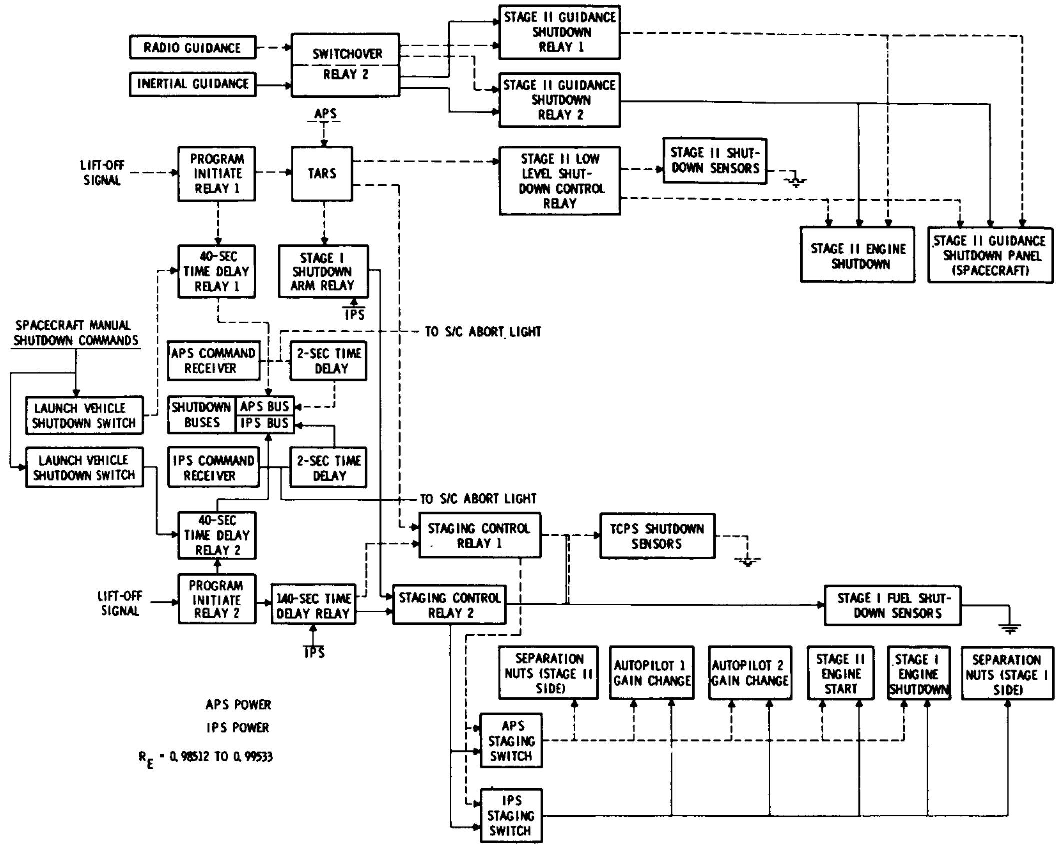

FIG. 31. ELECTRICAL SEQUENCING SYSTEM

TABLE 5

FLIGHT SEQUENCE DIFFERENCES: TITAN II AND GLV

Function	Time from Liftoff (sec) GLV	T-II	Source of Function GLV	T-II
98FS1 Stage I ignition	-3.3	-3.3	MOC	MOC
Thrust chamber switch closure	-2.2	-2.2	MOC	MOC
Fire nuts	-0.2	-0.2	MOC	MOC
Program initiate	0	0	Program initiate Relays Nos. 1 & 2	Digital control
Roll program start	+4.0	+5.0	TARS	DCU
Roll program end	+20.48	+10.0	TARS	DCU
Pitch program start Step No. 1	+23.04	+12.0	TARS	DCU
Spacecraft shutdown lockout	+40.0	N/A	40 seconds to relays Nos. 1 & 2	N/A
Pitch program, complete Step No. 1, start Step No. 2	+89.6	N/A	TARS	N/A
Flight control gain change	+104.96	+105	TARS	DCU
Start telemetry FM/FM recorder	+139.52	140	TARS	DCU
Arm staging initiate sensors	+139.52	140	TARS	DCU
	+140	N/A	140 seconds to relay	N/A
Staging	+149	+150	TCPS, shutdown sensors	TCPS, APS staging switch
(1) 87FS2 Stage I shutdown			APS & IPS staging switch	
(2) 91FS1 Stage II ignition				
(3) Flight control staging gain change				
(4) Remove power to Stage I gyros				
Pitch program complete Step 2	+156.16	N/A	TARS	N/A
Radio guidance initiate	+156.16	N/A	TARS	N/A
Arm Stage II low level sensors	+322.56	N/A	TARS	N/A
91FS2 Stage II shutdown	+332	326	RGS or IGS	IGS

