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GEMINI LAUNCH ESCAPE

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INTRODUCTION

The Gemini launch escape modes have been tailored to the design and dynamic characteristics of the launch vehicle and spacecraft. Before the pilot's role is discussed with respect to the operation of the escape system, a brief review of the spacecraft and the launch vehicle configuration will show why it has been possible and desirable to include a flight crew in the abort decision loop.

GEMINI LAUNCH SEQUENCE SYSTEM

The Gemini spacecraft incorporates a manual sequence system to control the major spacecraft sequences and system operations. This concept is in contrast to the Mercury spacecraft design where unmanned orbital flights required an automatic sequencing system involving numerous timers and interlocks.

The launch vehicle for Gemini is the standard Titan II modified to attain increased mission reliability and pilot safety. As shown in figure 1, redundant guidance, autopilot, and hydraulic control systems

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have been provided. A redundant hydraulic control system is standard on most modern aircraft. This Gemini backup system can be triggered manually or automatically.

A primary area of concern in manned launch vehicles is dynamic behavior in the event of guidance or hydraulic malfunctions in the high dynamic-pressure flight regime. Aerodynamic instability will cause the Gemini launch vehicle to diverge to breakup attitudes within 1 second when control failure occurs at maximum dynamic pressure. Because of the need for immediate switching under this condition, the backup guidance and control system is automatically triggered. Automatic switching is accomplished by abnormal rate gyro signals, full-engine gimbal position, low-hydraulic pressure, or loss of stage power. The attitude rate switching level for the first stage is 4 degrees per second in pitch and yaw and 12 degrees per second in roll. During second-stage flight, where no aerodynamic divergence is involved, the levels are opened up to 10 degrees per second in pitch and yaw. Manual switchover will be initiated if guidance or control malfunctions cause a slow divergence which is sensed by the pilot or by ground tracking. It is evident that design action has been taken to reduce the possibility of catastrophe due to guidance or control failures.

Based on a thorough failure analysis by Martin-Marietta Corp., Aerospace Corp., and the National Aeronautics and Space Administration, additional malfunction sensors are incorporated. The malfunction

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parameters which are sensed in the launch vehicle and displayed in the spacecraft are as follows:

1. Fuel-tank pressure (Stages I and II)
2. Oxidizer-tank pressure (Stages I and II)
3. Engine-chamber pressure (Stages I and II)
4. Primary-guidance or control failure
5. Excessive rates
6. Staging signal

Stage tank pressures are critical because of structural and pump suction head requirements. Figure 2 shows various pressure time histories for the Stage I fuel tank. The two top curves indicate normal excursion envelopes of gas ullage pressure. The fuel tank is pressurized by fuel-rich exhaust products from the turbo pump. The structural threshold curve starts out at ambient pressure at liftoff, increases to about 3 psi above ambient pressure at 60 seconds (maximum dynamic pressure) and increases beyond 90 seconds because of tank stresses due to increasing axial acceleration. Pump suction head limits in terms of tank ullage pressure are shown. At periods greater than 70 seconds, pump head requirements are more critical than the structural requirements. Superimposed on the plot shown in figure 2 are tank pressure histories resulting from two kinds of malfunctions: a broken autogenous pressurizing line and an ullage leak of 2 square inches. From the pilot's standpoint, the critical time for these types of failures is near liftoff when ullage volume is small and ullage pressure decreases rapidly. Tank pressure requirements as a function of time are very nonlinear. This type of parameter can best be monitored manually. If an automatic malfunction sensor were designed to follow these pressure curves, it would be nonlinear and complex.

Figure 3 is a photograph of the launch vehicle displays in the spacecraft. The tank pressures are displayed on vertical meters with two indicators per tank. These meters have two secondary uses. If the left set of needles is rapidly driven upward full scale, it is an indication that the primary guidance power system has failed. When all four of the Stage I tank pressure indicators peg out, it is an indication of physical staging. The numbers in the center of the Stage I display are time markings which indicate the structural limits of the tanks as a function of time. The crosshatched area in the lower portion of the figure indicates to the pilot the pump head pressure requirement. The small diamond-shaped symbol in the Stage II display indicates the minimum pressure for Stage II engine start.

Engine chamber-pressure lights above the tank pressure indicators are activated if either Stage I or Stage II chamber pressures drop below 65 percent. Consequently, the Stage II engine light is on during Stage I flight and is extinguished at staging. The Titan II staging sequence involves a simple fire-in-the-hole technique where the second stage is ignited before separation from Stage I and before Stage I thrust has completely decayed. This procedure eliminates the need for ullage rockets. One might rationalize that physical acceleration cues would negate the need for an engine status indication. However, partial losses in thrust are not always immediately detectable; consequently, a visual indication of chamber pressure is desirable.

