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SUMMARY

A detailed study was made of manual backup control systems suitable for the first stage of the Saturn V launch vehicle. A technique to measure the manual control system reliability with a piloted simulator was developed. Two manual backup control systems were considered: a "load relief" and a "no load relief" system. Both systems allowed the pilot to close an adaptive control loop that is parallel to the primary automatic control system. The system failure modes associated with the primary system, as well as those associated with the additional hardware for the piloted backup system, were considered. An analog piloted simulation was used and included rigid-body, engine-actuator, vehicle-bending, propellant-sloshing, and control-system dynamics. Over a thousand simulated flights with randomly selected failures were made with three research pilots. The results indicate that for the failure modes and automatic system considered, the piloted manual backup system can reduce the probability of mission failure by a factor of 2. Trajectory dispersions at first-stage cutoff were significantly reduced. The hand-controller and pressure-suit configurations had little effect on system performance.

INTRODUCTION

Since early 1963, the Ames Research Center and the George C. Marshall Space Flight Center (MSFC) have been jointly investigating the problem of piloted guidance and control of large launch vehicles. Early studies of this group and others (refs. 1-5) have established the feasibility for this class of vehicles.

At the conclusion of these feasibility studies, the question was posed: "Can the probability of mission success be improved by incorporating the pilot in a backup mode to the primary automatic control system of the Saturn V launch vehicle?" The purpose of the present investigation was to answer this question.

The first section of this report is a description of the system to be controlled and its environment. It also includes a discussion of the candidate manual backup control systems to be considered, as well as a discussion of the constraints on the guidance and control system. The second section outlines the technique used to measure the reliability contributed by the addition of a manual backup control system. It also discusses the failure modes considered, the details of the simulation used, and pilot procedures. The third section discusses the results obtained, which include structural loads experienced, trajectory dispersions, and mission reliability. The effect of pressure-suit and hand-controller configurations on performance is also discussed.

Some of the results presented herein are also presented in reference 6. The purpose of this report is to amplify these results and, in addition, to discuss several other aspects of the problem. A discussion of handling qualities, hand-controller and pressure-suit configurations, and detailed pilot procedures are added. In addition, more complete data on structural loads and trajectory dispersions are included.

This investigation was limited to the first stage of flight, the S-IC.

VEHICLE AND CONTROL SYSTEM DESCRIPTION

The Saturn V vehicle configuration (fig. 1) has three booster stages, an instrument unit, and the Apollo spacecraft. The vehicle has a relatively high fineness ratio (length/diameter) with a first flexible body mode frequency of about 1 Hz. Several of the propellant tanks have significant sloshing modes with frequencies of about 0.5 Hz.

The first stage burns for approximately 150 seconds. Figure 2 shows some typical trajectory parameters. Maximum dynamic pressure, q, occurs at about 78 seconds. The maximum thrust-to-weight ratio, T/mg_0, reaches almost 5.

The first stage is powered by five F-1 engines. The attitude of the Saturn V during the first stage is controlled by swiveling the four outboard F-1 engines. Each engine is swiveled in the pitch and yaw planes by separate hydraulic actuators. Roll is controlled by combined use of pitch and yaw actuators.

The feasibility study (ref. 1) indicated that a piloted control system including load relief (reduction of aerodynamic loads) had merit; however, the load relief feature requires additional hardware. Therefore, two manual backup control systems were proposed for study: a "load relief" system and a "no load relief" system. Figure 3 shows the elements of these systems. The solid lines in the lower half of the figure indicate the elements of a launch-vehicle automatic control system. Reference 7 presents a comprehensive discussion of this system. The engine actuator command signals are attitude rate and attitude error, summed, gained, and filtered in the control computer. The configuration of the filters used during the study is presented in appendix A. The solid lines in the upper half of the figure indicate pertinent (to manual launch control) existing spacecraft control-system components. Existing pilot display items include the attitude, altitude, velocity, flight-path angle, etc., of the spacecraft. The dashed lines indicate the items added for the proposed manual backup systems. These items will be discussed in detail below. Not shown in figure 3 are switches that allow the pilot to selectively open the attitude rate or attitude error feedback loops making up the engine actuator commands. Both systems allowed the pilot to form an adaptive parallel control loop that he can activate when failures occur in the primary system.

Load Relief System

Attitude error, from the launch-vehicle guidance system, as well as outputs from body-mounted accelerometers in the launch vehicle was added to the pilot display for the load relief system. The accelerometers were located near the instantaneous center of rotation of the vehicle (ref. 1) so that their outputs were nearly proportional to $q\alpha$, the product of dynamic pressure and angle of attack. Aerodynamic loads on the vehicle are directly related to this product. The output of the pilot's controller was passively filtered and summed with the output of the launch-vehicle automatic system at the control computer. This filter was a passive second-order network with natural frequency of 2.7 rps and a damping ratio of 0.5.

To determine whether the characteristics of the automatic control system were satisfactory for manual control, a brief investigation was made. A pilot was asked to fly the simulated vehicle with various settings of the gains in the automatic control system. The pilot's subjective opinion of the system and system performance data were recorded for each combination. The pilot opinion data are shown in figure 4, and the maximum structural bending moment experienced, ratioed to the breakup value, is shown in figure 5. The abscissas indicate the rate loop gain used. The units are in degrees of engine gimbal command per degree per second of attitude rate. The ordinates indicate the attitude loop gain used (degrees of engine gimbal command per degree of attitude error). A pilot opinion rating of 3-1/2 or less is considered satisfactory, while a rating of 4-1/2 or less is considered acceptable for normal operation (ref. 8). Figures 4 and 5 indicate that while higher values of damping (rate gain) are desirable, the nominal values for the automatic system are satisfactory. These nominal values cause the vehicle rigid-body short-period mode to have (near maximum dynamic pressure) a natural frequency of about 0.8 rad/sec and a damping ratio of about 0.4.

No Load Relief System

The no load relief system was identical to the load relief system except that it had no body-mounted accelerometers or associated display.

GUIDANCE AND CONTROL CONSTRAINTS

The principal constraints on the launch-vehicle guidance and control system are guidance accuracy and structural loads. Since the study reported here considers the first stage of flight, structural loads were the primary constraint. 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

$\ddot{\eta}_j$ acceleration at the nose of the vehicle of the jth flexible body normal mode, m/sec^2

ξ_k amplitude of the kth propellant tank sloshing mass, m

The effect of propellant sloshing damping forces was neglected. The partial derivatives above were assumed to be time varying. Typical values near the time of flight corresponding to high q are as follows:

$$4 \frac{\partial M}{\partial \beta} = \frac{1}{5} \text{ per deg}$$

$$\frac{\partial M}{\partial \alpha} = \frac{1}{11} \text{ per deg}$$

$$\frac{\partial M}{\partial \ddot{\eta}_1} = 0.04 \text{ per m/sec}^2$$

$$\frac{\partial M}{\partial \xi_k} = 0.2 \text{ per m}$$

Because of the different vehicle loading for one engine thrust out, the following equation was used for this failure mode:

$$M = \left(\frac{\partial M}{\partial \beta}\right)' \sum_{i=1}^{4} \frac{T_i}{T_n} (\beta_i \pm \beta_0) + \left(\frac{\partial M}{\partial \alpha}\right)' \alpha + \ldots$$

The primes on the partial derivatives indicate different values corresponding to the different longitudinal loading in the vehicle. The β_0 term results from unsymmetrical vehicle loading, while T_i/T_n is the ratio of actual thrust of the ith engine to nominal thrust.

While not as significant during first-stage control, trajectory dispersions were also calculated as a measure of guidance performance. Distance and velocity normal to the nominal trajectory at first-stage cutoff were used.

Two synthetic wind magnitude profiles were used for this study. They were obtained from reference 9 and are shown in figure 6. The steady-state value of the larger magnitude profile will not be exceeded 95 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 steady-state value of the other profile will not be exceeded 50 percent of the same time period. Peak wind shear occurs near the altitude corresponding to vehicle maximum dynamic pressure. A preliminary investigation showed that the small-amplitude gusts discussed in reference 9 had little effect on the manual control problem. Two wind directions were chosen, 135° and 225° relative to vehicle launch heading. Previous experience has shown that these quartering winds are the most difficult for piloted control.

METHOD OF ANALYSIS

The systematic analysis technique used to assess the effect of the piloted backup system on system reliability and mission success is similar to one that has been used for the automatic system. It is also similar to the "Pilot-Controller Integration for Emergency Conditions" concept (ref. 10), which was refined and applied to the X-22A V/STOL vehicle (ref. 11).

The seven steps of the technique are shown in figure 7 and are discussed below.

1. Define the system. Collect the necessary information on the vehicle, systems, trajectory, mission, etc., to enable a simulation to be conducted. Define the manual control system.

2. Define the major failure modes. Predict major failure modes, define failure dynamics and obtain necessary information to simulate failure modes, obtain unreliability number (probability of occurrence) for each major failure mode.

3. Simulate the system and failure modes. Use the data gathered in steps 1 and 2 and appropriate mathematical models to develop a real-time piloted flight simulation of the vehicle and its major failure modes.

4. Define the pilot procedures. Use the flight simulation developed in step 3 to conduct a systematic investigation wherein the failure modes investigated are made to occur at various times of flight with the pilot in control of the simulated vehicle. From this investigation, develop a background of information from which the crew can learn to detect and correctly identify each failure as well as to follow the correct pilot procedure in the event of a failure. (Most of the emergency section of the pilot's handbook is written during this study phase. Also, at this time, preliminary changes to the proposed manual system can be made.)

5. Conduct simulation with random failures. Use several subjects and a large number of simulated flights with random failures to determine the conditional probability of mission failure (effectivity) for each of the major failure modes.

6. Calculate the probability of mission failure. Use the unreliability numbers from step 2 and the effectivity numbers from step 5 to calculate the failure mode criticality (effect of failure on probability of mission failure).

7. Modify the system and procedures as necessary. Analyze the results of step 6 to determine which failure modes have the greatest influence on mission failure. Redesign the system of step 1 or modify the procedures developed in step 4 as necessary to reach a suitable level of "probability of mission success."

The application of these seven steps to the first stage of Saturn V follows.

Step 1: Define the System

The system is defined in the preceding Vehicle and Control System Description.

