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SATURN V MANUAL BACKUP GUIDANCE AND CONTROL PILOTED SIMULATION STUDY

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Ames Research Center
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SUMMARY

Fixed base simulation studies of manual backup guidance and control for the Saturn V have been conducted. The aim of the studies was to investigate systems which could be used, with minor hardware changes in the Apollo/Saturn system, to provide guidance and control in the event of a failure of the primary attitude control system in the launch vehicle. A manual attitude trim control system was devised using the spacecraft inertial platform and command module computer to provide a backup attitude control loop. The guidance loop was provided by the pilot who observed digital displays of trajectory parameters, compared them with nominal values, and input controller commands to bias vehicle attitude, so as to follow the nominal flight trajectory. Another backup system called the rate command system fed the pilot controller commands directly to the launch vehicle where they were summed with the launch vehicle rate gyro output. In both systems, an on-off mode of the pilot controller system was used, similar to that used for vehicle attitude control while in earth orbit.

The two backup systems were evaluated first for upper stage guidance and control, from second stage ignition to earth orbit injection. The results indicate that either system could be used effectively to guide the vehicle into earth orbit. Evaluation of these two systems for the first stage flight showed that the rate command system was not suitable because the direct on-off signal from the controller adversely affected the sloshing and bending dynamics of the vehicle. The attitude trim system, however, was suitable for first stage control.

A brief study was made of vehicle backup control during first stage burn with actuator hardover or thrust failures. A comparison of the data for the attitude trim system with and without the pilot's inputs was made. With manual trim, considerable reduction in trajectory dispersions at staging was obtained. Average lateral velocity and position at first stage burnout were reduced by a factor of two or better for two actuator hardover type failures, and by about one order of magnitude for single actuator hardover failures. Only slight reductions in bending moment were obtained with manual trim.

INTRODUCTION

Studies at Ames Research Center have shown the feasibility of using the pilot to guide and control a large flexible launch vehicle such as the Saturn V (refs. 1 and 2). The pilot's contribution to the reliability of the booster control system has also been investigated (ref. 3). These studies assumed some flexibility in control system design such as the use of a proportional type hand controller and filtering of the controller output to reduce body bending excitation. The results of these studies prompted a request by Manned Spacecraft Center to study the possibility of providing a manual backup guidance and control system in the Saturn V. However, major constraints were placed on this application study. The constraints were: (1) any manual-backup control system would entail no hardware changes to the existing Saturn V control system, and (2) the backup system would result in only minimal changes to software in the Apollo command module computer.

Subject to these constraints, two control systems were proposed: a rate command system and an attitude trim system. The simulation study of these control systems was divided into two phases: the atmospheric flight phase (S-IC, first stage), and the flight phase outside the sensible atmosphere (S-II and S-IVB, upper stages). The investigation included: controller gain variations, performance for nominal flight with wind disturbance, earth orbit injection performance, and a brief look at the contribution of manual backup control to mission reliability for various failure modes during first-stage flight. The reliability analysis techniques used for studying the failure modes are discussed in reference 4.

MANUAL BACKUP GUIDANCE AND CONTROL SYSTEMS

The proposed manual backup guidance and control systems would use the existing Apollo/Saturn V hardware represented in figure 1. The components below the dashed line represent system elements in the launch vehicle, and those above the dashed line represent system elements in the spacecraft. The primary guidance and control system contained in the launch vehicle consists of an inertial platform, a guidance computer, and a data adapter. The guidance computer uses an iterative guidance scheme to calculate an attitude command which the vehicle should follow for a trajectory with nearly optimum use of the propellant. This calculated command attitude is compared to the measured attitude from the inertial platform, and an attitude error signal ($\Delta\varphi$) is generated if an error exists. This signal is fed through the "L/V GUID" (launch vehicle guidance) switch to the control computer. The launch vehicle is rate stabilized by feedback from the rate gyros into the control computer. The control computer processes these signals and produces an engine-actuator angle command (β_c) for thrust vector control.

The L/V GUID switch is a hard wire interface between the launch vehicle and the spacecraft which allows control inputs from the spacecraft. The original purpose of this interface was to permit spacecraft control of the launch vehicle attitude while in earth orbit. It appears feasible to use this same input system to provide manual backup attitude control in the event of a failure in the launch vehicle platform, computer, or data adapter. The spacecraft components which could be used for such a backup system are represented in the top of figure 1. The basic elements used are the spacecraft inertial platform sensors. The resolvers in the inertial platform provide measured attitude information for display. Attitude data plus velocity data from the

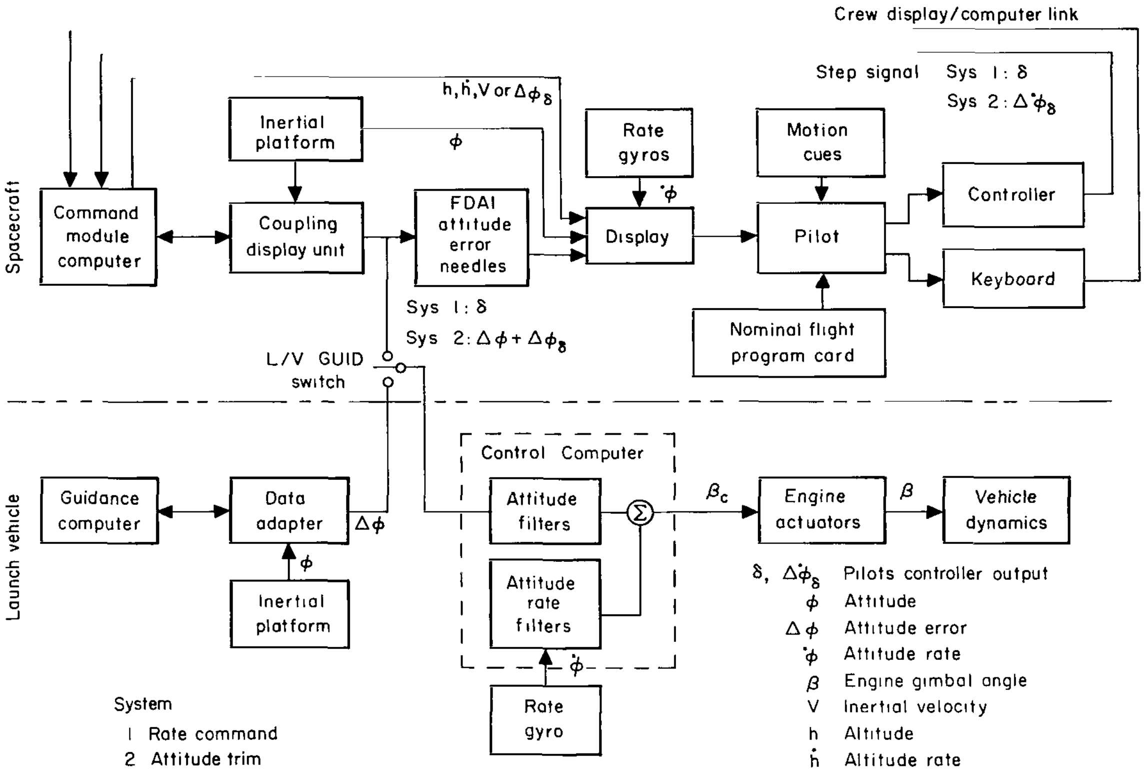


Figure 1.- Manual backup guidance and control systems.

integrating accelerometers on the inertial platform are sent to the command module computer through the coupling display unit. The pilot can communicate with the computer through the keyboard. For this study, it was assumed that he had used the keyboard to call up velocity, altitude, and altitude rate for display on digital readouts in the display panel. The pilot's side-arm controller signal is also fed to the command module computer. The controller output mode was restricted to an on-off or step type (as opposed to a proportional type) for this study. This is the mode used for attitude control of the launch vehicle while in earth orbit.

The two manual backup systems mentioned earlier, the "rate command system" and the "attitude trim system," are related to the manner in which the controller signal and command module computer are used. The simplest method is to send this step type signal from the hand controller directly through the command module computer and coupling display unit without any processing. This step signal is then fed directly to the control computer when the L/V GUID switch is in the S/C (spacecraft) position. This signal bias is nulled by the rate gyro feedback signal when a compensating launch vehicle attitude rate is established, hence the name, rate command system. The pilot is the sole source for attitude feedback control signals in this backup system. His reference value information is obtained from a reference flight program card containing attitude, altitude, altitude rate, and velocity as a function of time. He generates his commands by comparing the tabulated reference trajectory data with the measured data displayed on the cockpit instrument panel.

The attitude trim system is slightly more complex, and would require more extensive changes in the spacecraft computer. For this system, a nominal flight, pitch attitude program is stored in polynomial form in the spacecraft computer. This polynomial is used to compute the nominal attitude which is then compared with the measured value to give an attitude error signal ($\Delta\varphi$) that is fed to the launch vehicle control computer. Thus, we have a backup autopilot attitude loop for the launch vehicle rate-stabilized control system. The pilot can provide backup trajectory guidance by monitoring the actual trajectory values, comparing them with reference flight program card values, and providing an attitude trim if needed. The step signal from his controller is treated as a trim rate ($\Delta\dot{\varphi}_\delta$). This trim rate is integrated in the computer and summed with the attitude error signal ($\Delta\varphi + \Delta\varphi_\delta$).