The staging light is illuminated by the staging signal and is extinguished 1 second later when physical separation occurs.

The rate light is triggered at 4-degrees-per-second pitch and yaw rates from the rate gyros in the launch vehicle. The launch vehicle gyro outputs are filtered to reduce their response to the launch vehicle natural bending modes. This discrete indication of launch vehicle overrate can be cross-checked with rates which are measured in the spacecraft and displayed on analog needles superimposed on the eight-ball attitude indicator.

The guidance light illuminates when the backup guidance and control system is selected.

The digital timer located near the launch vehicle displays is very useful in correlating launch vehicle events.

Figure 4 shows the three Gemini launch escape modes. Ejection seats are used from the launch pad to an altitude of 70,000 feet. Above 70,000 feet, the spacecraft drag has reduced sufficiently to permit separation of the spacecraft by salvo fire of the retro-rockets. For this escape mode, the top section of the adapter is retained and the resulting configuration (fig. 5) is aerodynamically stable, small end forward. The adapter section is separated at apogee of the escape trajectory. After staging, when dynamic pressure is negligible, the escape mode involves shutting down the booster and separating with the translational rendezvous propulsion system.

The escape hatches and the ejection seats are triggered by the actuation of either pilot's D-ring located on the forward portion of the seat. The launch vehicle is shut down and the retro-rockets are fired by a control on the left console. The maneuver rockets are fired by a translational control handle located just above the pilot's left knee. Figure 6 shows the location of these controls in the spacecraft.

LAUNCH SIMULATION PROBLEM

A launch simulation was conducted to evaluate the launch vehicle displays and to confirm that the crew could assess the status of the launch vehicle and take abort action if required. The simulator chosen for this study was the moving base aerospace flight simulator at Ling-Temco-Vought, Inc. A photograph of the simulator is shown in figure 7.

Three degrees of angular freedom were available in the moving base cockpit with adequate displacement and washout capabilities to simulate small perturbations of the normal vehicle accelerations. In addition, a pitch rotation of ±100 degrees from the horizontal permitted a partial simulation of the direction and magnitude (up to 1 g) of axial accelerations. The cockpit motions were accomplished with hydraulic servo mechanisms driven by analog signals. The cockpit was rotated in pitch to 57 degrees from the horizontal for the launch position. At liftoff the cockpit was rotated from 57 degrees to 75 degrees, producing the sensation of lift-off thrust to the pilot. The cockpit then continued to rotate up to 90 degrees, representing the first few seconds of acceleration after launch. For abrupt changes in axial acceleration, such as staging and loss of thrust, the cockpit was rotated rapidly downward.

The combination of engine and aerodynamic noise was simulated by a high fidelity speaker system located in the astrodome surrounding the cockpit. The noise spectrum was reproduced from an actual cockpit recording obtained during a Mercury-Atlas launch. The maximum intensity level inside the closed cockpit near the pilot's head was 104 decibels which occurred at maximum dynamic pressure. Corresponding noise deviations were programed as applicable for each simulated malfunction.

The simulation program involved 51 malfunction runs representing nine major types of malfunctions which are as follows:

1. Partial loss of thrust - one engine (Stage I)
2. Total loss of thrust - one engine (Stage I)

3. Total loss of thrust - both engines (Stage I)
4. Staging failures
5. Tank (fuel and oxidizer) pressure losses
6. Roll malfunction (Stage I)
7. Direct-current power failure
8. Instrument malfunction
9. Display-light failure

The selected malfunctions were based on failure analysis data for the Titan II launch vehicle. Technical personnel from NASA and Ling-Temco-Vought, Inc., established the number of runs for each type of malfunction, and the time that the malfunctions were to begin. The selection was based on the criticality of the malfunctions with respect to anticipated pilot difficulty in detecting and evaluating the cues and the response time for taking corrective action. Normal launch vehicle runs were interspersed throughout the simulation. The most difficult malfunction runs were selected for use in the simulation regardless of their probability of occurrence in actual flight.

The NASA astronauts who participated in the simulation were given only 1 day of indoctrination. Each pilot was scheduled for approximately 75 runs, 65 runs having malfunctions and 10 being normal. Each of the 51 malfunction runs was presented to the pilots at least once, and the 14 most difficult runs were presented twice to each pilot. The runs were randomly distributed so that the pilots had no way of knowing which problem would be presented next. They were aware of the general nature of the possible malfunctions, but they were not aware of the time during flight at which the malfunctions were programed.

A digital computer controlled simulator attitudes, vibration, and the magnetic tapes. With the exception of pilot action which completed the runs, the simulator was operated in an open loop configuration. In addition to the launch-vehicle-related displays, the cockpit contained a D-ring ejection seat handle, launch vehicle shutdown and spacecraft abort handle, and a secondary guidance switch. The pilot's control response to each run was recorded, as was his verbal assessment of each run.