Step 2: Define the Major Failure Modes

The launch-vehicle major failure modes (as opposed to component failure modes) can be divided into three categories: control-system hardware failures (sensors, wiring, etc.), engine actuator failures (hard over, null, oscillating), and thrust failures. The first 10 failures in figure 8 are the major failure modes for the launch vehicle considered in the order of their assumed unreliabilities (probabilities of occurrence). The unreliability numbers are typical for Saturn V. Failure modes 11 through 19 are associated with the displays and controller, which were added for the piloted backup system. The unreliability data for the pilot displays are not shown because, as will be seen later, no mission failures were caused by a display failure.

Failure 1 in figure 8, one engine actuator hard over, can be caused by a servoamplifier or a valve blockage. It was simulated by a step change in the actuator command to a saturate value.

Failure 2, loss of thrust on one control engine, is caused by a loss of thrust chamber pressure and was simulated by a step change on one engine, to zero thrust.

Failure 3, pitch and yaw actuators inoperative on the same engine, can be caused by a loss of hydraulic pressure on one engine. The engine will then drift under the influence of the aerodynamic, inertial, and thrust misalinement forces present. It was simulated by drifting the engine to either hard-over deflection in pitch and yaw (5°) or to a 2.5° deflection in pitch and yaw.

Failure 4, loss of the attitude reference platform, can be caused by hard-over platform. The hard-over signal is interrupted by the reasonableness check in the digital computer. The attitude error signal is then frozen at the last reasonable value. It was simulated by freezing the attitude error signal at the value existing just before the time of failure.

Failure 5, one engine actuator oscillatory, can be caused by a failure of the mechanical feedback from actuator to value. It was simulated by opening the position feedback loop in the simulated engine actuator dynamics.

Failure 6, loss of attitude rate signal to the control computer, can be caused by a rate gyro or demodulator going to null. It was simulated by a step change to zero on the attitude rate signal.

Failure 7, one engine actuator inoperative, can be caused by a servoamplifier output going to null, causing the engine deflection to null in one axis. It was simulated by a step change to zero in the actuator command.

Failure 8, loss of attitude error signal to the control computer in one axis, can be caused by a failure between the launch vehicle data adapter and the summing amplifier in the control computer so that the attitude error signal path is opened or blocked. It was simulated by a step change to zero on the attitude error signal.

Failures 9 and 10, attitude signal saturate and attitude rate signal saturate, can result from a control computer failure. The saturated attitude signal was simulated by a step change to 11.5° in the attitude error signal. The saturated rate signal was simulated by a step change to 10.0° per second in the attitude rate signal.

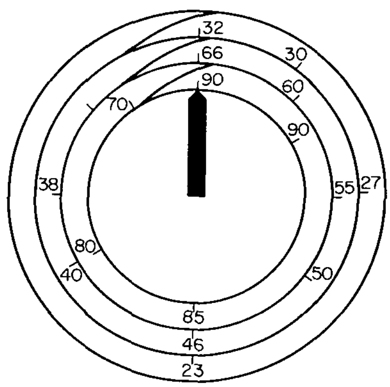
Failure 11, attitude display lock, jump, or drift, can be caused by a variety of component failures (e.g., loss of motor drive or loss of feedback). Failure 11 was simulated by causing the drive signal to the display to freeze at the time of failure value, to jump to some arbitrary value, or to start drifting at a constant rate from the time of failure value.

Failures 12 through 17, attitude-error, attitude-rate, and accelerometer null or saturate, were simulated by a step change in the displayed signal to null or saturate, respectively.

Failures 18 and 19, the pilot's hand controller null or saturate, were simulated by a step change in one axis of the controller output to null or saturate, respectively.

Step 3: Simulate the System and Failure Modes

Display panel.- A comprehensive fixed-cab analog simulation was used. The display panel (fig. 9) was representative of Apollo. The all-attitude indicator in the center of the panel displayed vehicle attitude on the sphere. For the load relief manual system, attitude error was displayed on the auxiliary meters on the left and top of the indicator with the outputs of the body-mounted accelerometers displayed on the flight director needles. Attitude error for the no load relief manual control system was displayed on the flight director needles (no acceleration signal was displayed). Attitude rates were presented on three separate indicators as shown. The clock used is just to the left of the all-attitude indicator. The following nominal boost attitude profile is plotted around its circumference for comparison with the mission time as displayed by the sweep second hand of the clock.



If it is assumed that the second hand is shown in the zero mission time position, the nominal attitude should be 90°. One full minute later, the nominal attitude is 66°. The nominal attitude at staging (2-1/2 min) is 23°. The failure warning lights are to the right and below the all-attitude indicator. The lower five lights indicate thrust loss on any one of the five F-1 engines. Of the remaining lights, only the one labeled "L/V Guidance" was used. This light indicates a failure in the launch-vehicle attitude reference platform. The six switches at the lower left of the panel could be used to open the three rate and three attitude loops in the automatic system. The hand-controller switch activated the pilot's representative three-axis hand controller. A thumb button switch on the controller could be used (in parallel with the one on the instrument panel) to activate the controller.

Pilot's hand controllers.- Two hand controllers were used in this study: an Ames controller and an Apollo Block I controller. The Ames three-degree rotational hand controller was used for the basic data of the reliability study (fig. 10). It is a sidearm controller having three rotational degrees of freedom that command the roll, pitch, and yaw of the vehicle. Strain gages mounted on flexure pivots provide the electrical output signal. The handle is mass balanced for use in a high g environment. The characteristics of this controller are shown as output voltage versus displacement and torque in figure 11. A nonlinearity was added to the Ames controller output, which gave the pilot a variable gain controller that desensitized his input for trajectory control, and yet gave him full authority at maximum displacement for handling failure situations. The characteristics of this nonlinearity are shown in figure 12.

An Apollo Block I hand controller was obtained for an evaluation and comparison with the Ames controller (fig. 13). It is a three-axis rotational hand controller with discrete and proportional outputs. The proportional mode was used for this study. Its output voltage versus displacement and torque characteristics are shown in figure 14. The controller is characterized by large breakout forces with values equal to 50 percent of maximum values.

Pressure suit.- While the basic data of the reliability study were obtained with the pilots in shirt sleeve garb, a pressure suit was obtained for one of the pilots to determine its effects on pilot performance. The Gemini type pressure suit (model SPD-766-1 made by International Latex Corporation) shown in figure 15 was used both in the pressurized and unpressurized mode. In the event of a cabin pressure failure in the Apollo Command Module, the suit pressure is maintained at 3.7 psia. This pressure differential was simulated for the pressurized mode with a gage pressure of 3.7 psi above atmospheric pressure. A compressed air-flow rate of 4 ft^3/min was used for body cooling. Figure 16(a) shows the pilot, with a nonpressurized suit, seated in the simulator cab with the Apollo Block I hand controller. The "ballooning" effects of pressurization can be seen by comparing this figure with figure 16(b).

Equations of motion.- Time-varying coefficient, linearized equations of motion were simulated to describe vehicle dynamics. These included three propellant sloshing modes, two flexible-body modes, six degree-of-freedom rigid-body dynamics, engine actuator dynamics, and the control-system shaping networks. The equations used are similar to, but more extensive than, those of reference 1. A 400-amplifier analog computer complex with extensive function generation capability was used. Switching was used to allow selective simulation of the various system failure modes.

Step 4: Define the Pilot Procedures

Initial simulation results were used to develop a comprehensive set of pilot procedures. The complete procedures are presented in appendix B and are summarized below.

The primary task of the pilot before a system failed was to monitor the displays. His only control inputs (load relief system only) were those necessary for load relief in the event of large wind-induced aerodynamic loads.

He reduced the loads by closing the piloted parallel loop by use of the displayed output signals of the body-mounted accelerometers. Reducing these aerodynamic loads gives the vehicle a greater margin of safety in the event of a system failure.

In the event of failure of the launch-vehicle system (i.e., failures 1 to 10), the pilot's "overriding" procedure was to keep the attitude of the vehicle at the nominal value. He did this by operating as an adaptive element in the loop that paralleled the automatic flight-control system.

For hardware failures in the launch-vehicle control system (i.e., loss of platform, attitude rate, attitude signal, etc.), the pilot used information displayed from sensors located in the spacecraft to stabilize and control the vehicle attitude. Specifically, if the launch-vehicle attitude-rate loop malfunctions (i.e., failure 6 or 10) and the vehicle motions become unstable, the pilot, using the displayed-rate information (which is sensed from gyros located in the spacecraft), takes over and stabilizes the vehicle motions. If hard-over control system failures occur, the pilot removes the saturated signal by activating the appropriate switch on the display panel.

In the case of engine actuator or loss of thrust failures, the vehicle develops asymmetric rotational moments. In this case, the pilot acts as an integration-type element in that he injects trimming or bias commands to null the unbalanced or asymmetric rotational moments. The unsymmetrical loading caused by a loss of thrust can be further reduced by a small bias of the nominal attitude toward the failed engine. This induces an alleviating aerodynamic load on the vehicle.

When a single display failed, the information displayed was sufficiently redundant that the pilot was able to detect which instrument had failed and continue to fly the vehicle using the remaining displayed information. The pilot used the ground rule that two indications of a failure were necessary before he assumed control of the vehicle using the backup control system.

Step 5: Conduct Simulation With Random Failures

The following items had to be considered for the actual conduct of the simulation with random failures: the basis for performance comparison, simulation variables, pilot training, and briefing.

Basis for performance comparison.- The principal consideration was "Is the automatic flight-control system plus a piloted backup system more or less reliable than the automatic flight-control system taken alone?" The reliability level of the automatic flight-control system forms the reference condition, thus making it necessary to measure the reliability of the automatic system using the same flight simulation setup, the same flight conditions, etc., that were used for the piloted system.

Simulation variables.- Several variables were considered in the simulation: the number of failures (19 for the load relief system, 17 for the no load relief system, and 10 for the automatic system), the wind magnitude (2, previously described), time of failure (3 major times; before, at, and after high q), and the direction of the wind with respect to the failure (i.e., for some failures the vehicle turns into or away from the wind). From these variables, it was determined that there were 176 basic failure situations for the load relief system, 166 for the no load relief system, and 116 for the automatic system. To make the number of failures approximately proportional to the unreliability, 79 additional situations were added. Each of three pilots flew 255 simulated flights using the load relief system. Since many of the no load relief system situations were similar to the load relief system situations only an abbreviated study was conducted. One pilot was used for 92 simulated flights. A single unknown (to the pilot) failure occurred at an unknown time during each flight. Display and controller failures were deleted for the automatic system reliability study, resulting in 195 simulated flights.