In all the systems described above, the control signal is fed through an attitude error filter in the control computer. The control computer also contains an attitude rate filter to shape the rate gyro signal. The shaping filters were designed to stabilize the launch vehicle with respect to bending modes. For this study, the low-frequency portion of a typical set of Saturn V shaping filters was used. The filters for the pitch and yaw axes control signals were as follows:

Upper stages

Attitude signal

$$\frac{\varphi_o(s)}{\varphi_i(s)} = \frac{6.5}{s + 6.5} \qquad \text{(S-II \& S-IVB)}$$

Attitude rate signal

$$\frac{\dot{\varphi}_o(s)}{\dot{\varphi}_i(s)} = \frac{(7.5)(12.0)}{(s + 7.5)(s + 12.0)} \qquad \text{(S-II)}$$

$$\frac{\dot{\varphi}_o(s)}{\dot{\varphi}_i(s)} = 1 \qquad \text{(S-IVB)}$$

First stage

Attitude signal

$$\frac{\varphi_o(s)}{\varphi_i(s)} = \frac{(0.0378)(31.7)(s + 0.0951)}{(0.0951)(s + 0.0378)(s + 31.7)}$$

Attitude rate signal

$$\frac{\dot{\varphi}_o(s)}{\dot{\varphi}_i(s)} = \frac{(2.95)(4.60)}{(s + 2.95)(s + 4.60)}$$

The gains and switching times used with the control system filters in the feedback loops were as follows:

Stage	S-IC		S-II			S-IVB
Time, sec	0-100	100-150	150-210	210-340	340-526	First burn
Attitude gain	0.82	0.45	1.12	0.65	0.44	0.81
Attitude rate gain, sec	0.66	0.44	1.89	1.10	0.74	0.97

Since this study considered only rigid body effects about the roll axis, roll control signals were not filtered. Constant values of attitude and attitude rate gains were 0.17 and 0.11, respectively.

DESCRIPTION OF SIMULATION

Simulator Hardware

A previous feasibility study of manual control of a flexible booster (ref. 1) showed that pilot motion cues due to bending should not present a problem for accelerations on the order of 0.1 g. Therefore, a fixed cab simulator was used for this study. Figure 2 is a photograph of the simulator

Figure 2.- Photograph of simulator including computers and fixed cab.

arrangement. The vehicle and systems equations of motion were programmed on four analog computers having a total of approximately 400 amplifiers. The fixed cab is shown on the right. The pilot's display panel and seat are shown inside the cab. Not shown is the pilot's side-arm controller which was mounted on the right arm of the seat. An Apollo Block II, three-axis rotation hand controller was used.

A close-up of the display panel is shown in figure 3. It contains an Apollo FDAI (Flight Director Attitude Indicator), a sweep-hand clock to the

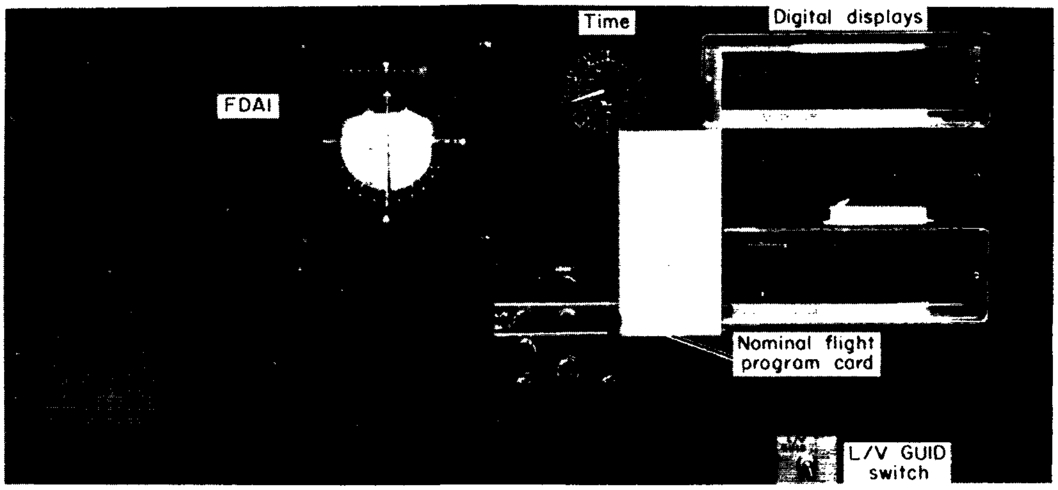
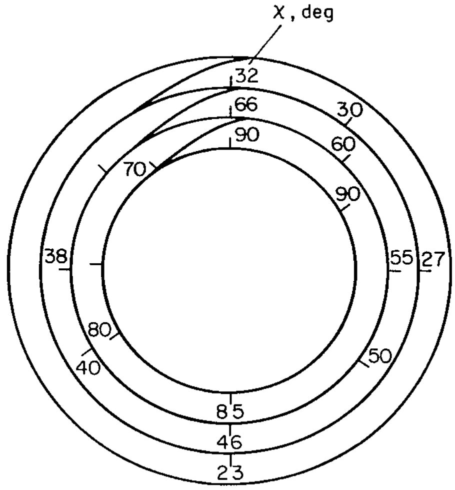


Figure 3.- Display panel.

right of the FDAI, and three digital display units. The FDAI contains the three-axis ball attitude display, three rate meters around the periphery of the ball, and three error needles or flight director needles across the ball face. The three digital readouts were used to display velocity in feet per second, altitude in nautical miles, and altitude rate in feet per second. The digital display units were updated every 2 seconds. To obtain sensitive scaling, the digital display of velocity was not active in this simulation study until S-II/S-IVB staging at the nominal initial velocity of 22,730 feet per second. For the attitude trim system, the trim value in degrees was available for display in place of velocity, when desired. A card on the left of the digital displays defines the reference flight trajectory as a function of time. Warning lights are to the right and below the FDAI. The pertinent warning lights used in this study were the lower five lights which indicate loss of thrust for a particular engine, and the light on the upper right which indicates a launch vehicle inertial platform malfunction (L/V GUID light). The simulated L/V GUID switch is shown in the lower right of the picture.

Figure 4 shows the two nominal flight program cards which were used. For first stage studies, a circular card placed around the clock showed the reference pitch attitude in degrees at various clock sweep-hand positions for the first 150 seconds of flight. The reference trajectory program card used for the upper stages study listed the trajectory parameters; altitude (h), altitude rate ($\dot{h}$), velocity (V), and nominal attitude (X), at 30-second intervals of time (t). Nominal S-II/S-IVB staging occurs at 8.96 minutes. The nominal orbit injection conditions (11.125 min) used in this study were: altitude = 108.8 nautical miles, altitude rate = 6 feet per second, velocity = 25,560 feet per second, and vehicle attitude = -23.0°.



First stage

t, min	h, nm	$\dot{h}$, fps	V, fps	X, deg
2.7	35.7	3038		22.7
3.0	45.0	2702		24.0
3.5	57.7	2276		24.0
4.0	68.6	1885		24.0
4.5	77.9	1568		22.1
5.0	85.3	1268		19.8
5.5	91.0	1000		16.8
6.0	96.2	755		13.9
6.5	99.4	567		10.8
7.0	101.7	416		7.7
7.5	103.7	293		4.4
8.0	105.1	215		0.0
8.5	106.3	182		- 3.0
8.96	107.0	200	22730	- 6.0
9.0	107.2	197	22743	- 6.1
9.5	107.8	141	23405	-10.7
10.0	108.5	75	24065	-15.0
10.5	108.8	35	24741	-18.5
11.0	108.8	10	25400	-22.0
11.125	108.8	6	25560	-23.0

Upper stages

Figure 4.- Nominal flight program for upper stages.

Equations

The equations of motion used to simulate the upper stages included six-degrees-of-freedom rigid-body motions, first mode bending, liquid oxygen slosh, and actuator/engine gimbal motions. The control system filters described previously were also included.

The equations of motion used to simulate the first stage included: six-degrees-of-freedom rigid-body motions, first- and second-mode bending, S-IC fuel and oxygen slosh, and S-II oxygen slosh modes, and actuator/engine gimbal motion. The first-stage control system filters were also simulated. A wind disturbance profile was included.

Performance Criteria and Initial Conditions

Accuracy of conditions at earth-orbit injection was used as the performance criterion for the upper stage portion of the study. The following values were provided by Manned Spacecraft Center as guidelines for injection performance.

$$\Delta h = 1 \text{ n.mi.}$$

$$\Delta \dot{h} = 45 \text{ fps}$$

$$\Delta V = 10 \text{ fps}$$

The altitude rate criterion ($\Delta\dot{h}$) is equivalent to a flight path angle accurate to 0.1° at injection.

The pitch plane initial conditions at S-II ignition, used in the upper stages study, were based on digital computer simulation data from Marshall Space Flight Center and took into account thrust, wind disturbances, and other first-stage burn perturbations. Values chosen for the worst case were 1.1 nautical miles altitude error, and 153 feet per second altitude rate error. These are 3σ values. Two error orientations were used in this phase of the simulation study:

(a) Both negative

$$\Delta h_{ic} = -1.1 \text{ n.mi.}$$

$$\Delta \dot{h}_{ic} = -153 \text{ fps}$$

(b) Both positive

$$\Delta h_{ic} = 1.1 \text{ n.mi.}$$

$$\Delta \dot{h}_{ic} = 153 \text{ fps}$$

In addition, some runs were made with 1.5, 6, and 12σ magnitudes. Yaw plane initial conditions were assumed to be zero for this study. Any actual yaw dispersions at S-II ignition could presumably be nulled either by ground voice command to the pilot or by the pilot using a presentation of yaw displacement error on one of the digital displays.