It became readily apparent to the pilots that the most critical malfunctions were engine failures or tank pressure losses immediately after liftoff or immediately after staging. The critical engine failures were readily detectable through redundant cues, including decrease in sound level, decrease in acceleration, and illumination of the combustion chamber pressure light. Pilot reaction time to this failure was as low as four-tenths of a second. Reaction time requirements varied from approximately 1 second to $2\frac{1}{2}$ minutes, depending upon the type and time of malfunction. Several of the malfunctions, such as sensor failures and gradual tank pressure losses were noncritical and required no abort action. For the majority of the failure modes, there were multiple cues such as is the case with engine failure. Tank pressure losses were sensed by redundant transducers driven by redundant power sources and presented on redundant meters. For tank pressure failures which occurred after the first 5 seconds, the rate of decay was relatively slow. The pilots were able to let the pressure graze the structural limit or were able to wait until the pressure dropped to within 1 psi of the structural margin before taking abort action.

With only 1 day of familiarization and with partially developed displays, the pilots were able to analyze and react correctly to the critical malfunctions. It became clear during this simulation that the pilot's presence in the abort control loop provides the potential to save missions which would probably be aborted by an automatic system. Extreme monitoring accuracy should be possible after instrument development has been completed and the pilots have received the intensive familiarization and training which will precede a manned Gemini flight.

A manual abort system will provide added operational flexibility by enabling the flight crew to choose an abort time which may reduce the possibility of aborts at high dynamic pressure; to choose optimum abort times compatible with contingency recovery areas; and to reduce the probability and the risk involved with an inadvertent abort.

There is good analogy here to aircraft operations where the pilot by using his flight instruments, engine displays, and physical cues is able to assess accurately the validity and seriousness of various warning or malfunction indications.

SUMMARY

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In summary, Gemini mission reliability and crew safety have been enhanced by incorporating a redundant guidance and control system and a manual launch vehicle monitoring system.

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FLIGHT CONTROL AND GUIDANCE SYSTEM