NOTE:

MOC Master Operations Console
DCU Digital Control Unit
TARS Three-Axis Reference System
TCPS Thrust Chamber Pressure Switch
APS Accessory Power Supply
N/A Not applicable

[redacted]

[redacted]

Stage II shutdown is normally accomplished by the Radio Guidance System command; however, it may also be accomplished by:

(1) IGS.

(2) Astronaut.

(3) APS and IPS command control receivers.

(4) Stage II propellant shutdown sensors.

Relay No. 2 switches shutdown capability from the Radio Guidance to Inertial Guidance System.

Aerospace Ground Equipment

The selection of Aerospace Ground Equipment (AGE) for the Gemini program was influenced by two major considerations: first, that the launch vehicle is a modified Titan II; second, that Launch Complex 19 will be available for this program.

A comparison of equipment selected shows that, of the 208 AGE control points, 143 involve Titan equipment used "as is," while 33 involve Titan-modified, and 32 Gemini-peculiar control points.

The Ground Instrumentation System at the launch complex consists of a telemetry ground station, data recording equipment, signal conditioning, power monitor and control, time code distribution, control console and associated patching and cabling equipment. This system provides a flexible recording system which can be used to acquire data through umbilical or transmitted telemetry links.

Checkout and Launch Control

Essentially, the checkout philosophy calls for a decentralized approach; i.e., for each major airborne system, an equivalent piece of equipment is provided to check the appropriate airborne system. Hence, the flight control system test set will check out the airborne flight control system, etc. The relationship of the various airborne systems and the checkout equipment is illustrated in Fig. 32.

Each checkout set can operate on its equivalent airborne system virtually independently of the other equipment. However, during the countdown phase, all operations performed by the checkout equipment must be coordinated by the launch control equipment. The checkout equipment will be predominantly manual, with automatic operation being used only during critical events or time periods. This philosophy assumes more importance than ever now that redundant flight controls and hydraulic components have been incorporated into the Gemini Launch Vehicle.

Launch control is obtained with the Master Operations Control System and other related equipment, including closed circuit télevision and a community time display board. The Master Operations Control System will provide time coordination during checkout of the launch vehicle and remote control of facilities such as the process water system and erector. It will also display the state of readiness of the entire complex as the various time checkpoints are reached. Lastly, through use of hold-fire and kill signals, it will provide the means of permitting or inhibiting launch at the predetermined T-O point.

Activation

Martin has been assigned the responsibility of integrating activation of Launch Complex 19 and the Gemini Launch Vehicle Support Area at AMR (Figs. 33 and 34).

Complex 19 is currently being activated, with all activities progressing as scheduled. Prima-