The primary performance criterion for first-stage studies was the ratio of maximum vehicle bending moment to breakup bending moment. This will be discussed later.

Pilot Background

Two Ames research pilots participated in both phases of this study. They have extensive background in both actual and simulator research flight programs. The third pilot is a former research pilot for Lewis Research Center who is now involved with astronaut crew safety and training at Manned Spacecraft Center.

PROCEDURES AND RESULTS

The remainder of the paper will cover the procedures used and the results from the two phases of the study.

Upper-Stage Guidance and Control

Rate command system.- The pilot's task was to hold attitude to zero in roll and yaw, and to null the initial trajectory dispersions in the pitch plane by flying the vehicle to match the reference trajectory tabulated on the program card. The proper level of the controller gain for this task was determined by the pilots in several simulated flights using different levels of gain. The data are shown in figure 5 for two pilots. The earth orbit

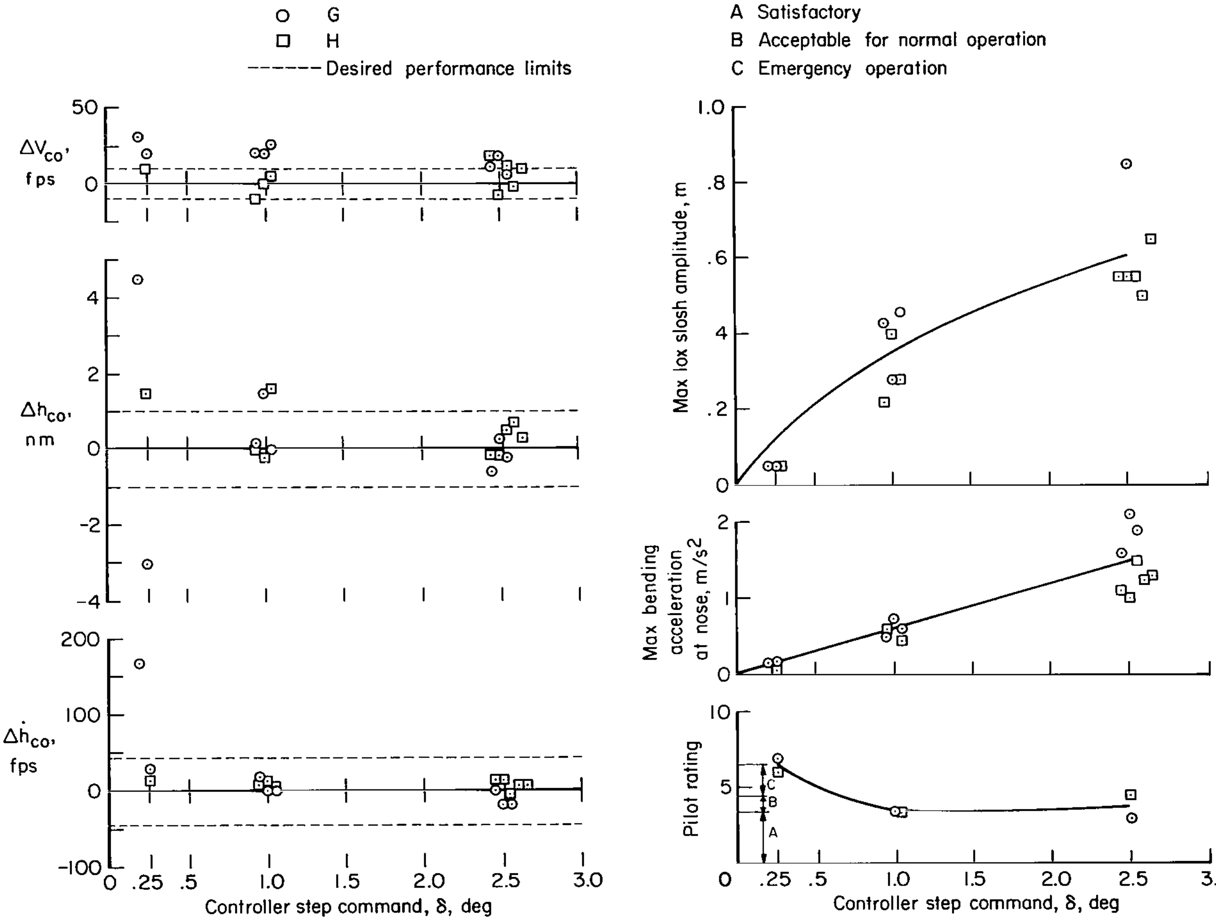


Figure 5.- Effect of controller gain on injection performance, sloshing, bending, and pilot rating for upper stage flights using the rate command manual backup system.

injection error parameters are shown on the left of the figure as a function of controller step command (δ). The injection error parameters are: velocity error at cutoff (ΔV_{co}), altitude error at cutoff (Δh_{co}), and altitude rate error at cutoff ($\Delta \dot{h}_{co}$). The dashed lines indicate the performance guidelines. The curves on the right show maximum values of liquid oxygen slosh, bending acceleration at the nose of the vehicle, and pilot opinion rating as a function of controller step command level. Controller step levels of 0.25°, 1.0°, and 2.5° were investigated. The corresponding launch vehicle rates resulting from these controller levels during the S-II stage burn were 0.15, 0.59, and 1.50 degrees per second, respectively. The resulting rates during the S-IVB stage burn were 0.21, 0.83, and 2.10 degrees per second, respectively.

Little difference in injection performance is evident for the 1.0° and 2.5° controller step command levels (fig. 5). The 0.25° level did not provide enough control power to make the necessary trajectory corrections; consequently, large errors in altitude and altitude rate occurred at injection. The pilots rated this level as unsatisfactory. Slosh and body-bending excitation increased considerably with controller step level. A controller step of 1.0° was chosen as a compromise between flexible body and fuel sloshing excitation on the one hand and adequate control authority on the other.

A series of simulated flights were then made by each of three pilots using the 1.0° step input controller level. The injection error data are shown in the appendix (fig. 10). The initial conditions were varied during the simulated flights. The altitude and altitude rate data lie well within the performance guidelines except for two cases. Mean error and standard deviation of altitude and altitude rate for the 3σ initial condition flights for all pilots were:

	Mean error	Standard deviation
Altitude	0.15 n.mi.	0.54 n.mi.
Altitude rate	5.0 fps	8.9 fps

Two variations in flight conditions were investigated by pilot H. The first variation occurred when an L/V inertial platform failure was simulated at 210 seconds (60 seconds after S-IC/S-II staging), with a corresponding activation of the L/V GUID light on the warning light panel. This light indicates that the launch vehicle inertial platform is not functioning correctly. The pilot's procedure was to move the L/V GUID switch from L/V (launch vehicle) to S/C (spacecraft) and manually control the launch vehicle. The other variation occurred during a simulated 3 percent increase in thrust. Neither of these variations significantly affected altitude or altitude rate errors at injection. The effect of initial dispersion magnitude variations on performance, along with velocity errors at injection will be discussed later.

Figure 6 shows a typical time record of pitch attitude, altitude, and their associated rates for a simulated flight by pilot G, with -3σ initial dispersions. The altitude record indicates that the guidance task has a long period of about 100 seconds with maximum rates less than 50 meters per second. Pitch rates were less than 0.02 radian per second (approximately 1.2 degrees per second) throughout the flight.

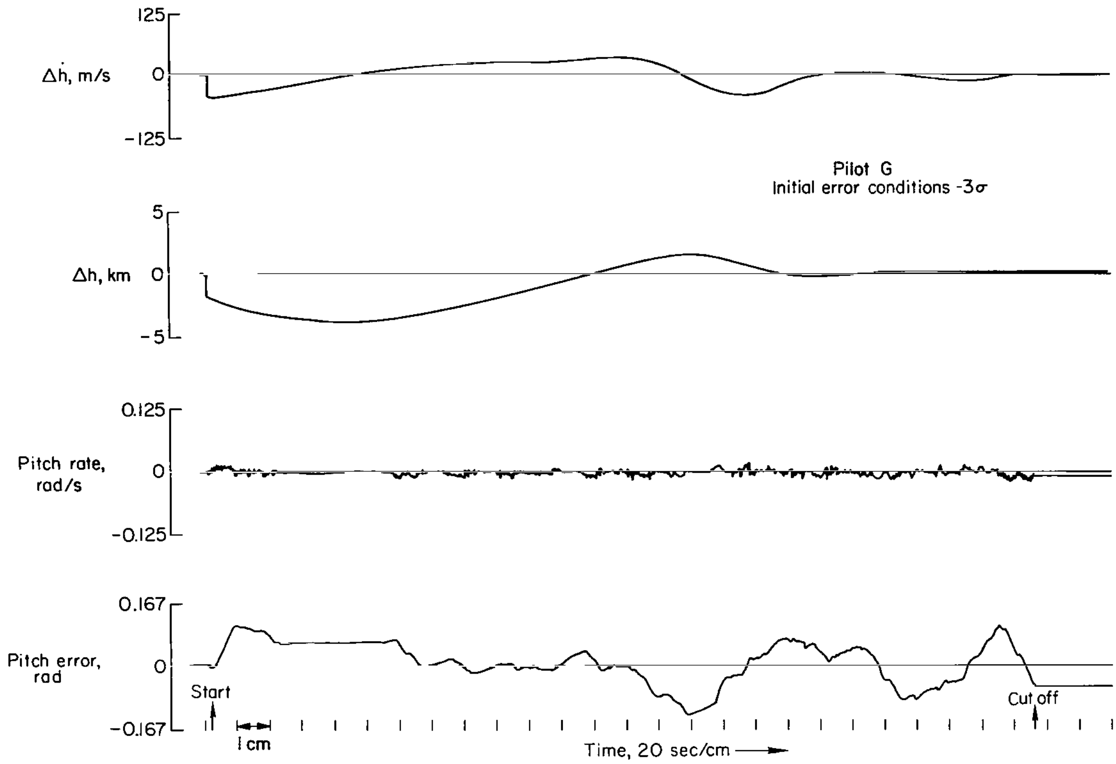


Figure 6.- Typical time history data for upper stage flights using the rate command manual backup system.