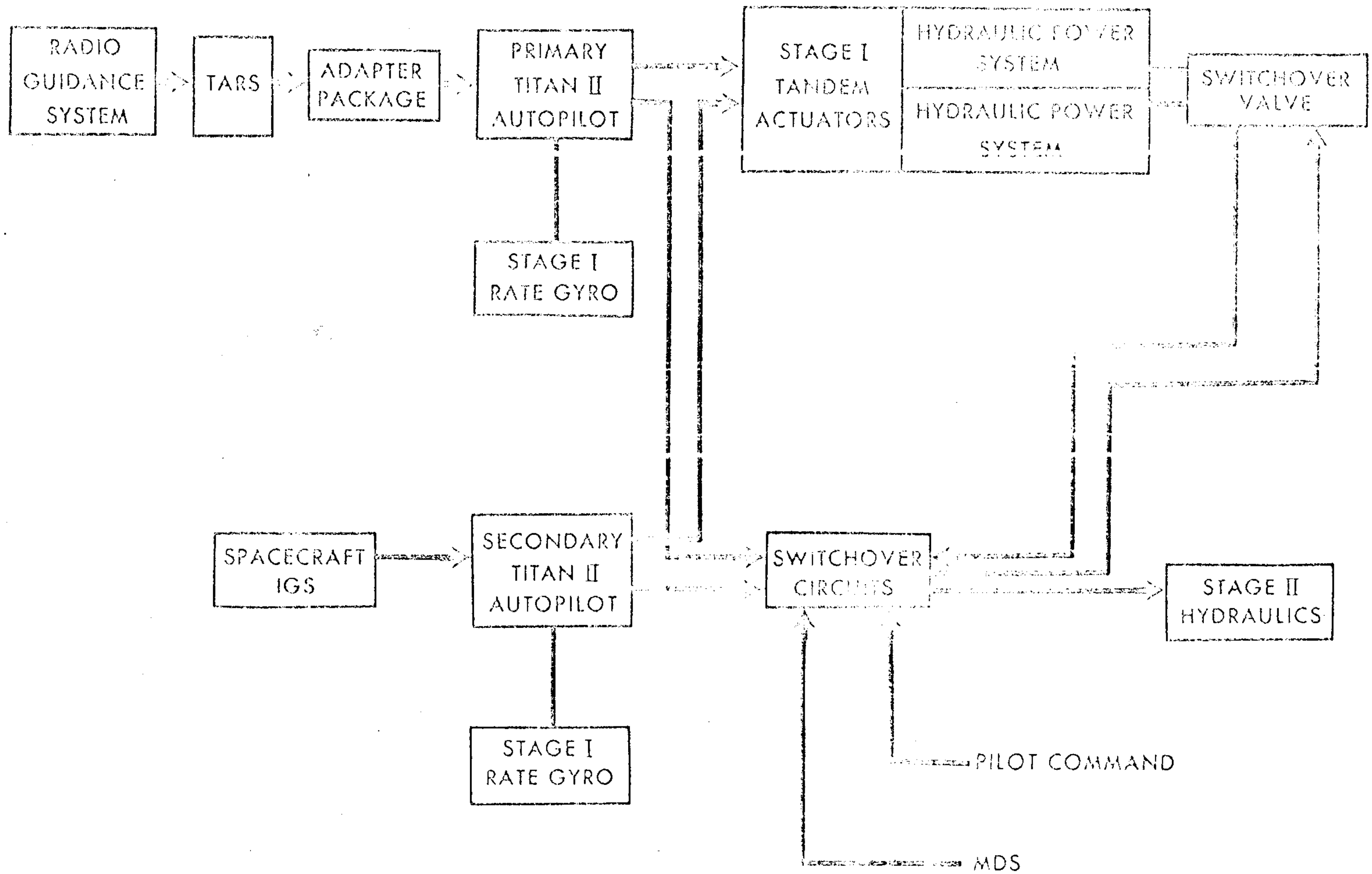

Figure 1

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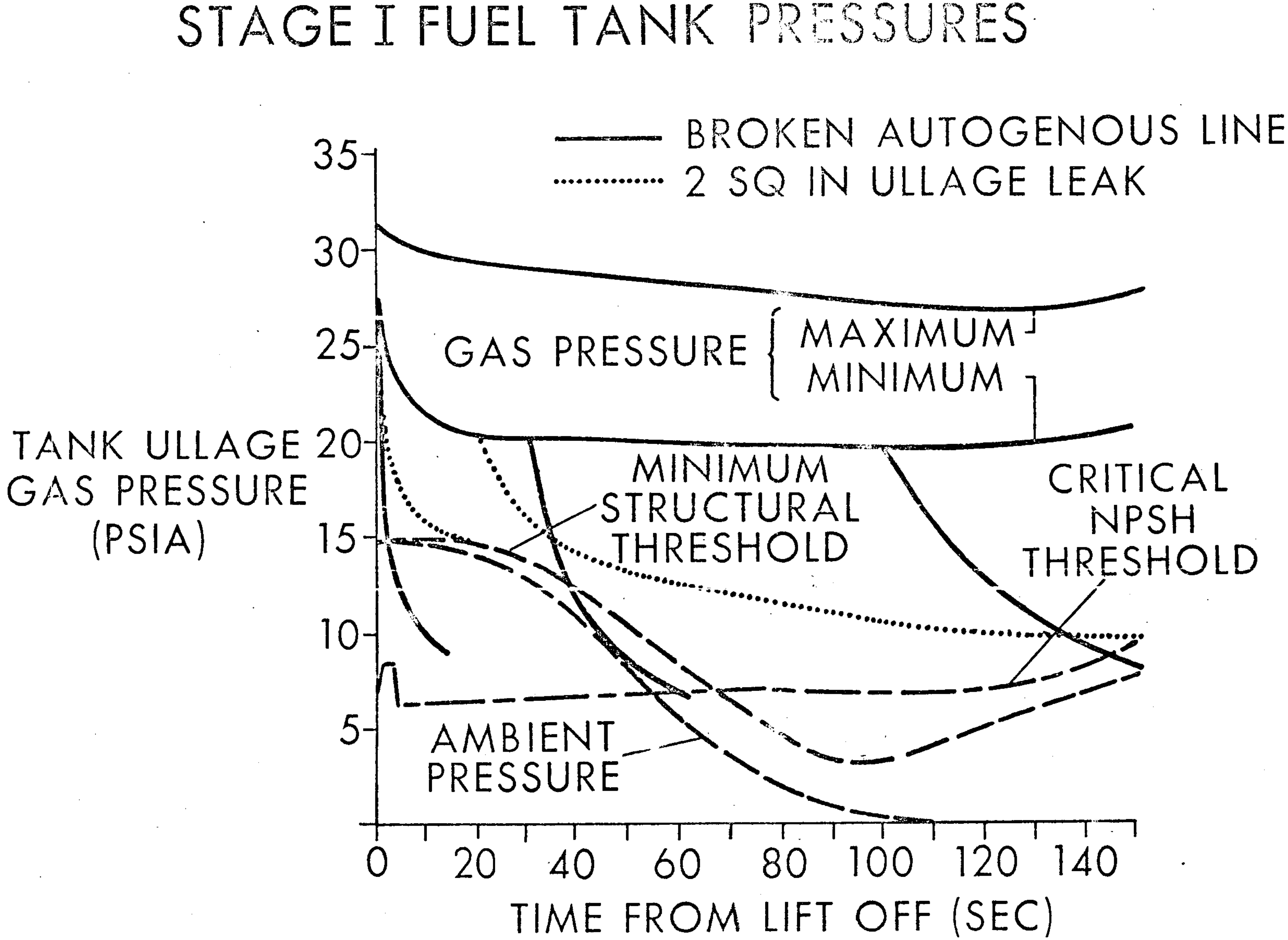