Pilot training and briefing.- A pilot training period preceded the simulated flight series with random failures. The time required varied because of previous pilot experience, but averaged about 30 hours per pilot. For the simulated flight series with random failures, the pilot was briefed on the wind direction and magnitude before each flight.

RESULTS AND DISCUSSION

Reliability Considerations

Step 5, in the analysis procedure discussed above, provided the necessary data for calculating failure mode effectivity (conditional probability of mission failure).

To illustrate the data reduction technique, simulation data for one system failure (one actuator hard over) and one wind magnitude (95 percent) utilizing the load relief control system are shown in figure 17. The times of failure are indicated along the abscissa in the three major time divisions. For the pre-max q and max q time of flight, the direction in which the hard-over actuator turns the vehicle with respect to the wind is also shown along the abscissa. The maximum structural bending moment, normalized to a factor of safety of 1, experienced during the flight is presented on the ordinate. For this example in figure 17, 3 flights (of a total of 45) exceeded the breakup value. These occurred near the time for the maximum wind shear. They were considered unsuccessful flights and yielded an effectivity (conditional probability of mission failure) of 0.045. This was calculated by noting that during the 40-second high q time period (150 seconds is the total first-stage flight time) 3 of the 18 flights were unsuccessful. Since the failure mode unreliability numbers are assumed proportional to time and there were no failures during the pre- and post-max q time periods, this gives an effectivity of (3/18) (40/150) = 0.045.

Simulation data for the example failure for the no load relief and the automatic control systems have been added to the data of figure 17 and are shown in figure 18. The effectivity for the no load relief system was computed as 0.322, which allows for the fact that there was not an equal number of failure samples that turned the vehicle away from the wind as into the wind, the effectivity for the automatic system was computed as 0.488. For this example, both piloted systems improve system reliability. Also, for this example, the load relief system significantly improves performance over the no load relief system (effectivity was reduced from 0.322 to 0.045). Figure 19 summarizes the structural load data for the 19 failure modes. (It should be noted that the maximum computer scaling for the ordinate was 1.5; therefore, if this value was exceeded, 1.5 was used in computing the average values.) The effectivity numbers calculated using this data are summarized in figure 20.

Step 6 in the analysis method (calculate the probability of mission failure) is presented in tabular form in figure 20. The 19 failure modes of figure 8 are in the first column with the unreliability numbers (probability of occurrence) in the second column. The effectivity numbers (conditional probability of mission failure, given occurrence of the failure mode) for the failures occurring with a 95- or a 50-percent wind are given in the fourth and sixth columns, respectively. The failure mode criticalities (the probabilities of mission failure) are the products of the unreliability and effectivity numbers and are indicated for the two wind conditions in columns five and seven. The effectivity and criticality numbers are given for the three systems investigated. The systems are noted in column three. As an example, consider the typical failure situation previously discussed: one actuator hard over, load relief system, and a 95-percent wind. The failure mode criticality is equal to the unreliability multiplied by the effectivity, or $(5450\times10^{-6})(0.045) = 245\times10^{-6}$, as indicated by the first bar of column five. The overall first-stage mission criticality, obtained by summing the individual failure criticalities, is shown at the bottom of the figure. Adding the individual failure criticalities to obtain mission criticality numbers assumes that the probability of more than one failure mode occurring during the first-stage flight is small. The results show (for the failure modes considered) that adding a piloted load relief backup control system reduces mission criticality by a factor of better than 2. For the 50-percent wind case, the no load relief system performs nearly as well as the load relief. For the 95-percent wind, adding the body-mounted accelerometers whereby the pilot could provide load relief, significantly improves the performance.

Step 7 of the analysis technique would feed back the results of figure 20 into changes in the system or pilot procedures. While this was not carried out for the present investigation, several interesting results are shown in figure 20.

The pilot can effectively compensate for certain engine actuator failures (nos. 1, 3, and 7 in fig. 20) and for the loss of the attitude reference platform in the launch vehicle (no. 4). For the other vehicle failure modes with significantly high unreliability numbers, the pilot contributed little to improving the mission reliability.

Loss of roll-attitude rate (no. 6), attitude-signal saturate (no. 9), and attitude-rate signal saturate (no. 7) require opening the automatic control-system loops with one of the six switches in the lower left of the display panel (fig. 9). Since the unreliability numbers for these failures are negligible, the criticality payoff is small considering the additional system complexity.

The pilot could detect the loss of the launch-vehicle attitude platform (no. 4) or loss of the launch-vehicle attitude signal (no. 8) by monitoring the displayed attitude-error signal from the launch vehicle. He compared this signal with the difference between vehicle attitude displayed on the attitude indicator (generated in the spacecraft) and the nominal attitude profile on the clock. Since the unreliability number associated with failure 8 is small, and failure 4 has a warning light, the criticality payoff associated with the display of launch-vehicle attitude-error signal again seems small.

Based on these considerations, it appears that the manual backup system of this study could be simplified without significant performance loss. The simpler system would include only the following elements in addition to those presently in the spacecraft.

(1) Three-axis proportional manual inputs summed with the output of the launch-vehicle automatic system control computer. These manual signals should be passively filtered to attenuate signal amplitudes at vehicle flexible body frequencies.

(2) Display of load relief information to the pilot. This could be from body-mounted accelerometers, angle-of-attack indicators, or perhaps the spacecraft digital computer.

Three additional results that may have application to other backup manual control systems are:

(1) The effect of pilot display failures had no effect on mission criticality (failures 11 through 17 of fig. 20 have zero effectivity) because sufficiently redundant information was presented and because the ground rule used by the pilot was that "two separate indications are necessary before a failure is assumed."

(2) With sufficient practice, the pilot could adapt to the failure dynamics from a strictly monitor mode as fast as from a more active mode of control. This is indicated by the fact that the mission criticality numbers in figure 20 for the 50-percent wind for the load relief and no load relief systems are almost the same. For the no load relief system, the pilot is strictly a monitor and takes no control action until he detects a failure; whereas for the load relief system, he actively reduces structural bending moments due to aerodynamic loading by controlling the outputs of the body-mounted accelerometers. (The small difference in effectivity also indicates that the payoff for accelerometer control for this wind magnitude is small.)

(3) An implicit result of this study, as well as earlier feasibility studies, is the ease with which the pilot can filter the flexible body effects from displayed signals. In the backup mode of operation considered, the pilot was required to act as an adaptive parallel loop to the automatic system. Depending on the failure mode, he was required to close attitude, attitude rate, and/or accelerometer load-relief loops. By observing his display panel, the pilot quite easily distinguished and disregarded the flexible body content of these sensor signals.

Trajectory Dispersions

The discussion so far has centered around the maximum structural bending moment as a measure of mission success. An additional performance index that can be considered is the trajectory dispersions. While dispersions do not affect mission success directly in first-stage flight, they must be kept within limits to allow the upper-stage guidance system to perform adequately. The trajectory dispersions obtained during the study are summarized in figure 21. Data for automatic system flights that were successful, piloted load relief flights with the same failure situations as the auto pilot runs, all successful piloted flights for the load relief system, and all successful piloted flights for the no load relief system are shown for each failure. The pilot contributed the most when large attitude errors (i.e., thrust loss and engine actuator failures) were introduced. Acting as an integration element in the parallel piloted loop, he minimized these attitude errors. The first two columns for each failure mode in figure 21, automatic flights and comparable load relief system flights, indicate that the average dispersions were reduced from 5000 to 2570 m and from 91 to 47 m/sec. This result could be significant to the upper-stage guidance problem. The third and fourth columns compare the performance for all successful load relief and no load relief system flights. The no load relief system has smaller trajectory dispersions as the attitude of the vehicle is held closer to nominal values.

Pressure Suit and Hand Controller Evaluation

Since the study had used a shirt-sleeve environment for the simulation pilots, a short investigation of the effect of a pressure suit on pilot performance was made. A Gemini-type pressure suit was obtained from the Manned Spacecraft Center for this study. In addition, an Apollo Block I hand controller was obtained from MSC and used with the pressure suit.

Pressure suit.- The effect of pilot suit was investigated for a number of failure situations. Three modes of pilot garb were studied: (1) shirt sleeve, (2) Gemini-type pressure suit on but not inflated, and (3) pressure suit on and inflated to 3.7 psig. Data for one subject were obtained as only one pilot could wear the semicustom-fitted pressure suit. Typical data are shown for the actuator hard-over failure situation in figure 22. The data show that the reliability contribution, as measured by the bending moment ratio, is not decreased significantly when the pilot is clothed in a pressure suit, even when the suit is inflated. These data along with the results for other types of failures are summarized in figure 23. The average of the bending moment ratio is plotted for the various system failures for the three suit modes. An overall average for all failure cases is plotted on the right. A small decrease in average bending moment was obtained with the deflated suit. This is believed to be due to pilot training. Inflating the pressure suit tends to degrade the ability of the pilot to fly smoothly and, consequently, it degraded performance slightly. The pilot commented that it was difficult to apply smooth precise inputs with the pressurized glove, particularly in pitch.

Hand controller.- Several of the failure situations flown using the Ames controller were duplicated by two of the subjects, using the Apollo Block I hand controller. Typical data are shown for the same actuator failure cases in figure 24. No distinct advantage is apparent for either controller. The data show the Ames controller better for failures during pre-max q region and vice versa for max q region. A summary of the controller results for several failures is shown in figure 25. Average values of maximum-bending moment to breakup-bending moment ratios are plotted versus failure types for the two controllers. The overall average for all failures is shown on the right. The data indicate that either controller could be used satisfactorily with no significant performance difference. Pilot comments, however, tend to favor the Ames controller over the Apollo Block I controller. The high breakout force of the Apollo Block I controller made it difficult to make small precise inputs.

CONCLUSIONS

A detailed study has been made of manual backup control systems for the first stage of Saturn V. It was concluded that:

1. The handling qualities of the booster with the nominal automatic system are satisfactory for manual control.

2. A comprehensive set of pilot procedures was developed that allowed the pilot to satisfactorily control most failure situations.