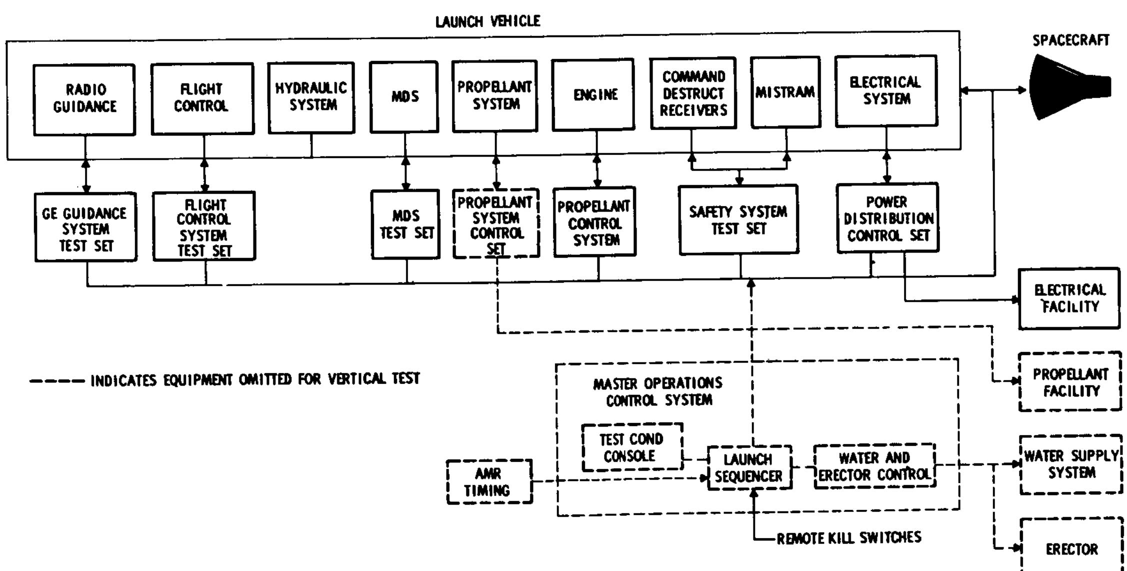

FIG. 32. AGE SYSTEMS

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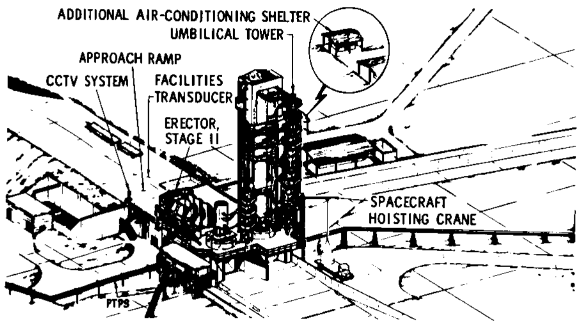

FIG. 33. AGE INSTALLATION--COMPLEX 19

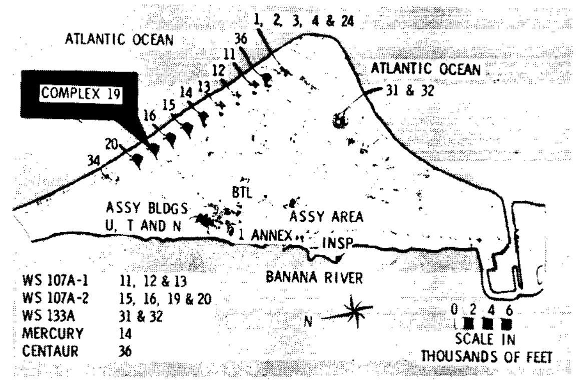

FIG. 34. CAPE CANAVERAL COMPLEX LAYOUT

rily, the activation effort on the complex consists of modifying the following existing facilities:

(1) Blockhouse: the air-conditioning system only.

(2) Ready building: double size to house NASA, McDonnell and Martin personnel.

(3) Launch deck: external north end.

(4) Complete vehicle erector: add white room, second elevator and spacecraft hoist system.

(5) Second-stage erector: relocate work platforms.

(6) Complete vehicle umbilical tower: extend height to accommodate two additional booms for spacecraft.

(7) Second-stage umbilical tower: relocate existing booms.

(8) Flume: enlarge and rearrange to permit quick runoff of expended fluids.

(9) LOX holding area: use as storage area for spacecraft AGE service carts.

(10) Roads and grading: modify south road to accommodate fuel and oxidizer holding areas.

In addition to the facilities to be modified, the following new facilities will be added to Complex 19: a new road, located at the north end running north and south for delivery of the LH_2 to the spacecraft on the pad; an oxidizer holding area; a fuel holding area; a decontamination building and an air-conditioning facility for spacecraft servicing. No new facilities are required in the launch vehicle support area, except for a components cleaning facility which is expected to be provided by AFMTC for all contractors to use.

The design of modified and new facilities has been accomplished by Rader and Associates of Miami, Florida, in accordance with Martin's "Facilities Design Criteria," ER 12053. The construction of these facilities will be accomplished by the Army Corps of Engineers. New and modified AGE will be installed in all those facilities previously mentioned. All AGE to be installed and checked out is listed in the plan.

Martin will install all AGE on Complex 19 and in the Launch Vehicle Support Area. Each agency providing such equipment for installation will check out and maintain its own equipment throughout the program.

The activation phase of the program will be considered complete immediately after the first satisfactory flight-readiness demonstration.

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