Figure 2

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BOOSTER MONITORING DISPLAYS

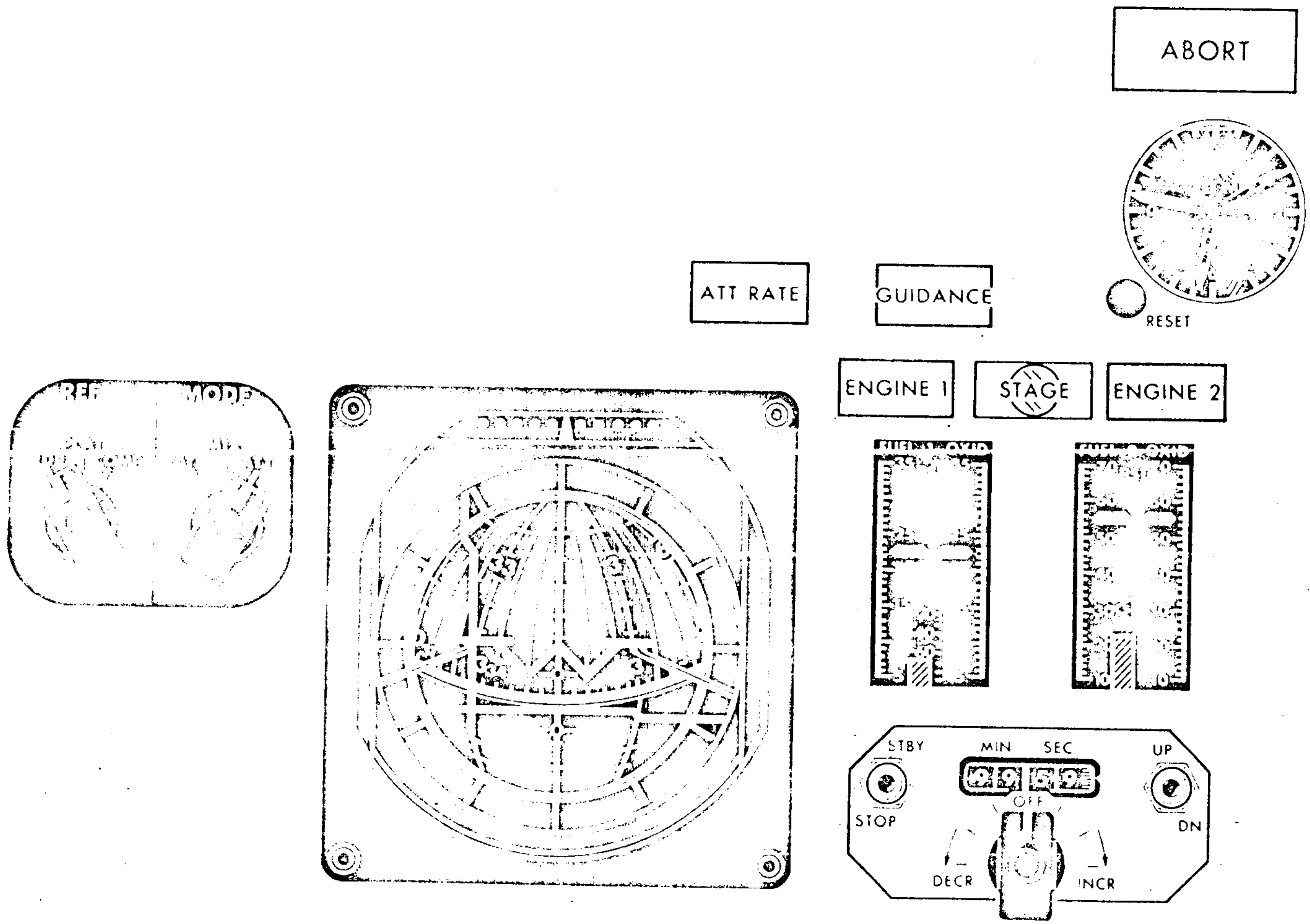

Figure 3.