Attitude trim system.- The effect of varying controller gain for the attitude trim system is shown in the data of figure 7. The injection error parameters, vehicle bending, oxidizer slosh, and pilot opinion rating are shown as a function of controller trim rate ($\Delta\dot{\phi}_\delta$). The difference in injection performance for the three trim rates is seen to be negligible. Since this control system causes a ramp input to the control computer (rather than the step type as for the previous system), bending accelerations are almost zero and maximum amplitudes of oxidizer slosh are considerably reduced. The pilots rated the 0.5 degree per second trim rate as near optimum.

A series of simulated flights were then made by each of two pilots using the 0.5 degree per second trim rate. The injection error data are shown in the appendix (fig. 11). The altitude and altitude rate data generally lie within the performance criteria. Mean error and standard deviations of altitude and altitude rate for the 3σ initial dispersion conditions for the two pilots were:

	Mean error	Standard deviation
Altitude	-0.14 n.mi.	0.60 n.mi.
Altitude rate	9.4 fps	24.9 fps

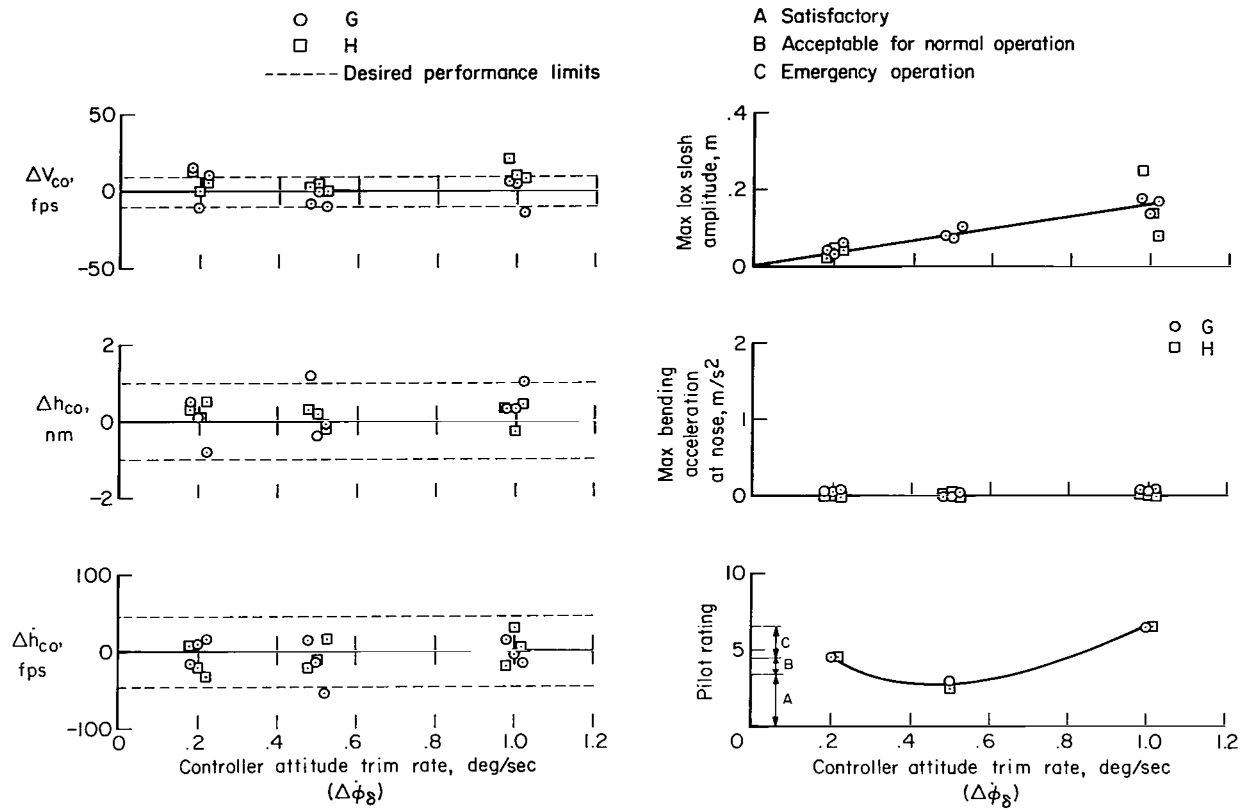


Figure 7.- Effect of controller gain on injection performance, sloshing, bending, and pilot rating for upper stage flights using the attitude trim manual backup system.

Some of these runs were made without the digital display of attitude bias $(\Delta\varphi_8)$. While both pilots preferred to have the bias data available, their performance data show no strong effect of not having it.

Figure 8 shows a typical time record of attitude, altitude, and associated rates for a simulated flight by pilot G with -3σ initial dispersions. The traces generally show characteristics similar to those for the rate command system discussed previously and shown in figure 6. However, the smoother control inherent in the attitude trim system is evident in the pitch error trace. Maximum pitch rates for this system were 0.01 radian per second (about 0.6 degree per second).

Comparison summary.- The mean errors and standard deviations of altitude and altitude rate at cutoff for the 3σ initial condition flights, for all pilots using the two manual backup control systems were:

		Δh_{co}, n.mi.		$\Delta \dot{h}_{co}$, fps	
System	No. of Runs	Mean	Standard deviation	Mean	Standard deviation
Rate command	25	0.15	0.54	5.0	8.9
Attitude trim	15	-0.14	0.60	9.4	24.9

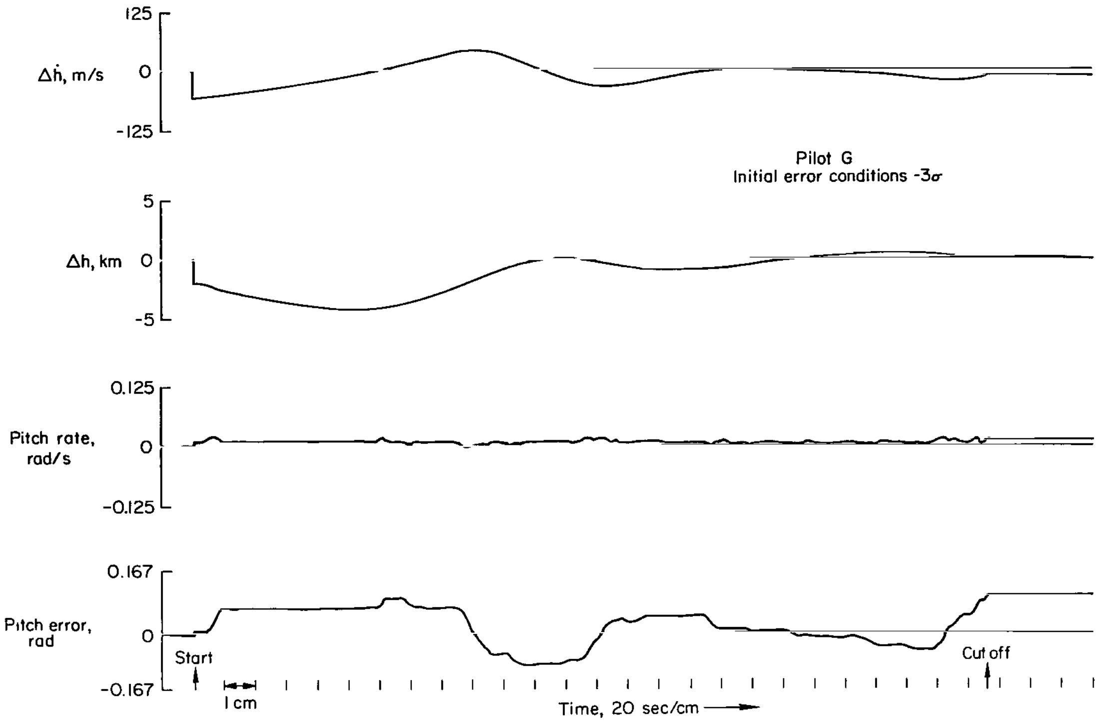


Figure 8.- Typical time history data for upper stage flights using the attitude trim manual backup system.

The performance of the rate command and attitude trim systems is similar. The altitude mean error and standard deviation are essentially identical for the two systems. The differences in the altitude rate error at cutoff probably result from the trim system being inherently more sluggish than the rate command system when last minute corrections are attempted. The data for both systems lie well within the guidelines; changing the initial condition magnitudes did not significantly change the performance.

While both the rate-command and the attitude-trim systems show good guidance performance, the control system characteristics cause large differences in propellant sloshing, body-bending excitation, and pilot opinion (figs. 5 and 7). The rate command system would be the simplest to mechanize as it would require a minimum of software changes. While the attitude trim system would be more difficult to mechanize, it would provide less excitation of the sloshing and bending dynamics.