3. An analysis method was developed that allowed the systematic determination of the reliability contribution of a pilot to a complex control system. The results indicate that for the failure modes and automatic system considered, the piloted manual backup system can reduce the probability of mission failure by a factor of 2.

4. Single failures of the pilot's display instruments do not affect mission success.

5. The pilot was able to control a failure situation from a monitor mode as well as from a more active control mode.

6. For the Saturn V vehicle, the pilot can act as a highly effective, frequency selective filter.

7. Trajectory dispersions were significantly reduced for the manual control case, particularly for those failure modes causing large attitude errors to be developed by the automatic system.

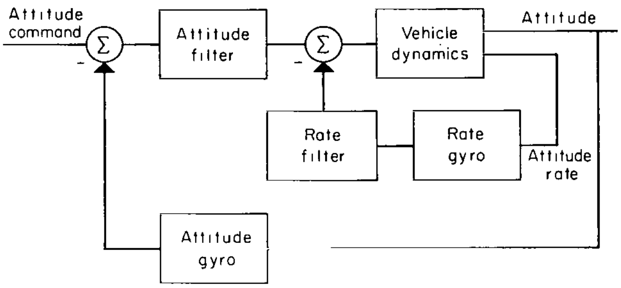
8. Pressure-suit and hand-controller configurations had little effect on pilot performance.

Ames Research Center
National Aeronautics and Space Administration
Moffett Field, Calif., 94035, Feb. 13, 1969
125-19-01-39-00-21

APPENDIX A

CONTROL SYSTEM FILTER CONFIGURATION

The launch-vehicle control-system configuration is shown in the sketch below. All three control axes are similar. A set of preliminary filters was obtained from the Marshall Space Flight Center (MSFC) at the beginning of the study and used for the data presented in the report. At the conclusion of the study, a later set of filter configurations were obtained and the effect on performance measured.



The preliminary filter configurations used for the pitch and yaw axes were

Attitude:

$$\frac{a_o(5.8)^2}{s^2 + 2(0.5)(5.8)s + (5.8)^2}$$

Rate:

$$\frac{a_1(5.8)^2}{s^2 + 2(0.5)(5.8)s + (5.8)^2}$$

where

t, sec	a_o, deg/deg	a_1, deg/deg/sec
0 - 120	0.5	0.5
120 - 150	.2	.3

The later configuration of filters used for the pitch and yaw axes were

Attitude:

$$\frac{a_o(12.6)(s + 0.0951)}{(s + 0.0379)(s + 31.7)}$$

Rate:

$$\frac{a_1(93.1)[s^2 + 2(0.14)(11.15)s + (11.15)^2]}{(s + 2.10)(s + 6.25)(s + 15.38)(s + 57.4)}$$

where

t, sec	a_o, deg/deg	a_1, deg/deg/sec
0 - 100	0.82	0.657
100 - 120	.45	.438
120 - 150	.15	.2

There were no significant differences in performance for the two sets of filters, although a small increase in performance was noted for failure 5 (fig. 8), one actuator oscillatory.

No control system filters were used in the roll axis as no flexible body dynamics were simulated in this axis. The preliminary loop gains used in the roll axis for the study were

$$a_o = 0.173 \text{ deg/deg}$$

$$a_1 = 0.106 \text{ deg/deg/sec}$$

While the later filters had slightly different values for loop gains in the roll axis, the effect was not investigated.

APPENDIX B

PILOT PROCEDURES

The following preliminary emergency procedures were developed and used by the simulation pilots during this study. The discussion covers each type of system failure considered, the indications of failure available to the pilot, and the pilot procedure subsequent to failure identification. The discussion relates to the control system shown in figure 3.

SATURN V EMERGENCY PROCEDURES

(S-IC STAGE)

A. RATE SIGNAL TO CONTROL COMPUTER

The primary cause of trouble in the rate signal to the control computer is the rate gyro itself, but others are possible (wiring, etc.). Two types of failure signals are considered probable: signal to null or signal to saturate.

Pitch or Yaw Signal to Null

In this failure mode, the pitch or yaw rate signal to the control computer changes instantaneously from its normal value to zero. This changes the rigid-body dynamics from a well-damped oscillatory system to a divergent oscillatory system (period = 15-20 sec for flight up to $t = 120$ sec). In addition, the lower modes of elastic dynamics are no longer phase stabilized and now operate essentially open loop (they are still coupled to a small extent by the attitude signal) with less damping.

Indications: 1. Rate, attitude, and accelerometer (during high q) displays start a divergent oscillation with a period of 15-20 seconds.

2. Elastic dynamics (apparent mostly on accelerometer signal) becomes more lightly damped.

Procedure: 1. Provide rate damping by using displayed attitude rate signal.

Roll Signal to Null

In this failure mode, the roll rate signal to the control computer goes to null. Since the attitude loop is still closed, a high-frequency undamped oscillation builds up in the roll channel.

Indications: Roll rate and attitude displays start a high-frequency divergent oscillation difficult for the pilot to control.

Procedure: 1) Disconnect roll attitude augmentation.

2) Continue flight and provide manual rate and attitude augmentation in roll channel.

Pitch, Yaw, or Roll Signal Saturate

In this failure mode, the rate gyro signal to the control computer in one axis changes instantaneously from its normal value to a saturated (limited) value of 10°/sec. The rate loop signal to the engine will then be 5° of engine angle for a pitch or yaw signal saturate (a_0 = 0.5 before 120 sec), and 0.5° for roll. If the other signals to the engine are small, the engines will attempt to go hard over. As the vehicle diverges, the attitude loop and pilot will be making corrective inputs to decrease the engine angle.

At high q (t = 70 sec) and under the influence of a maximum design wind, the vehicle reaches its design load in about 0.3 second. This increases to almost a full second for a failure at t = 40 seconds with the maximum design wind; while at t = 130 seconds, where q is negligible, the vehicle never reaches its design load, but diverges initially in attitude at about 6 or 7 deg/sec.

This failure mode is uncontrollable unless the augmentation system for the affected axis can be disconnected. In most cases, the Emergency Detection System (EDS) will cause an automatic abort before any pilot action is possible.

Indications: 1. Indications of rate, attitude, and accelerometer rapidly diverge.

2. Pilot controller has no apparent effect.

3. There is large normal acceleration at the pilot's station.

Procedure: 1. Disengage the augmentation system from the affected channel as soon as possible.

2. Regain control and continue the flight, flying the affected channel manually.

3. Standby for automatic abort during high q region of flight.

B. ATTITUDE SIGNAL TO CONTROL COMPUTER

Three types of failures are considered possible in the attitude signal to the control computer: signal to null (single axis), signal to saturate (single axis), and signal locks (all axes).

Pitch, Yaw, or Roll Signal to Null

An open circuit in the attitude control loop is a controllable situation. However, a large transient will occur at the time of failure if an attitude error exists (engine command will change by $a_o \Delta\phi$).

Indications: Loss of attitude stabilization is evidenced by a divergent motion of attitude and accelerometer displays.

Procedure: Provide attitude stabilization manually.

Pitch or Yaw Signal to Saturate

If for some reason, the attitude loop signal steps abruptly to a saturate level of 11-1/2° coincident with a severe wind spike, the pilot has about a half second to abort. For Emergency Detection System (EDS) vehicle rate limits of ±4 deg/sec, the abort will be automatic since this rate is exceeded under the above conditions. If the failure occurs after $t = 120$ sec (a_o changes from 0.5 to 0.2), it is possible to control the vehicle; however, the rate limit is again exceeded.

Indications:

1. Rapid divergence of attitude signal.
2. Step change in attitude rate.
3. Abrupt normal acceleration at pilot station (±0.8 g).

Procedure:

1. Disengage the attitude augmentation from the affected channel.
2. Continue flight with manual control.
3. Standby for automatic abort during high q region of flight.

Roll Signal to Saturate

If the roll attitude error signal saturates, the vehicle will go into a steady-state roll rate.

Indications: Steady-state roll rate.

Procedure: Override attitude signal with controller and continue flight.

Signal Freeze

Certain failures in the L/V inertial platform can cause the attitude error signal to the control computer to freeze at the value present at time of failure.

Indications: Loss of attitude stabilization in all axes.

Procedure: Continue flight with manual attitude stabilization in all axes.

C. ENGINE ACTUATOR FAILURES

Four types of engine actuator failures are considered possible: one actuator to null, two actuators drifting (same engine), one actuator hard over, and one actuator oscillatory.

One Actuator to Null

With loss of engine angle command signal, the actuator will abruptly go to the null position. This situation is easily controlled, but the loss of the actuator reduces the available thrust control component and causes an unbalanced roll torque when only three engines respond to a pitch or yaw command. A compensating roll input signal must, therefore, accompany a pitch command if one pitch actuator control is lost. Likewise, if one yaw actuator control is lost, a compensating roll input must accompany a yaw command. The roll attitude indicator is used to determine the sense and amount of roll input under these conditions. There will be similar, but smaller in magnitude, cross-coupling effects for pure roll inputs.

Indications: 1. Cross-coupling of roll with pitch or yaw command input.

2. Reduction of control power.

Procedure: 1. Fly attitude ball to obtain nominal conditions (cross-coupled inputs are required).

Two Actuators Drifting (Same Engine)

Loss of all hydraulic pressure on one engine (due to a hydraulic pump failure) will allow both actuators to drift under the influence of aerodynamic forces, thrust bias, and inertial effects due to cg offset. Depending on the magnitude and direction of these forces, the engine can drift to a variety of positions. It is quite probable that it will drift up to the limit gimbal deflection of 5° in both axes.

While the failure is controllable, if it occurs during high q, the transient while regaining control of the vehicle may overload the vehicle.

Indications: 1. Cross-coupling of roll with pitch and yaw inputs.

2. Reduction of control power on all axes.

Procedure: 1. Regain control and continue flight.

2. For failures before high q, save some control authority for load reduction at wind spike.

Actuator Hard Over

A hydraulic system failure could cause the actuator to abruptly move to the limit of its travel, that is, saturate at an engine angle of ±5° (pitch or yaw). In addition to incurring a large pitching or yawing moment, a roll rate of about 8 deg/sec is induced. This condition can be satisfactorily controlled by a continuous compensating stick command input.