GEMINI LAUNCH PARAMETERS

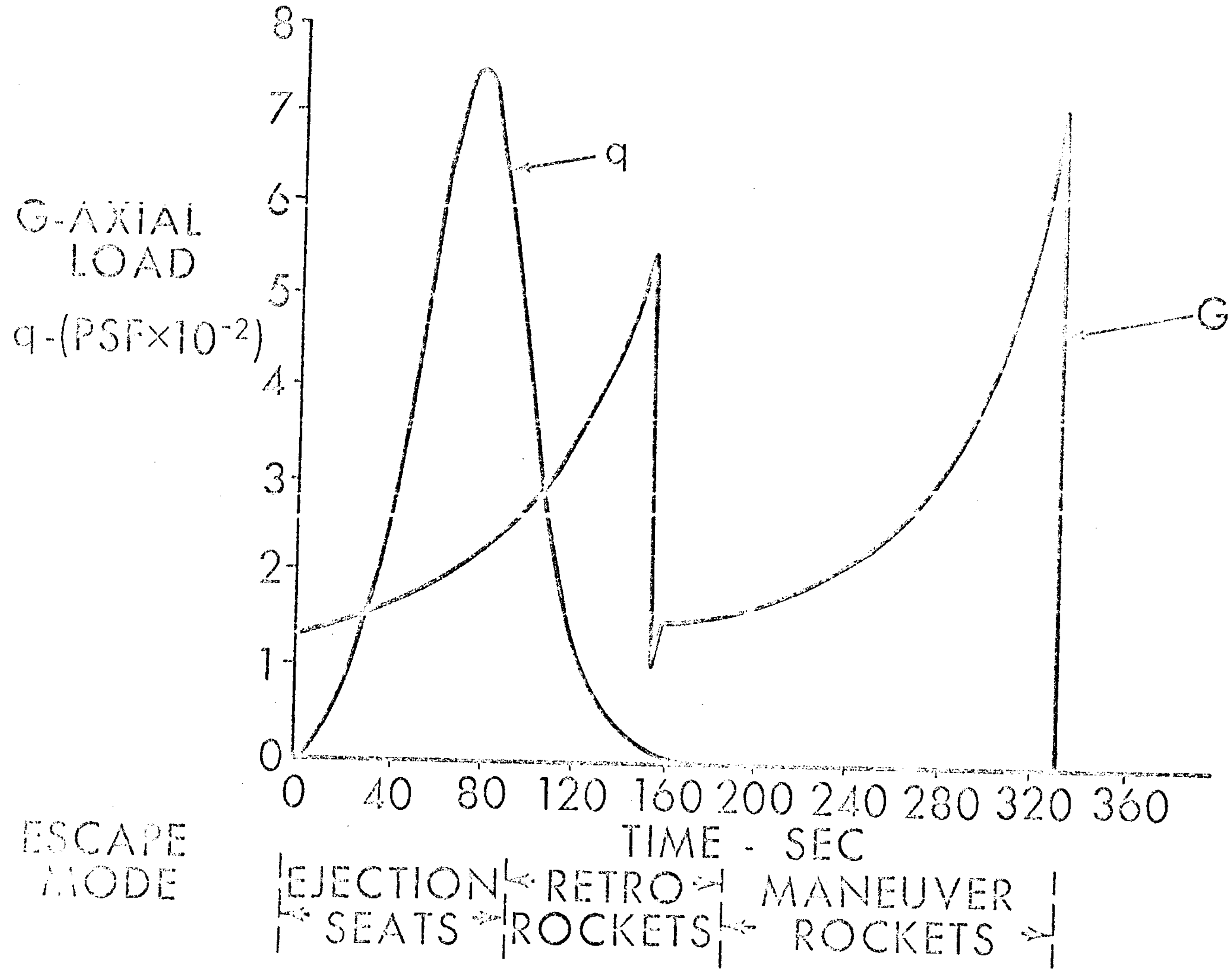

figure 4

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GEMINI RETRO ESCAPE CONFIGURATION