Thrust cutoff.- A second crew member monitored the velocity display and gave the pilot a shut-down command. The pilot then pushed a simulated thrust termination button. In the actual vehicle, thrust would be terminated by a timer sequence activated by rotating the translation controller counterclockwise with his left hand. Since the velocity display was updated at 2-second intervals, the velocity change had to be monitored in increments of approximately 40 feet per second and interpolated for third stage cutoff. The ΔV_{co} results in figures 5, 7, 10, and 11 indicate that the crew member was able to do this. Mean and standard deviation of the error in velocity at cutoff were 6.70 and 10.3 fps, respectively.

Fuel penalties.- Some preliminary studies at Marshall Space Flight Center (MSFC) indicate that the fuel penalties with the manual backup systems would be quite small. In a digital computer simulation, MSFC forced their vehicle model to fly a typical piloted trajectory (including initial dispersions) obtained from the present analog simulation study. The result was compared to that obtained with the Saturn V primary, iterative-guidance mode. Automatic thrust cutoff was assumed for both digital computer cases (as opposed to the manual technique used in the analog study). The extra fuel used to obtain earth orbit for the piloted trajectory for this one example was 260 kilograms. This is about 0.057 percent of the nominal amount used by the upper stages through earth orbit insertion. Additional fuel may be required to correct the orbit plane errors that may develop with a manual control system. This has not yet been determined.

First Stage Control

The remainder of the results concern the control of the Saturn V first stage. This section is divided into two parts: (a) selection of the pilot's controller gain, and (b) pilot procedures and system performance during system failures.

Controller gain.- The two controller systems (rate command and attitude trim) were also examined for first stage control. For the controller sensitivity study, two representative, single axis (yaw plane) control tasks were chosen: (a) to assist the automatic system to maintain near zero attitude error in the presence of a maximum design wind (95 percent wind, with a 99 percent shear near the time of maximum dynamic pressure, ref. 5), and (b) to control the vehicle's attitude with a simulated single-engine actuator hard-over (engine hard-over to 5° at 20 seconds), with no wind.

Rate command system.- The pilots' performance for the two control tasks as a function of controller gain as measured by three performance indices (see appendix, fig. 12) (bending moment, lateral accelerations, and pilot opinion) indicate that this system is not useful for first-stage control especially in the case of a simulated failure. (Only one data point for lateral acceleration was available for the failure task because of recorder malfunction.) The main problem, of course, is the effect of the step controller input on the highly flexible first-stage vehicle. The excitation of the flexible body dynamics not only contributes directly to the bending moment, but causes fairly severe motion cues and obscures the rigid body content of the displays. For these reasons, the rate command system was not studied further.

Attitude trim system.- Figure 9 shows the pilots' performance as a function of controller gain for the two control tasks with the attitude trim system. Performance indices are maximum attitude error, bending-moment ratio, lateral acceleration motion cues, and pilot-opinion rating. For the actuator failure tasks (solid symbols), the maximum vehicle bending moment occurred at the time of failure, and was independent of controller input. Therefore, the measure of performance was his ability to follow the nominal trajectory and was indicated by the maximum attitude error rather than by bending moment.

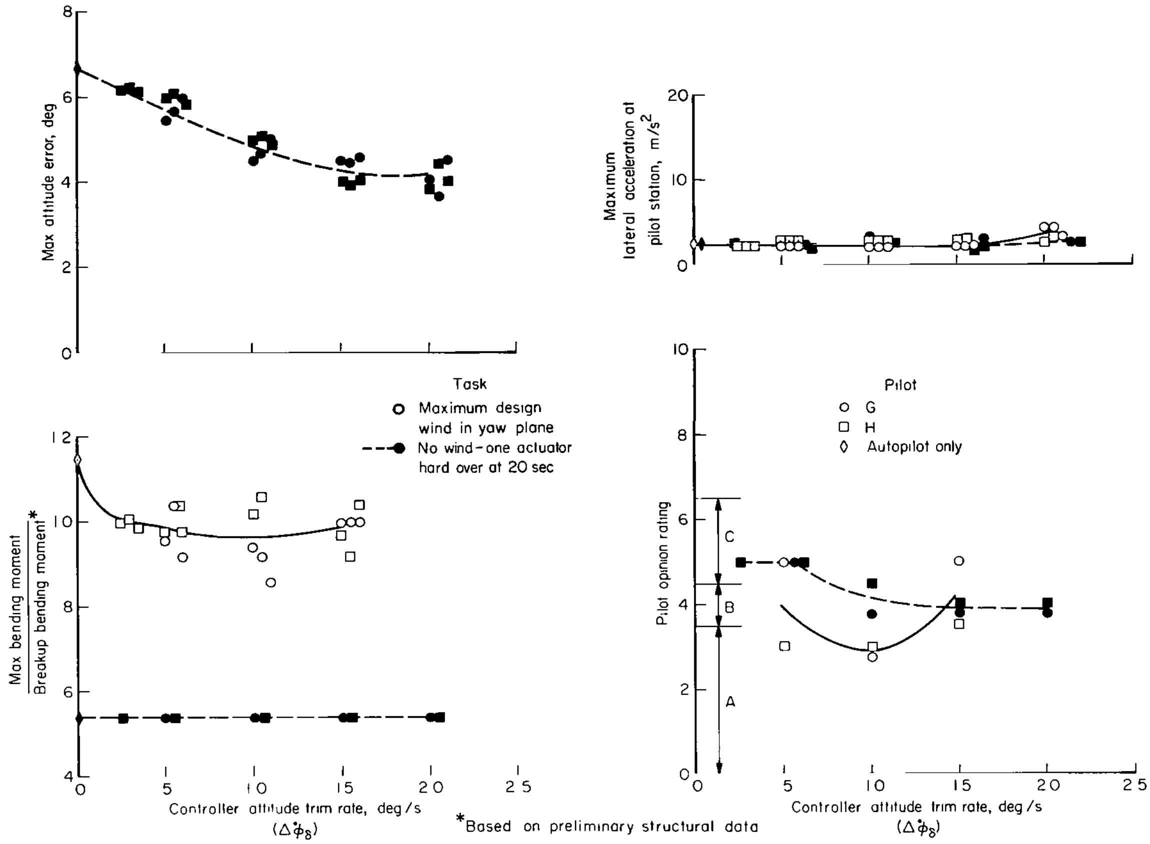


Figure 9.- Effect of controller gain on attitude error, bending moment, motion cues, and pilot rating for first stage flights using the attitude trim manual backup system; S-IC stage data, single axis task (yaw).

Five controller attitude trim rates were studied for the two control tasks mentioned above; the performance was compared with that for an "autopilot only" (zero trim rate).

The bending moment data indicate that the vehicle would breakup for many of the piloted runs and for the autopilot run when subjected to the maximum design wind (fig. 9). The data for the autopilot runs (diamond symbols) show that the design moment was exceeded by more than 10 percent. Preliminary structural data were used to determine these bending moment ratios; the design moment will change when structures are modified or when a different wind environment is used for predicting air loads. Therefore, the important factor in this study is to note the relative difference in bending moment data for the autopilot and the piloted runs. The pilot's contribution to the control task can be seen by comparing the data from the piloted system with that for the autopilot only (diamond symbols) in figure 9. The bending-moment data (open symbols) indicate the ability of the pilot to reduce the aerodynamic loads by trimming the vehicle's attitude in the presence of wind. The decrease in bending moment was about the same for all the trim rates used. However, at the higher gain or trim rate, the controller becomes oversensitive. This is shown in the pilot-opinion rating data. The best (minimum) pilot rating average for this task occurred for trim rates of about 1°/sec. Transverse accelerations at the pilot station were low in all cases.

The solid symbols show the data for the failure-mode control task (one actuator hard over). The maximum attitude error and pilot-opinion rating data show an increase in performance with increased trim rate up to 1.5°/sec. The curves tend to flatten for rates of 1.5° and 2.0°/sec. A trim rate of about 1.5°/sec seems best for this control task.

From the data for both tasks, it appears that a single gain level of about 1 degree per second will give adequate performance. Therefore, a gain level of 1 degree per second was chosen for the remainder of this study.

Pilot Procedures and System Performance Under System Failure Conditions During First Stage Burn

After the attitude trim system and controller gain were selected, a brief study (similar to that of reference 3) was made of pilot procedures for controlling the vehicle during first stage burn for certain types of system failures. Preliminary simulation results indicated that the pilot, using the attitude trim manual backup system could contribute little or nothing to improving system performance (i.e., reduction of bending moment) in the event of oscillatory instability of one actuator, loss of attitude rate, or if one actuator became inoperative. However, the pilot could assist the automatic control system during two types of failures: (1) engine actuators hard over, and (2) loss of thrust. These two types of failures also have a high probability of occurrence relative to the other failures mentioned above.

Various situations were considered for these two types of system failures during first stage simulated flights, in the presence of a 50 percent probability wind with a 99 percent shear occurring at 70 seconds. This synthetic wind profile was obtained from reference 5, which states that these steady-state values will not be exceeded 50 percent of the time during the windiest month of the year, nor will its vertical shear be exceeded 99 percent of the same time period. The peak wind shear was conservatively chosen to occur near the time corresponding to vehicle maximum dynamic pressure (70 sec).

The failure situation parameters were: time at which the failures occurred (before or during maximum q time regions); one actuator hard over in pitch or yaw, or two actuators hard over in pitch and yaw; loss of thrust; and the direction the vehicle rotates, relative to the wind vector, as a result of the failure. The vehicle rotates away from, normal to, or into the relative wind as a result of torques placed on the vehicle by the loss of thrust, etc. The direction of the vehicle rotation determines the aerodynamic loading effects at the time of failure. The various combinations of failure situations considered are listed below.