Indications:

1. Abrupt roll rate.
2. Abrupt pitch or yaw rate.
3. Cross-coupling of roll with pitch or yaw command input.
4. Reduction in control power.

Procedure:

1. Fly attitude ball to obtain nominal conditions (cross-coupled inputs are required).
2. Steady roll bias command input necessary.

Actuator Oscillatory

Under certain conditions (loss of actuator feedback), the hydraulic actuator vehicle system could abruptly go into an oscillatory mode at approximately 1/2 cps with ±5° engine angle amplitude. This induces large oscillatory loads in the booster structure, along with noticeable oscillatory normal g loading at the command module station. If this failure occurs during the high q region of flight, the extra loading can cause the vehicle to exceed structural limits.

Indications:

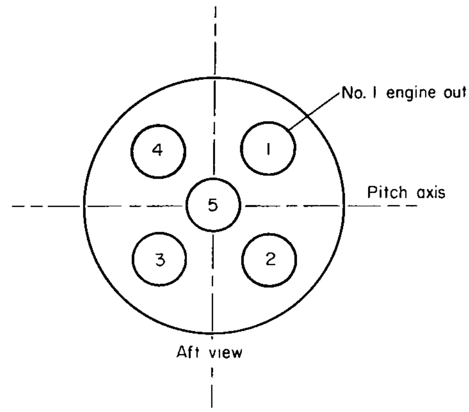
1. Rapid buildup of lateral or vertical oscillatory loads at the pilot station up to about ±0.6 g.
2. Abrupt buildup of pitch or yaw oscillation rate, coupled with an oscillatory roll rate.
3. Oscillatory pitch or yaw and roll attitudes.

Procedure:

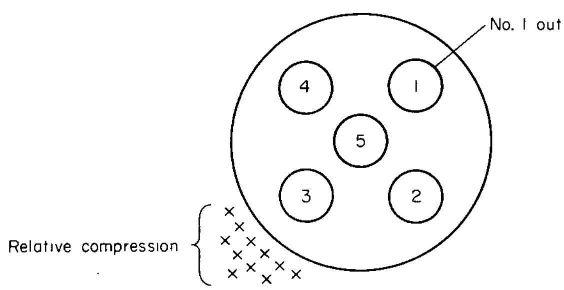
1. Disconnect attitude and attitude rate augmentation in channel with bad actuator (pitch or yaw). Leave roll augmentation in.
2. Continue flight providing stabilization in the affected channel.

D. LOSS OF THRUST ON ONE CONTROL ENGINE

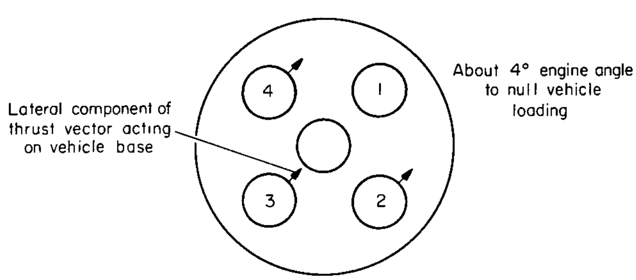
The mechanics involved in a thrust failure can best be explained by an example:



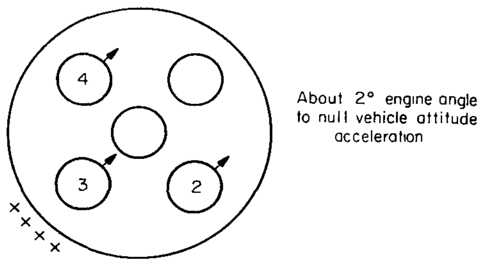
The structural loading in the vehicle is radically changed when the example engine thrust fails. First, with the compressive load in the vehicle now reduced by about 1/5, the load-carrying capability of the vehicle is changed. Since the vehicle is critical in tension (mostly due to interstage structure), the load-carrying capability will be less. In addition, with the engine positions nulled, there is a relative compression on one side of the vehicle due to no. 3 engine thrust.



If it is desired to balance this load, the remaining engines must be swivelled about 4° (52 < time < 89 sec, data for other times were not available).



The control problem also changes considerably when the example engine thrust fails. The unbalanced thrust of no. 3 engine causes an attitude acceleration on the order of $2°/sec^2$. To null this attitude acceleration requires about 2° of engine angle.



With this condition, it can be seen that there will still be a relative compression load on the vehicle. The pilot notes the loss of engine thrust in several ways:

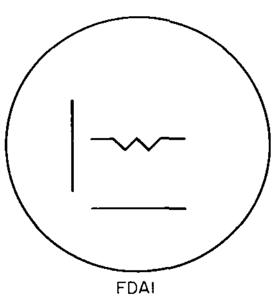
1. Attitude acceleration causes buildup of attitude rate and angle.

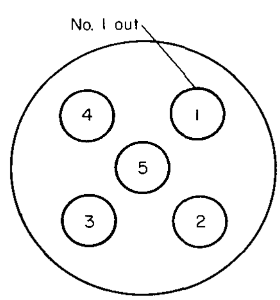
2. Initial attitude acceleration causes a step jump in the accelerometer display.

3. Engine out causes warning light on display panel.

4. Longitudinal acceleration is decreased by 1/5.

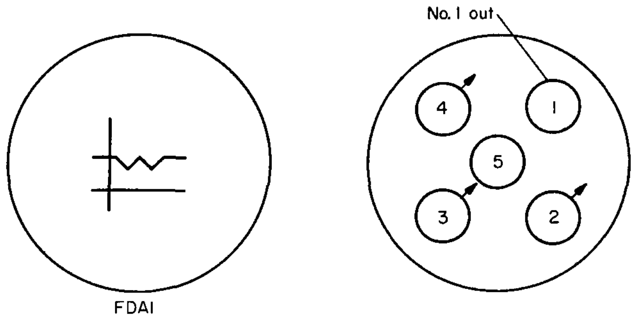
The first cue usually detected by the pilot is the accelerometer needle jump.





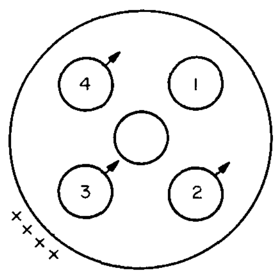
This initial needle jump at thrust failure varies from about 0.7 to 0.4 inch, depending on time of failure.

Because of the engine angle (2°) required to null attitude acceleration, the accelerometer needles will have a bias even after control has been regained.

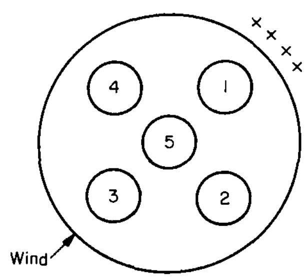


The magnitude of this bias as shown by the accelerometer needles is about 1/3 inch for all times of flight.

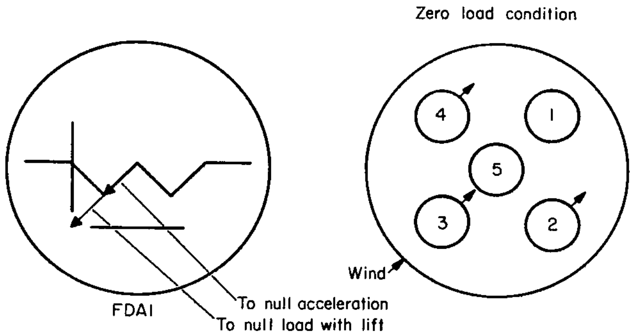
If the vehicle is in a region where the aerodynamic pressure is high enough to develop significant lateral aerodynamic forces, another possibility exists to modify the loading in the vehicle. With the attitude acceleration nulled out as above, a residual relative compression load still exists on the no. 3 engine side of the vehicle.



Since the predominant aerodynamic forces on the vehicle are developed at the nose and tail fins, a wind from no. 3 engine side will cause a balancing load.



This will cause an additional accelerometer indication (about 1/3 inch at max q) in the same direction as that to null the attitude acceleration.



It can be seen that if the pilot attempts to null the accelerometer display in a high dynamic pressure region, he will increase the loading on the vehicle.

Indications: As noted above.

Procedure:

1. Maintain control of the vehicle.
2. Identify which engine has failed as soon as possible (engine out light).
3. Regain attitude control of vehicle.
4. Bias the accelerometer display needles up to 2/3 inch away from the failed engine for high q flight regions.
5. If failure occurs prior to high q save some controller authority for load relief at wind spike.
6. Because of marginal vehicle strength with a failed engine, be prepared for abort.
7. If necessary, switch attitude program to secondary mission profile.

E. DISPLAY AND CONTROLLER FAILURES

Attitude Rate Display

Since the pilot's attitude rate display is used only for monitoring or in the event of other subsystem failures, failure of the rate display itself should have small effect. By comparing its output with the attitude and other displays, failure of the attitude rate display can be verified.

Attitude Error Display

Since the attitude error signal is generated in the L/V, it can be monitored by using the spacecraft, s/c, driven displays. By using the nominal attitude program displayed on the clock and vehicle attitude displayed on the FDAI, the pilot can generate a value of attitude error for monitoring purposes.

Attitude Display

As discussed in the section on attitude error display failure, the purpose of the attitude display is to monitor the attitude error display. If the attitude display failure is obvious (loss of electrical power, gyro tumbles, etc.), then no problem exists. If a slow drift or precession problem exists, then it is difficult for the pilot to know what display is in error, the attitude error or attitude. If a discrepancy exists between the attitude error reading and the FDAI attitude reading minus the nominal attitude and it is not obvious which display has failed, the trajectory (altitude, altitude rate, etc.), parameters and accelerometer display should be monitored closely.

Accelerometer Display

The most probable failure modes for the accelerometer display are saturate or null. These should be obvious to the pilot and, in addition to his knowledge of the existing wind conditions and use of the attitude error display, the pilot can cross check the display.

Pilot's Controller

The two most probable failure modes are saturate or null. In the event one axis of the pilot's controller saturates, the control engines will swing hard over and cause an automatic abort during high q regions. During noncritical times of flight (with auto abort de-armed), large attitude and rate errors will build.

Indications: 1. Divergent attitude and rate errors.

2. Divergent accelerometer indications during high q.

3. Normal acceleration at pilot's station.

Procedure: 1. Disconnect controller.