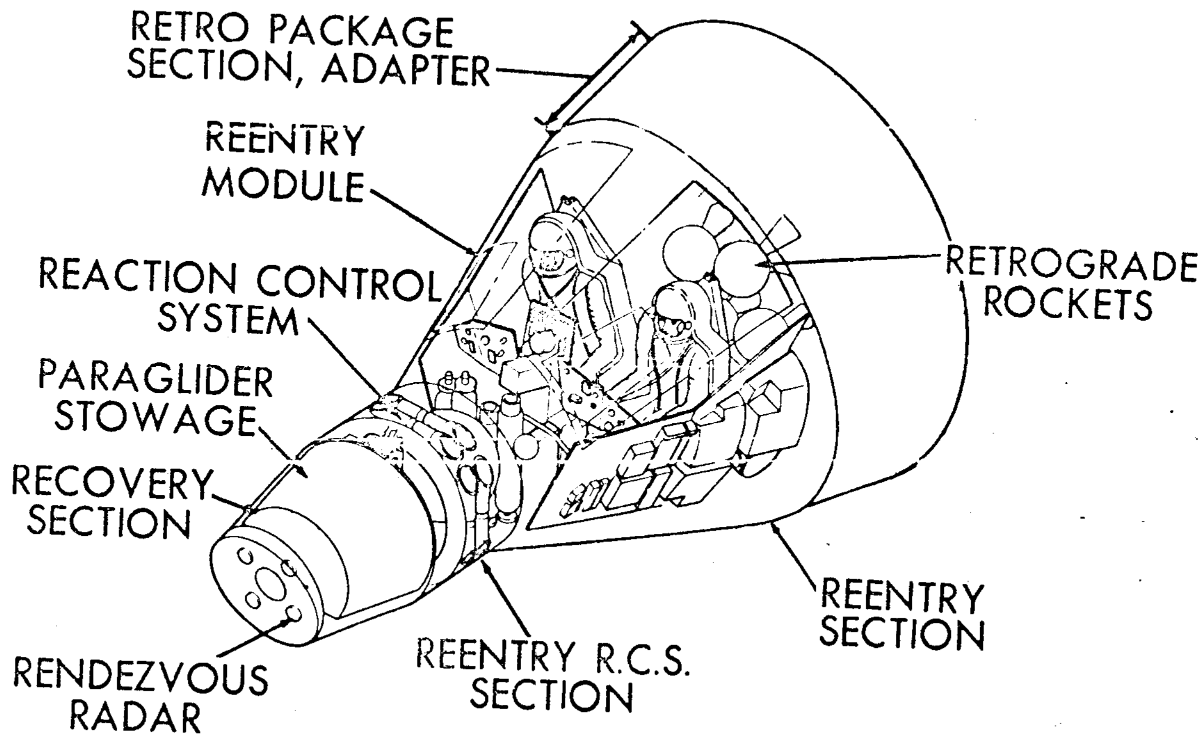

Figure 5

ESCAPE CONTROLS

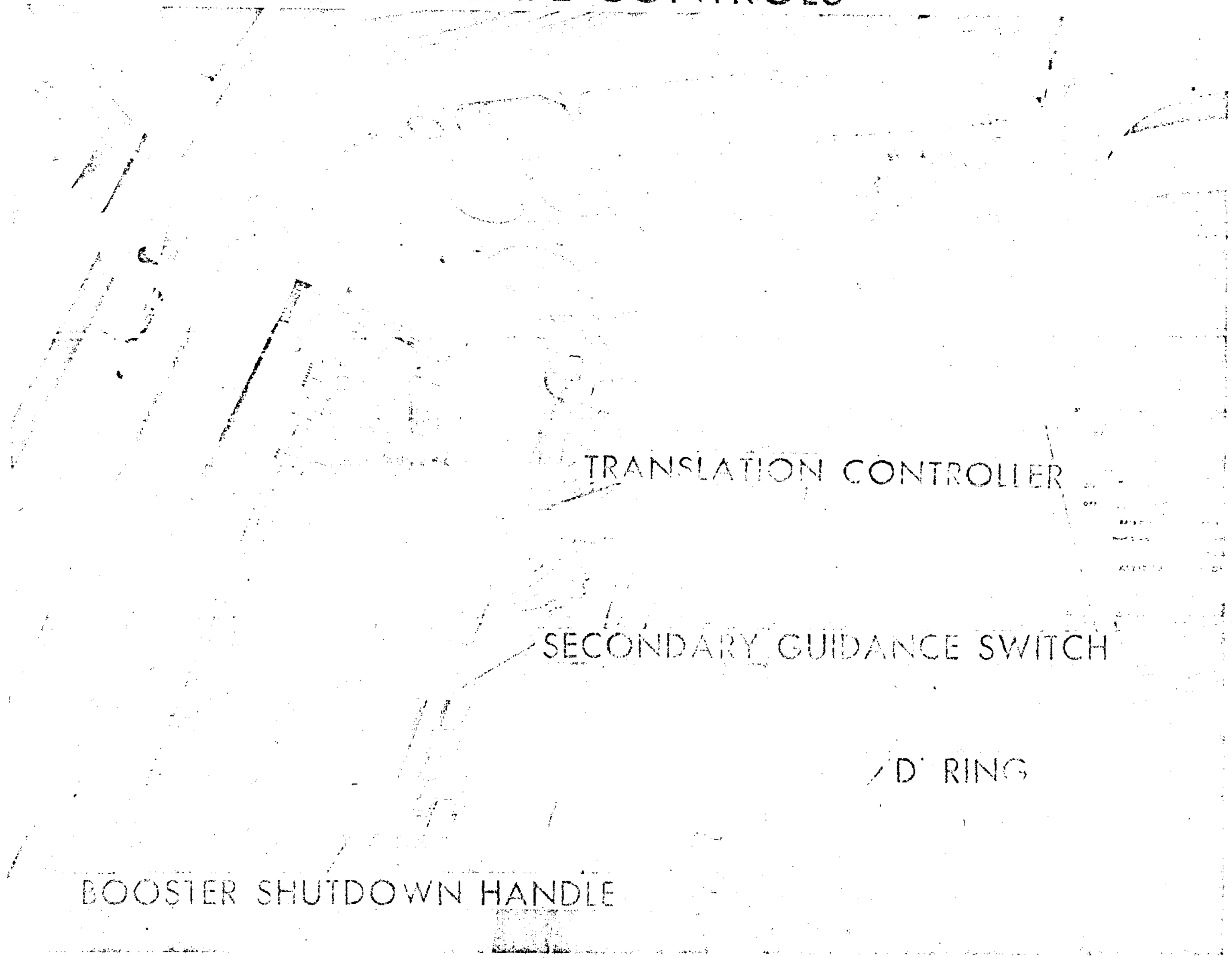

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Figure 6