Flight region	Type of failure	Failure turns vehicle
Pre max q[1]	One actuator hard over[2]	Away from wind
Max q[3]	One actuator hard over	Away from wind
Pre max q	Two actuators hard over[4]	Normal to wind
Max q	Two actuators hard over	Into wind
Max q	Two actuators hard over	Normal to wind
Max q	Loss of thrust, one engine	Into wind
Max q	Loss of thrust, one engine	Away from wind
Max q	Loss of thrust, one engine	Normal to wind

[1]25 to 40 seconds
[2]5° in pitch or yaw
[3]65 to 80 seconds
[4]5° in pitch and 5° in yaw

The primary performance criterion for the failure mode tasks is the ratio of maximum vehicle bending moment to breakup bending moment. The resultant of the pitch and yaw bending-moment ratios was used for data presentation.

The equation used for calculating the ratio of maximum vehicle structural bending moment to breakup bending moment is:

$$M = \frac{\partial M}{\partial \beta} \sum_{i=1}^{4} \beta_i - \frac{\partial M}{\partial \alpha} \alpha + \sum_{j=1}^{2} \frac{\partial M}{\partial \ddot{\eta}_j} \ddot{\eta}_j + \sum_{k=1}^{3} \frac{\partial M}{\partial \xi_k} \xi_k$$

where

M body bending moment normalized to unity at a factor of safety of 1

α aerodynamic angle of attack, deg

β_i swivel angle of the ith control engine, deg

η_j acceleration at nose of the jth flexible body normal mode, m/s^2

ξ_k amplitude of the kth propellant tank sloshing mass, m

The effects of propellant sloshing damping forces on bending moment were neglected. The partial derivatives above were assumed to vary with time. Typical values near the time of flight corresponding to high q are as follows:

$$4 \frac{\partial M}{\partial \beta} \; . \; . \; . \; \frac{1}{5} \text{ per deg} \qquad \frac{\partial M}{\partial \ddot{\eta}_1} \; . \; . \; . \; 0.04 \text{ per } m/s^2$$

$$\frac{\partial M}{\partial \alpha} \; . \; . \; . \; \frac{1}{11} \text{ per deg} \qquad \frac{\partial M}{\partial \xi_k} \; . \; . \; . \; 0.2 \text{ per m}$$

This equation was used to calculate the bending-moment ratio for all situations except for loss of thrust. The unsymmetrical loading, resulting from loss of an engine's thrust, requires an extra term in the bending-moment equation and an increase in the partial derivatives. The equation used to calculate the bending moment for the engine thrust loss condition was

$$M = \left(\frac{\partial M}{\partial \beta}\right)' \left(\sum_{i=1}^{4} \frac{T_i}{T_n} \beta_i \pm 3\beta_o\right) - \left(\frac{\partial M}{\partial \alpha}\right)' \alpha + \dots$$

where

$$\left(\frac{\partial M}{\partial \beta}\right)' = 1.25 \frac{\partial M}{\partial \beta}$$

$$\left(\frac{\partial M}{\partial \alpha}\right)' = 1.10 \frac{\partial M}{\partial \alpha}$$

$$\beta_o = 0.0697 \text{ rad } (4^\circ)$$

The β_o term results from the unsymmetrical vehicle loading, while T_i/T_n is the ratio of actual thrust of the ith engine to nominal thrust.

In the event of either an engine actuator failure or a loss of thrust, the pilot's primary procedure was to maintain vehicle attitude close to nominal values. It should be noted that the pilot's controller was activated at all times for this study. In other words, no switching action was required at the time of failure. The pilot was briefed on the wind direction before each simulated flight. In the event of an engine actuator failure, the pilot could use this knowledge of the wind direction to increase system performance. Large vehicle attitude transient errors occur when an actuator swings hard over to its limits of travel. The pilot can reduce structural loads in these cases by pointing the vehicle into the wind, thereby reducing aerodynamic loads. When a loss of thrust occurs, unsymmetrical loads are set up in the launch vehicle structure. In this case, the pilot procedure is to induce a compensating aerodynamic load on the vehicle. This is accomplished by controlling the vehicle so that some attitude error exists in the direction of the failed engine, rather than completely nulling the attitude error. In all situations, the pilots have to allow the vehicle time to follow their trim commands. Vehicle response for the attitude trim system is slow and there is a tendency to overshoot the desired attitude.

The various failure situations were demonstrated to each of the participating pilots who practiced the recommended procedures in the flight simulator. After these familiarization runs, data flights were made wherein the various failure situations were presented to the pilot in random order for a series of runs. The series was such that each pilot flew at least three flights for each failure situation. Relative performance was obtained by simulating these same failure situations with "autopilot only" control.

The performance of the manual backup attitude trim system is shown in the appendix (actuator-type failures in fig. 13 and loss-of-thrust-type failures in fig. 14).

As mentioned earlier, preliminary structural data were used in load computations, and therefore, the important point is the performance of the manual backup system relative to the "autopilot only" system. This comparison can be seen more clearly by the summary of the data presented in table I. As shown

TABLE I.- MAXIMUM BENDING MOMENT SUMMARY

Type of failure	Flight region	Vehicle rotation relative to wind	Average of [Max bending moment / Breakup bending moment][1]		Change in average maximum bending moment ratio due to pilot input, percent
			Autopilot only	Pilots	
One actuator hard over (5°)	Pre max q max q	Away Away	0.58 1.28	0.58 1.12	0 -12.5 Average -6.2
Two actuators hard over (5° in pitch, 5° in yaw)	Pre max q max q max q	Normal Into Normal	0.95 1.06 1.14	0.95 1.04 1.17	0 -1.9 2.6 Average 0.3
Loss of thrust (one engine)	max q max q max q	Into Away Normal	0.99 1.48 1.29	1.10 1.10 0.99	11.1 -24.5 -23.2 Average -12.2

[1]Breakup bending moment is based on preliminary structural data.

in the right-hand column, the average change in bending-moment ratio for the piloted system data versus the "autopilot only" data is -6.2 percent for single actuator failures, 0.3 percent (increase) for double actuator failures, and -12.2 percent for loss of thrust. The negative sign indicates a reduction in bending-moment ratio. Previous studies at Ames have shown that further reductions in bending moment could be obtained if a proportional controller and load relief information were given to the pilot (ref. 3).

An additional benefit, which results from the use of the pilot attitude trim system, is the significant reduction in trajectory dispersions. The "autopilot only" system requires an attitude error signal to balance the torque effects caused by an actuator hard over or loss of thrust; therefore, the vehicle drifts away from the nominal trajectory. The pilot can bias this torque directly and thereby keep vehicle attitude and trajectory closer to nominal values. The appendix shows lateral velocity and position dispersions at the end of the first stage burn for the actuator failure cases (figs. 15 and 16). The resultant lateral velocity and position error data were calculated as the vectorial sum of the v_2 plus v_3, and x_2 plus x_3.[1] The absolute magnitudes of these resultant errors are summarized in table II.

TABLE II.- VELOCITY AND POSITION ERROR SUMMARY

Failure	Average of absolute value of resultant lateral	Autopilot mean	Pilot data	
			Mean	Standard deviation
One actuator hard over	Velocity error (m/s) Position error (km)	177 7.55	18 0.93	33 2.05
Two actuators hard over	Velocity error (m/s) Position error (km)	264 11.19	76 4.87	54 3.29

With a one actuator hard-over failure, use of the piloted attitude trim system reduced the mean value of velocity error by a factor of 10 and the position error by a factor of 8. For two actuators hard over, the reductions were a factor of 3 in velocity error and better than 2 in position error. The pilot's ability to maintain the desired attitude would indicate that a similar decrease in dispersions could be expected for the thrust-loss failures. (This simulation did not accurately compute trajectory data after thrust failures.)

CONCLUSIONS

Fixed-cab piloted simulation studies of manual backup guidance and control of the Saturn V launch vehicle from liftoff to earth orbit insertion have been conducted. The results indicate that a manual "attitude trim" of an autopilot control system can be used effectively as a manual backup system. The autopilot used for this backup system is a closed loop attitude system using the spacecraft inertial platform and a stored nominal attitude program in the spacecraft computer. Another system called the "rate command" system, which used manual attitude feedback, was found effective for upper-stage control, but the step input from this system adversely affected the sloshing and bending dynamics during first-stage flight.

Upper stage studies showed that the pilot can successfully inject the Apollo system into a circular earth orbit within the guidelines specified by Manned Spacecraft Center. Mean error and standard deviation (1σ) of the injection parameters when using the rate command system were: altitude error = 0.15 ±0.54 n.mi., altitude-rate error = 5.0 ±8.9 fps. For the attitude trim system, mean error and standard deviation were: altitude

[1]The orientation of the velocity and position error components is: v_1 and x_1 in line with the nominal inertial velocity, positive for downrange error; v_2 and x_2 perpendicular to v_1 and the nominal pitch plane, positive to the south for an eastward launch; v_3 and x_3 perpendicular to v_1 and v_2, positive toward the earth.