In the event of the pilot's controller going to null in one axis, no action is required of the pilot.

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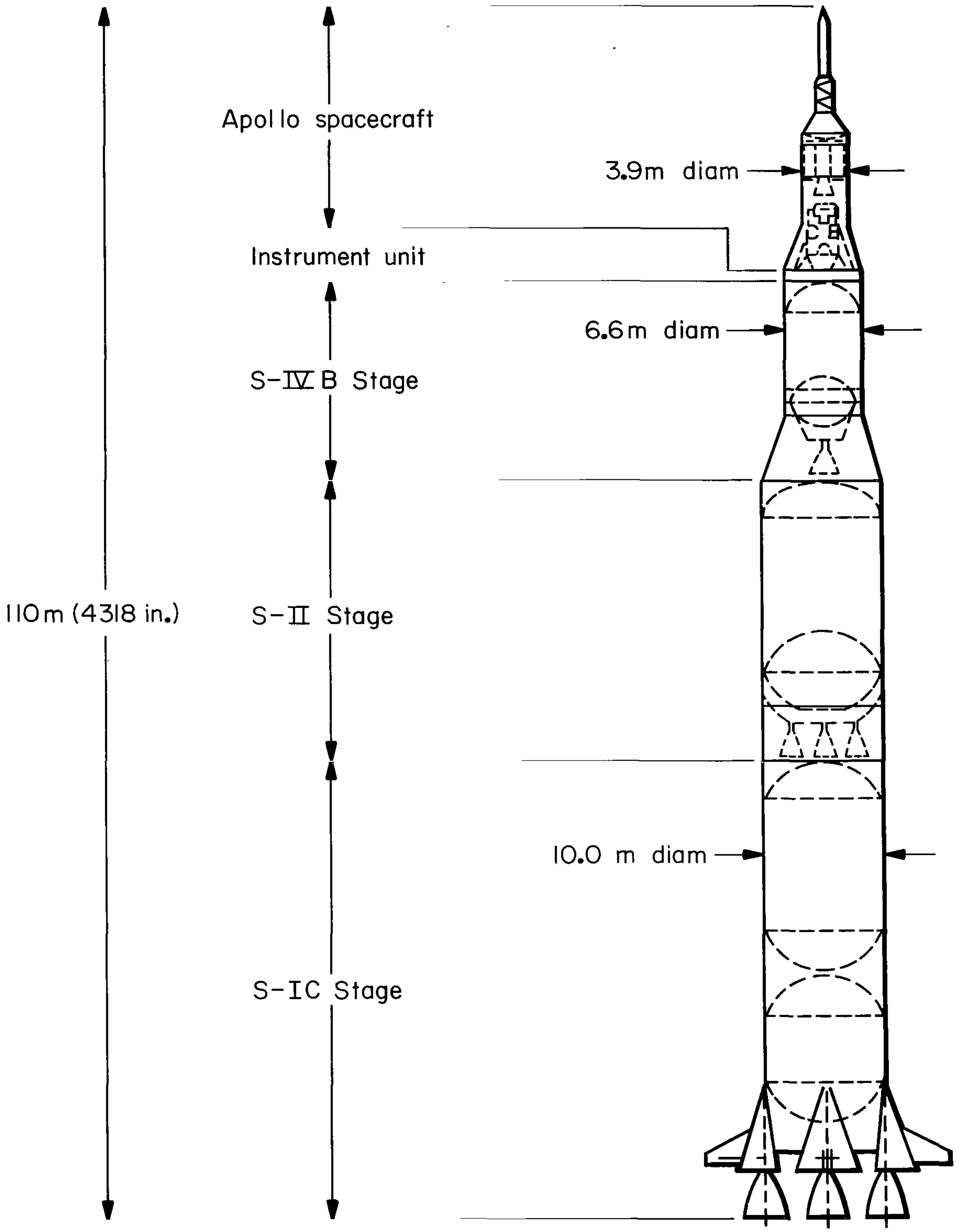


Figure 1.- Saturn V vehicle configuration.

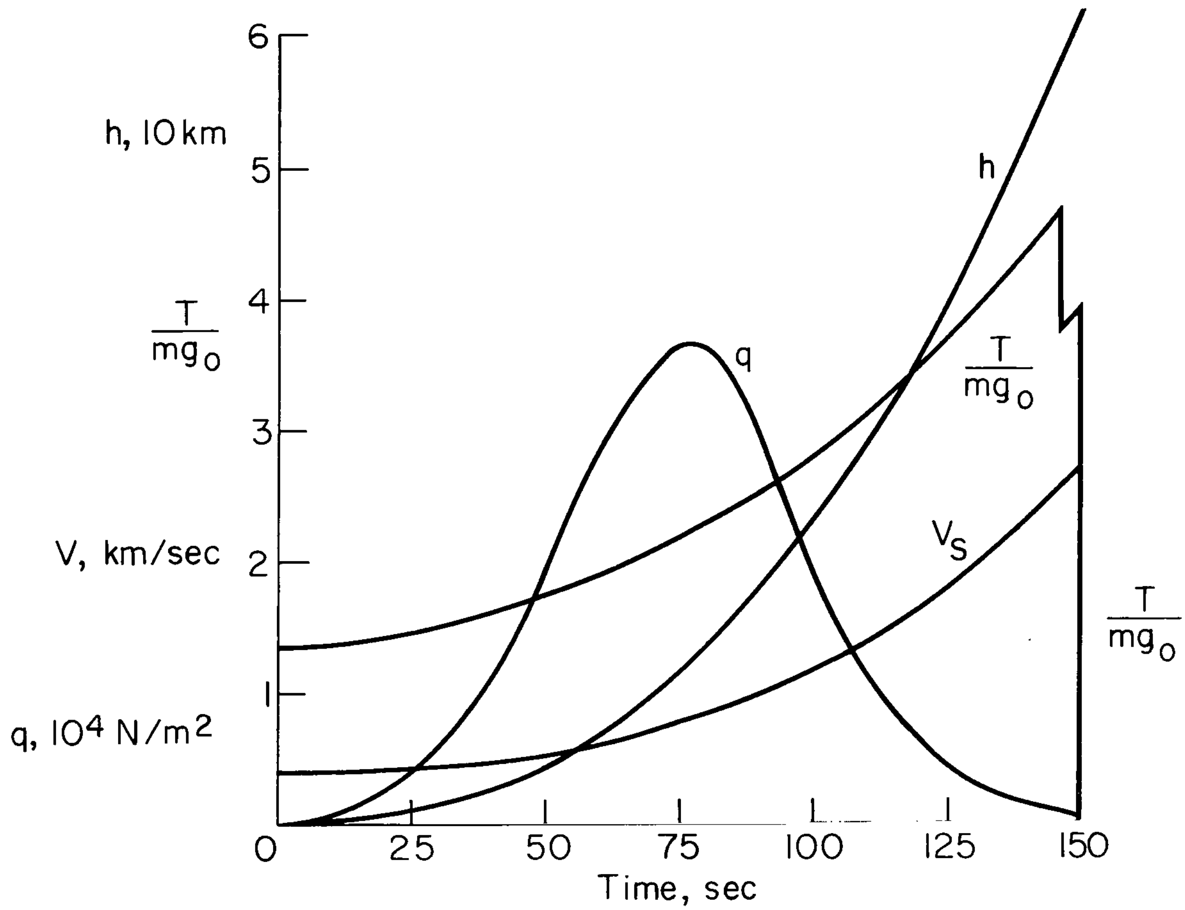
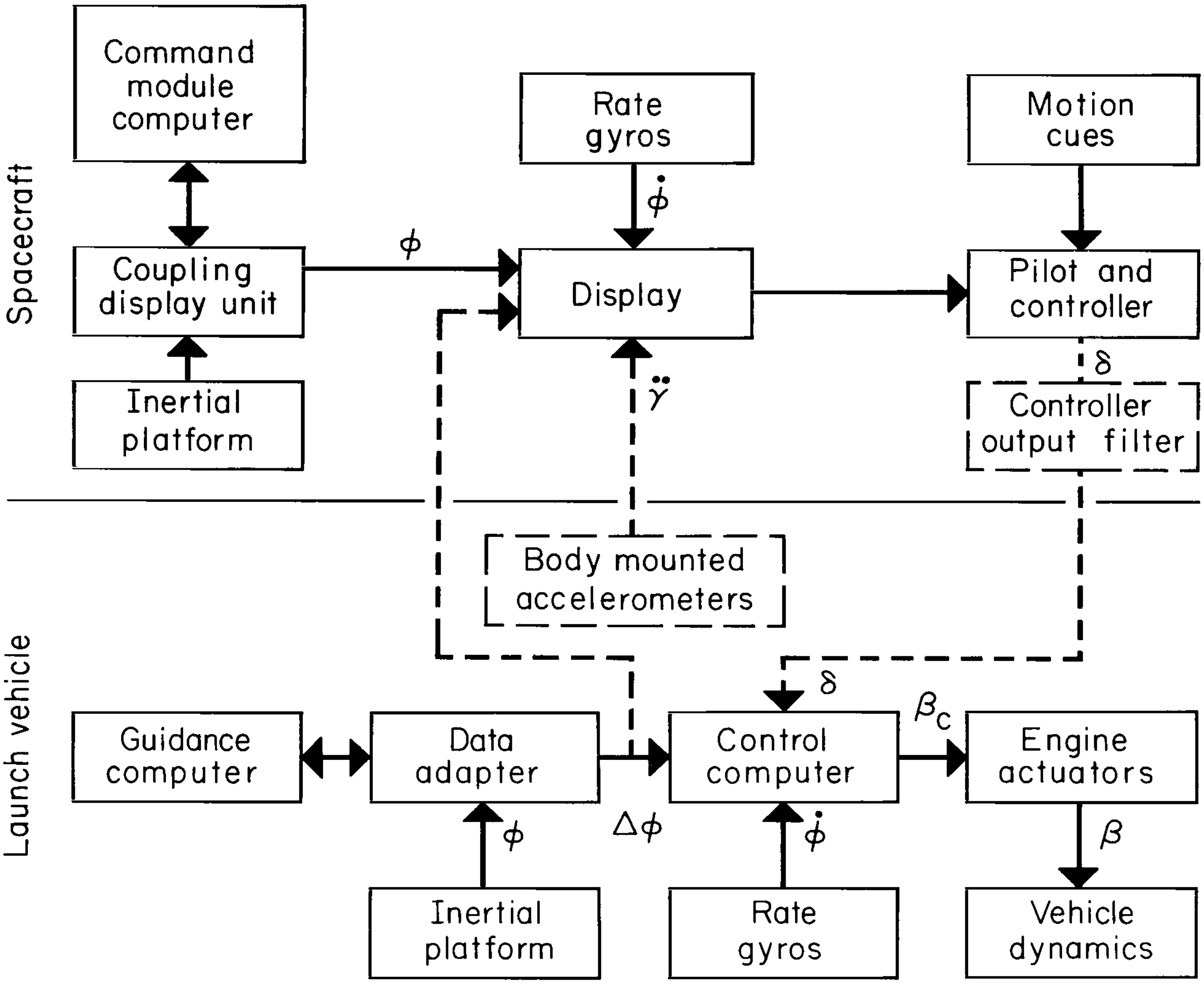


Figure 2.- Typical trajectory.