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GENERAL VIEW OF MANNED AEROSPACE FLIGHT SIMULATOR

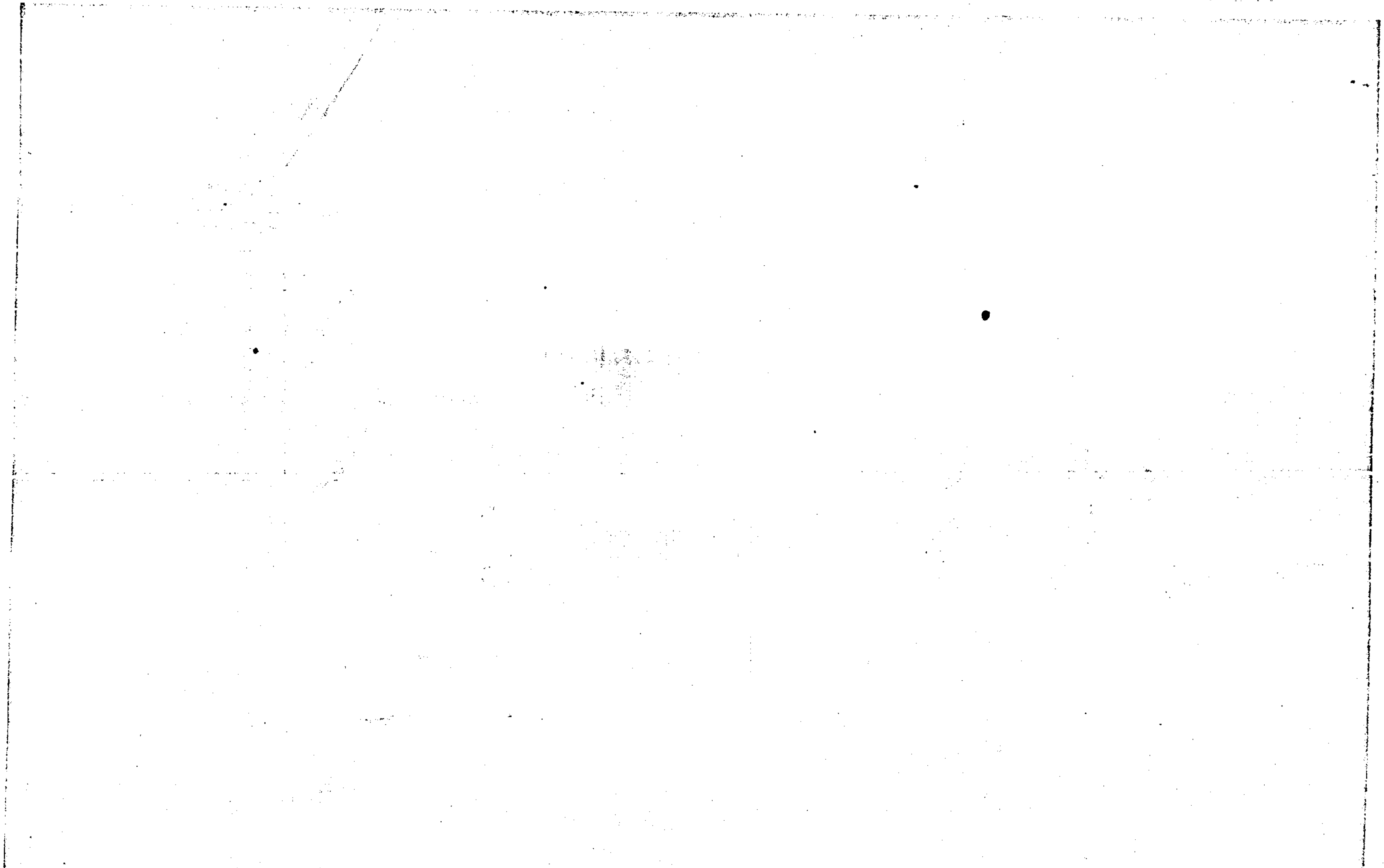

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Figure 7