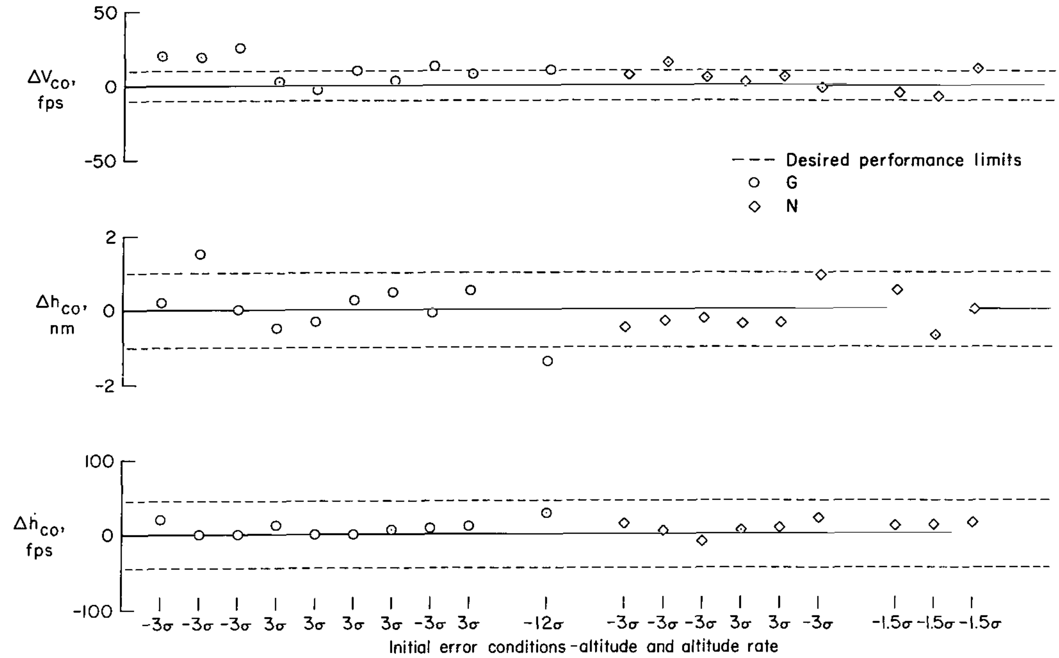
error = -0.14 ± 0.60 n.mi., altitude-rate error = 9.4 ± 24.9 fps. Combining the thrust cutoff data for both systems shows mean and standard deviation velocity error values of 6.7 ± 10.3 fps. Preliminary studies indicate fuel penalties using the manual backup system are about 0.05 percent of nominal.

First-stage manual backup control, in the presence of actuator hard-over or thrust failures, showed slight reductions in bending moment using the attitude trim system, when compared to the backup autopilot only system. However, considerable reduction in trajectory dispersions was found for the failure modes and wind conditions studied. Average lateral velocity and position errors at first-stage burnout were reduced by a factor of 2 or better for a two-actuator hard-over failure, and by about a factor of 10 for a single actuator hard-over failure.

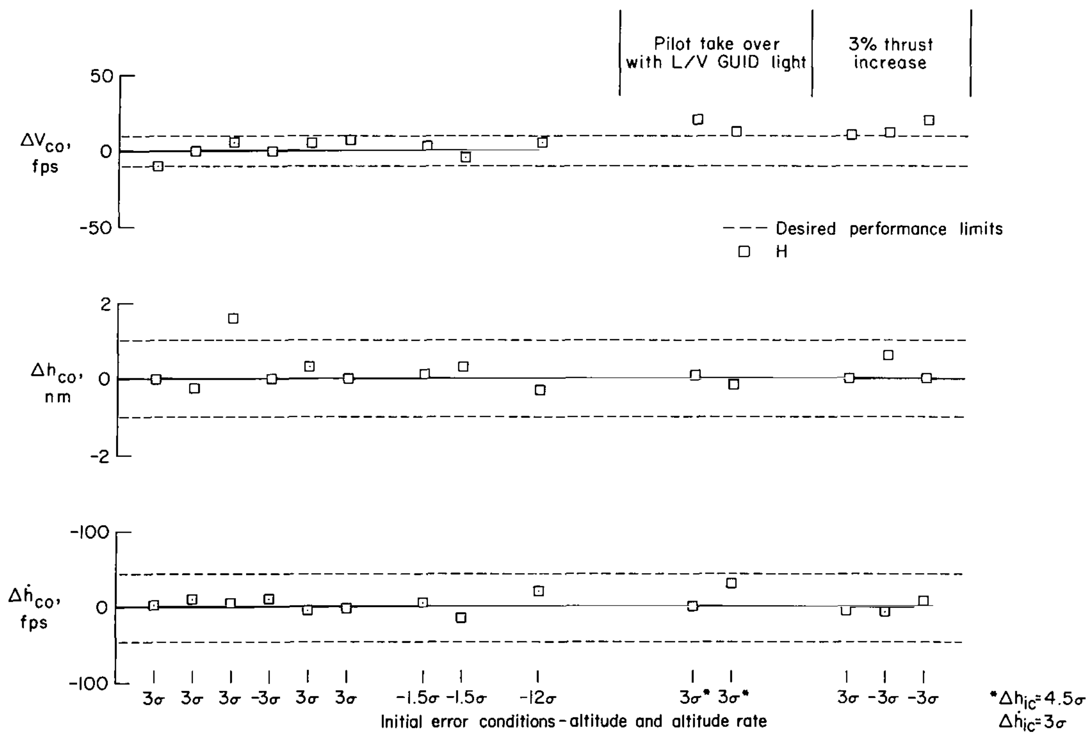
Ames Research Center
National Aeronautics and Space Administration
Moffett Field, Calif., 94035, Nov. 13, 1967
125-19-01-32-00-21

APPENDIX

PILOT PERFORMANCE DATA



(a) Subjects G and N.



(b) Subject H.

Figure 10.- Earth orbit injection performance for upper stage flights using the rate command manual backup system.

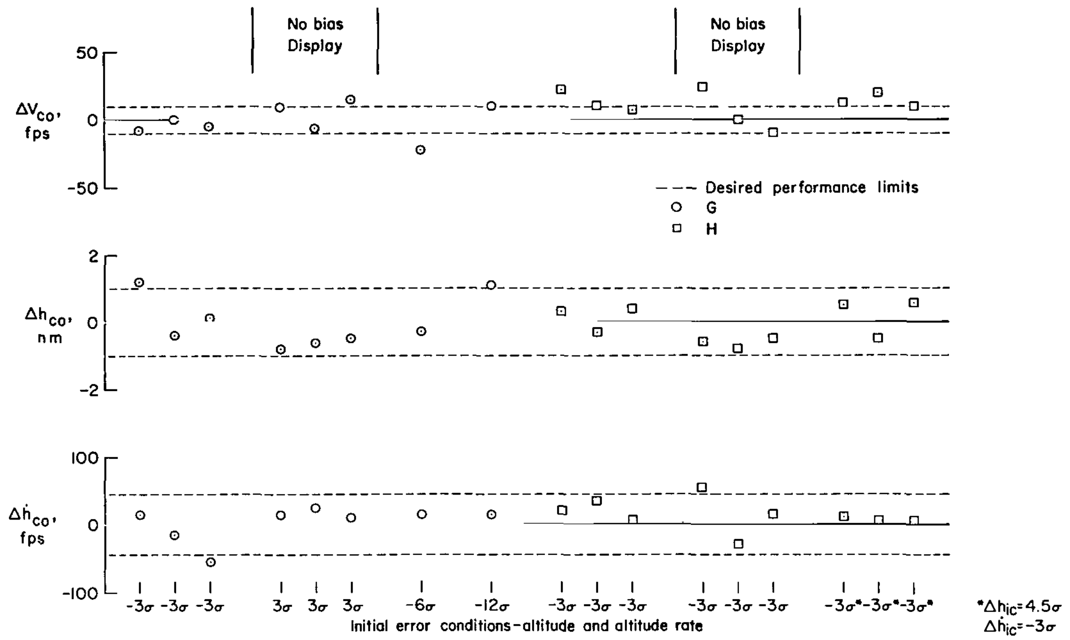


Figure 11.- Earth orbit injection performance for upper stage flights using the attitude trim manual backup system.

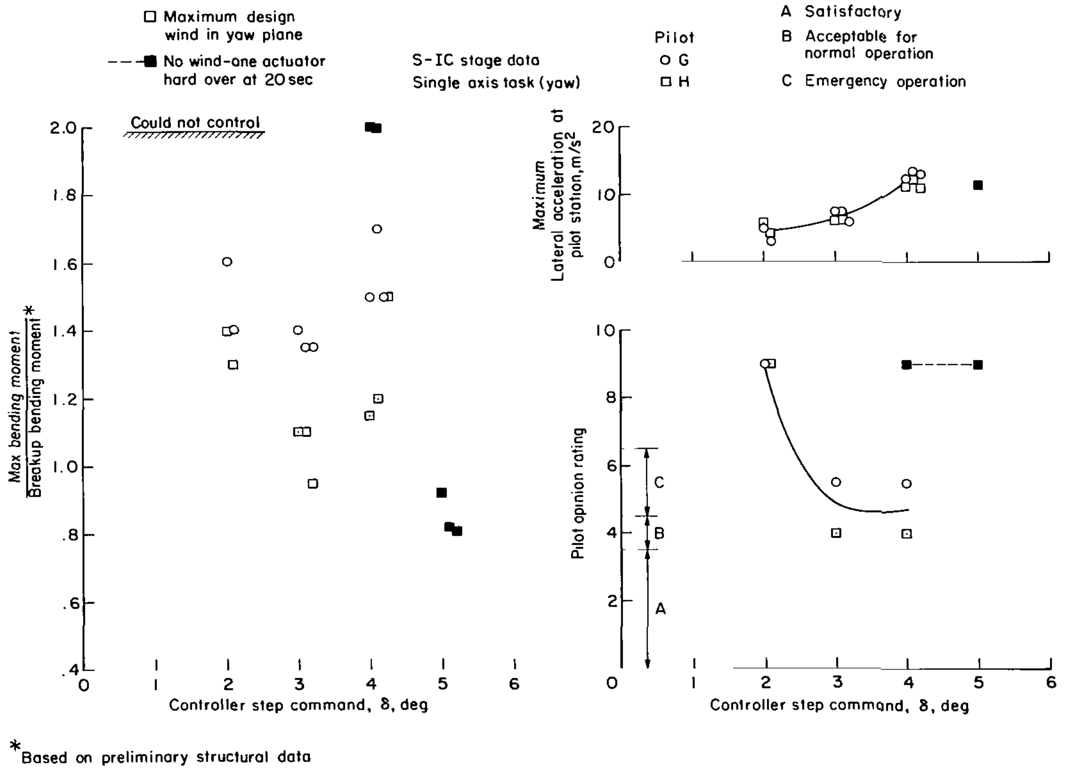


Figure 12.- Effect of controller gain on bending moment, motion cues, and pilot rating for first stage flights using the rate command manual backup system.