δ Pilot's controller output
ϕ Attitude
$\Delta\phi$ Attitude error
$\dot{\phi}$ Attitude rate
$\ddot{\gamma}$ Accelerometer output
β Engine gimbal angle

Figure 3.- Saturn V backup guidance and control system.

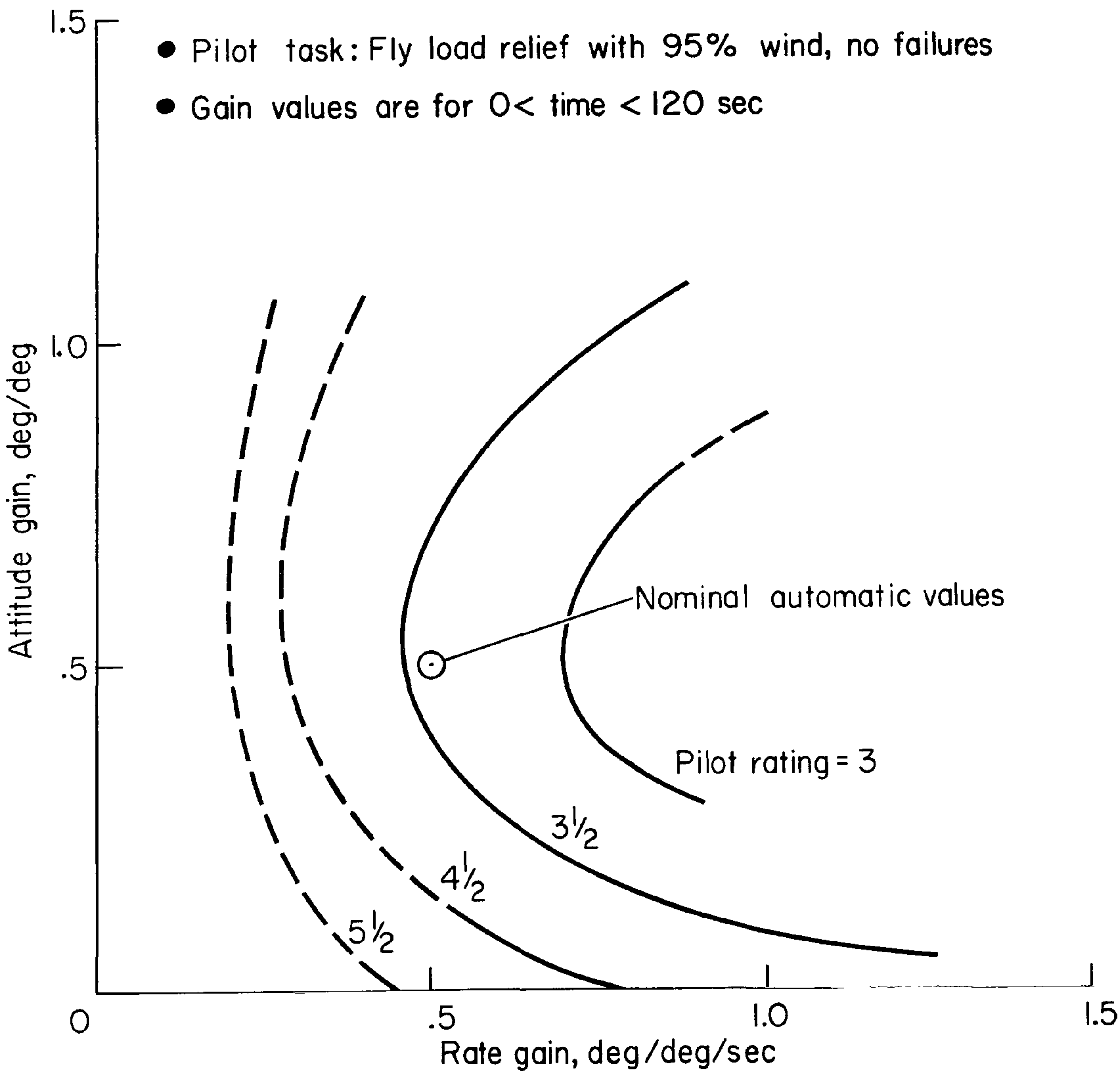


Figure 4.- Pilot opinion rating.

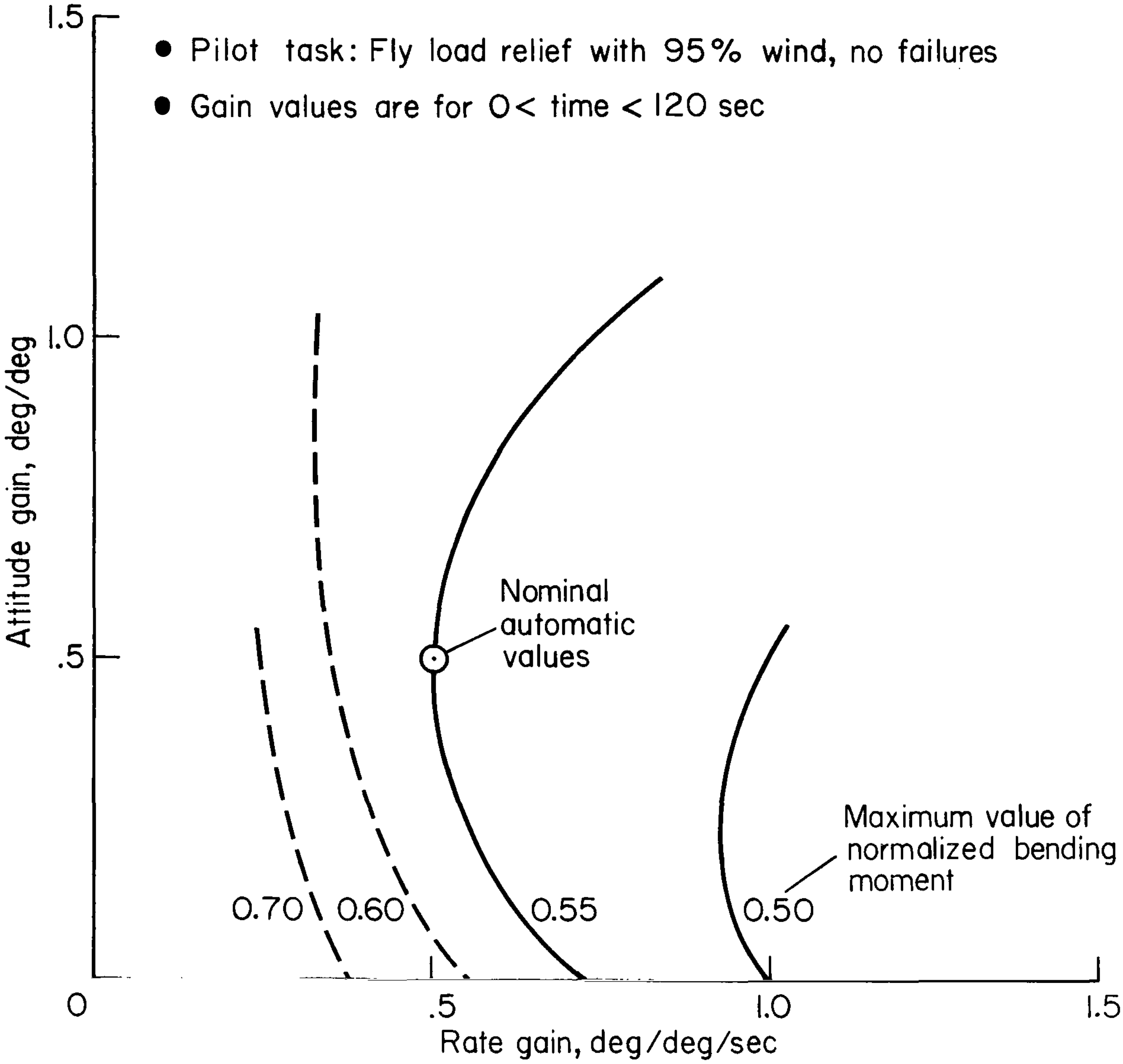


Figure 5.- Structural load performance.