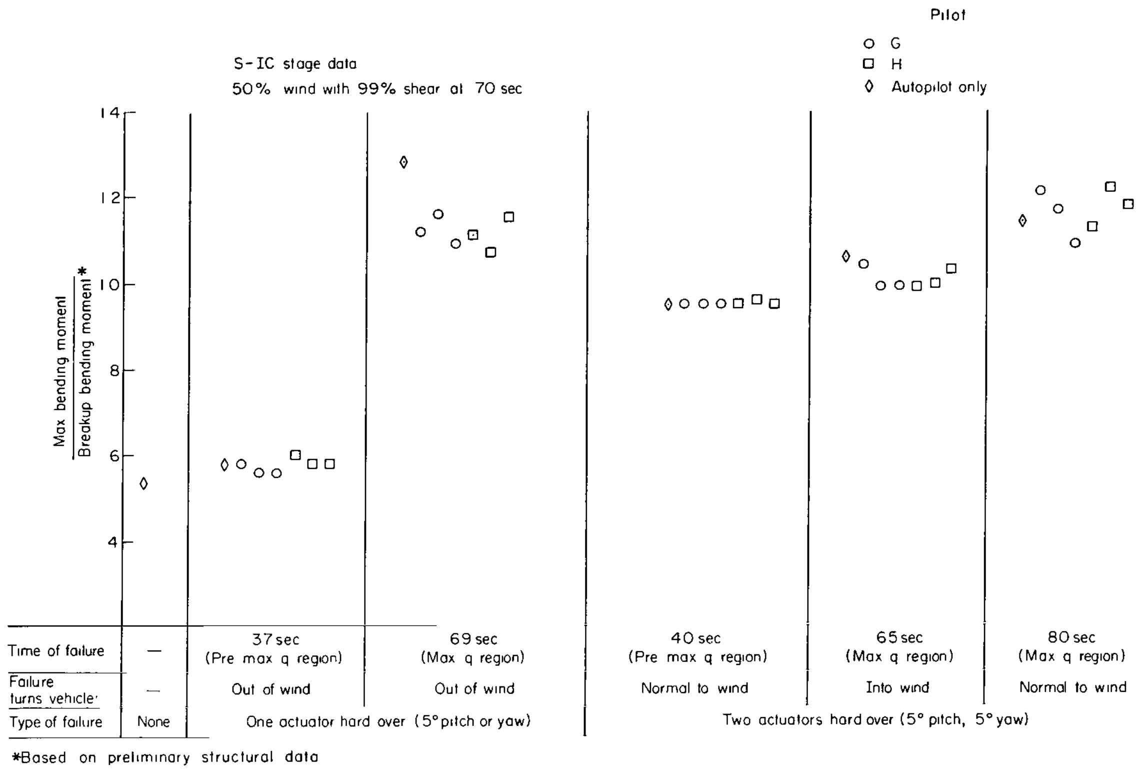


Figure 13.- Maximum bending moment ratios for actuator failures during first stage flight.

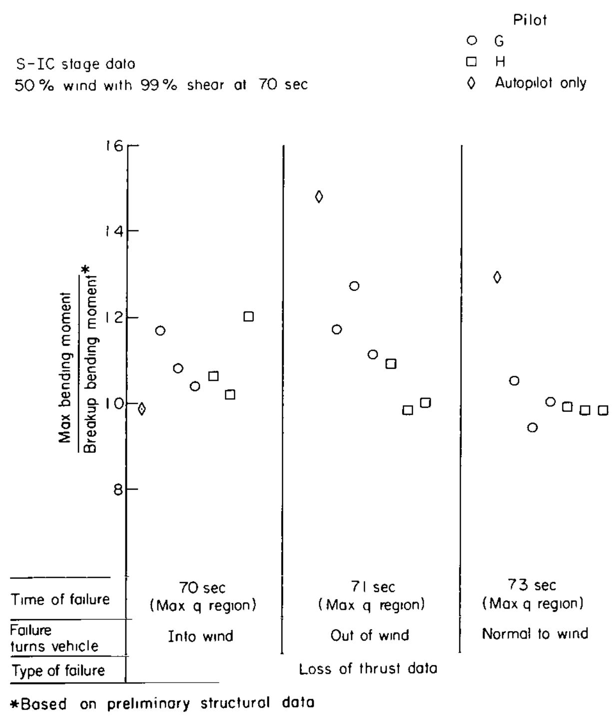


Figure 14.- Maximum bending moment ratios for thrust failures during first stage flight.

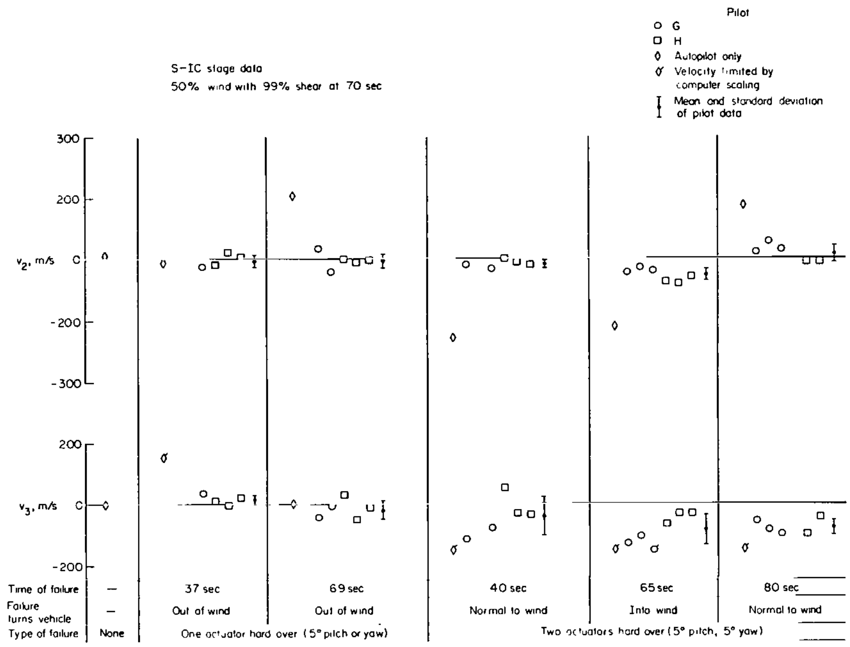


Figure 15.- Velocity dispersions at first stage burnout for actuator failure cases.

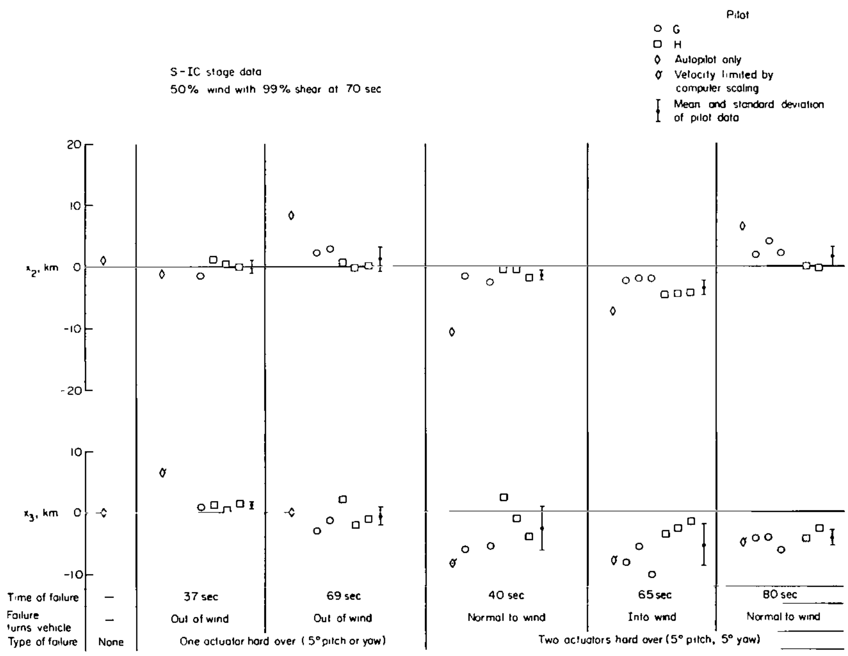


Figure 16.- Position dispersions at first stage burnout for actuator failure cases.

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