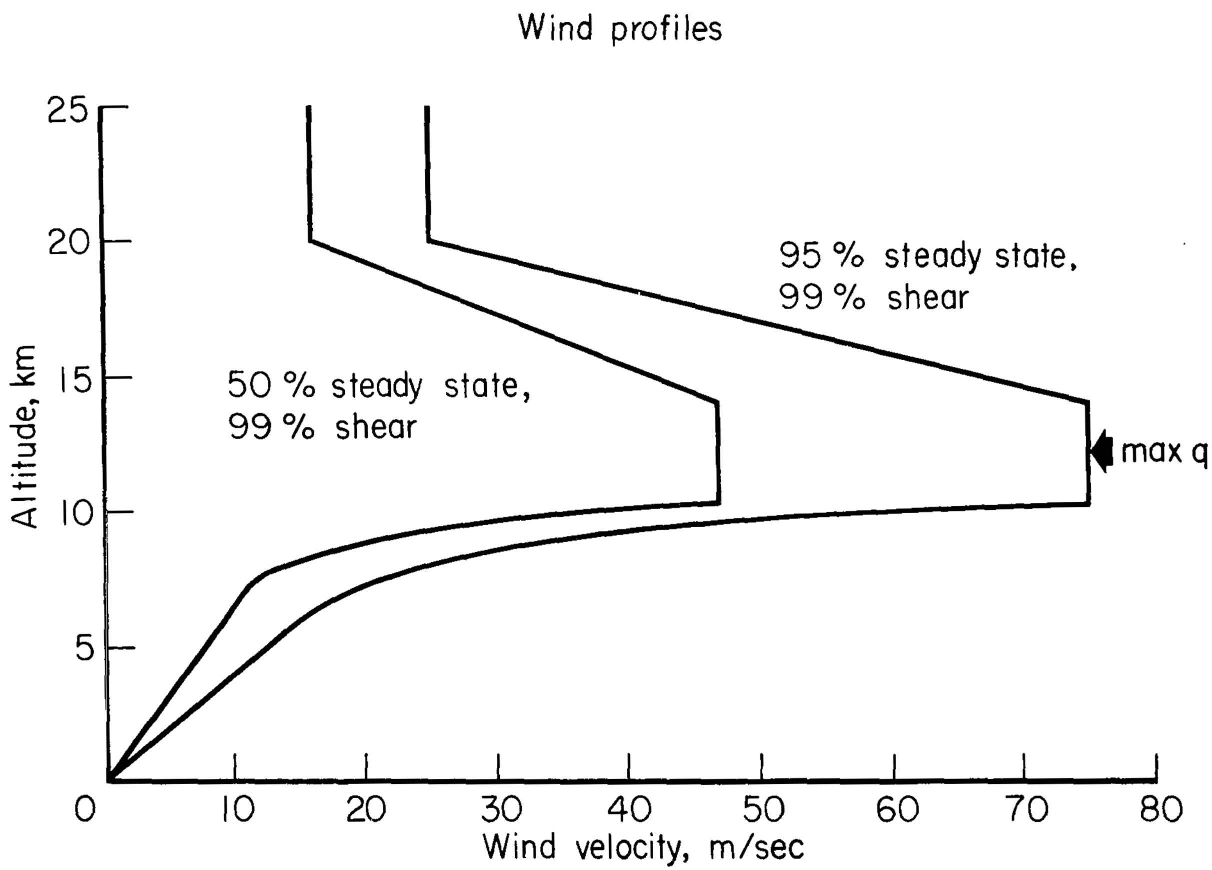


Figure 6.- Wind profiles.

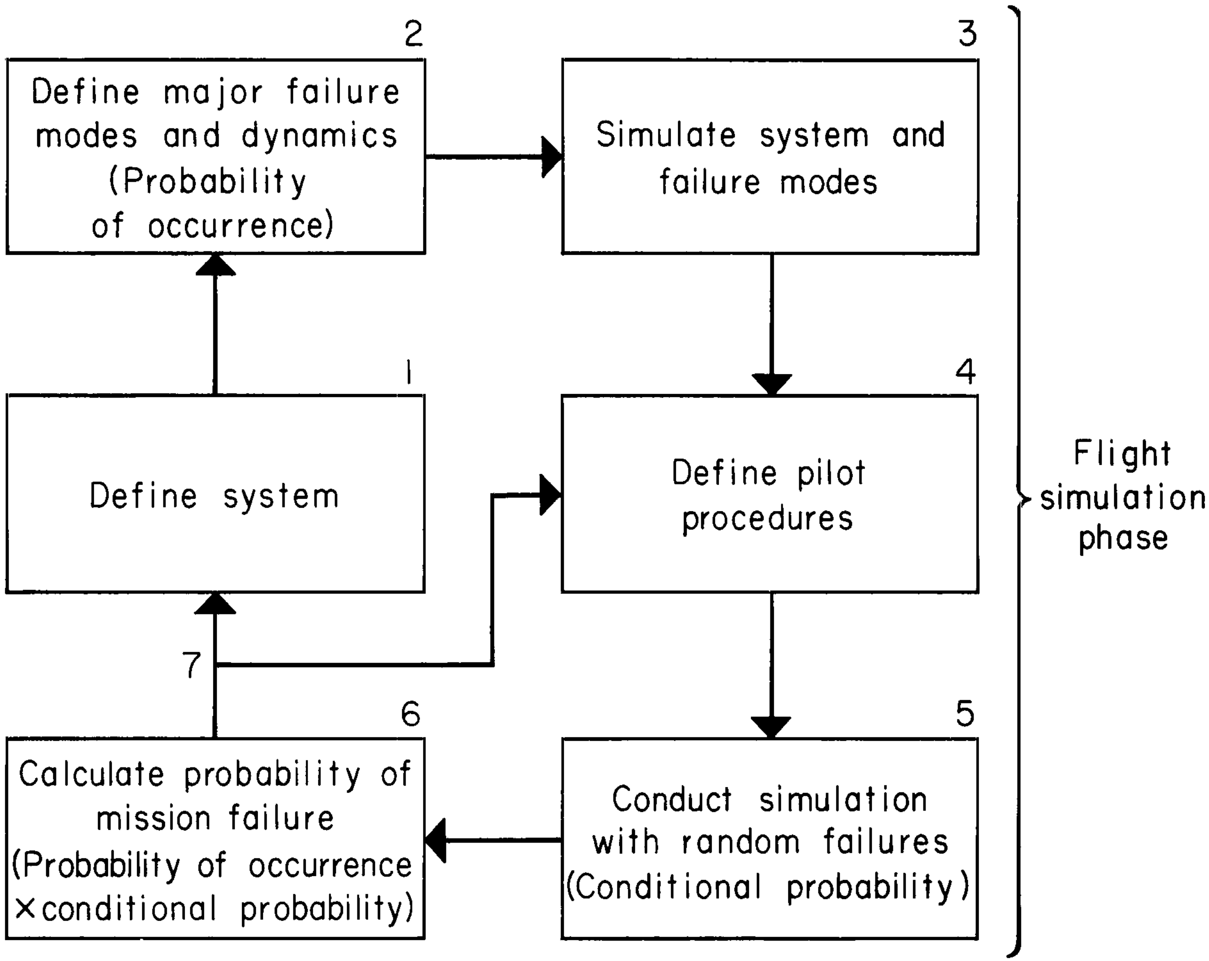


Figure 7.- Technique for measuring probability of mission success.

Number	Failure	Probability of occurrence
1	One actuator hard over	5450×10^{-6}
2	Loss of thrust (One)	5000
3	Two actuators inoperative	3100
4	Loss of platform	1500
5	One actuator oscillatory	1200
6	Loss of attitude rate	300
7	One actuator inoperative	200
8	Loss of attitude signal	*
9	Attitude signal saturate	*
10	Attitude rate saturate	*
11	Attitude display (Lock, jump, drift)	—
12	Attitude error display null	—
13	Attitude error display saturate	—
14	Attitude rate display null	—
15	Attitude rate display saturate	—
16	Accelerometer display null	—
17	Accelerometer display saturate	—
18	Hand controller null	50
19	Hand controller saturate	50

* Less than 20

Figure 8.- List of failures studied and probability of occurrence.

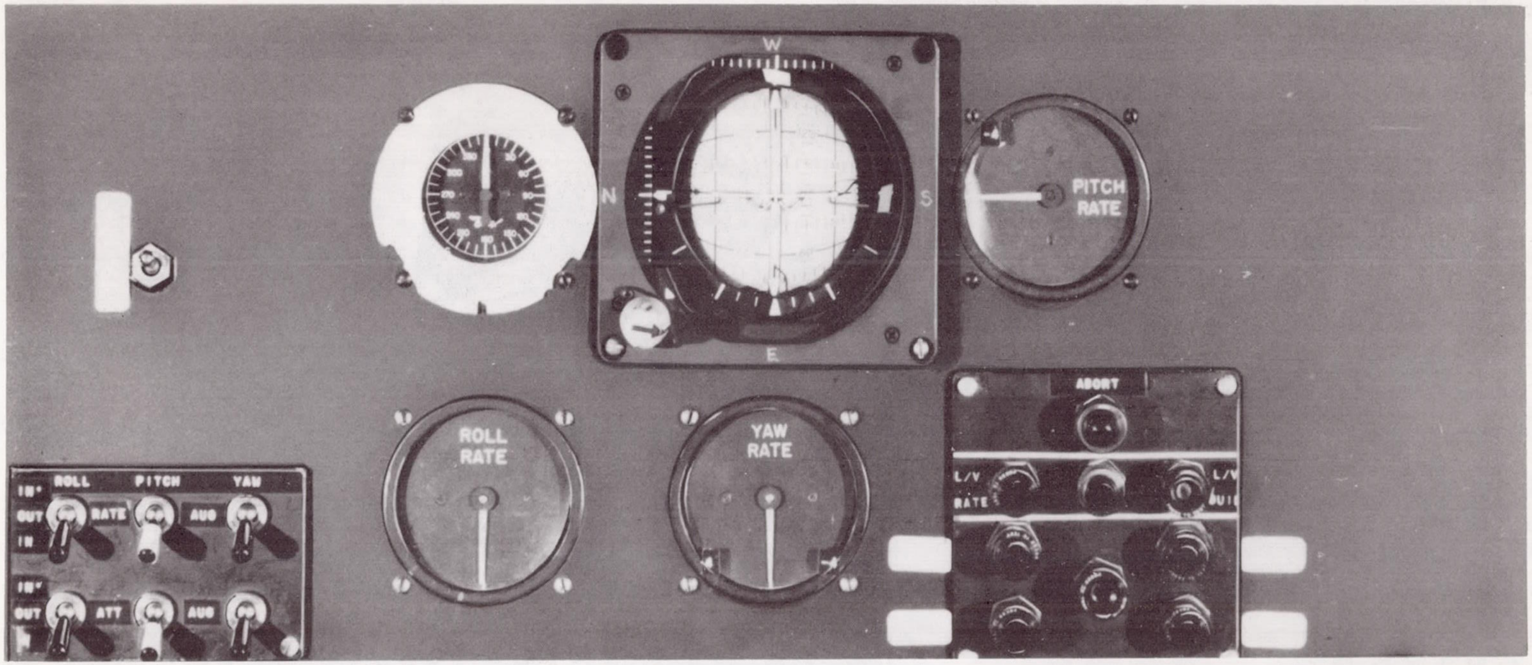


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Figure 9.- Display panel.

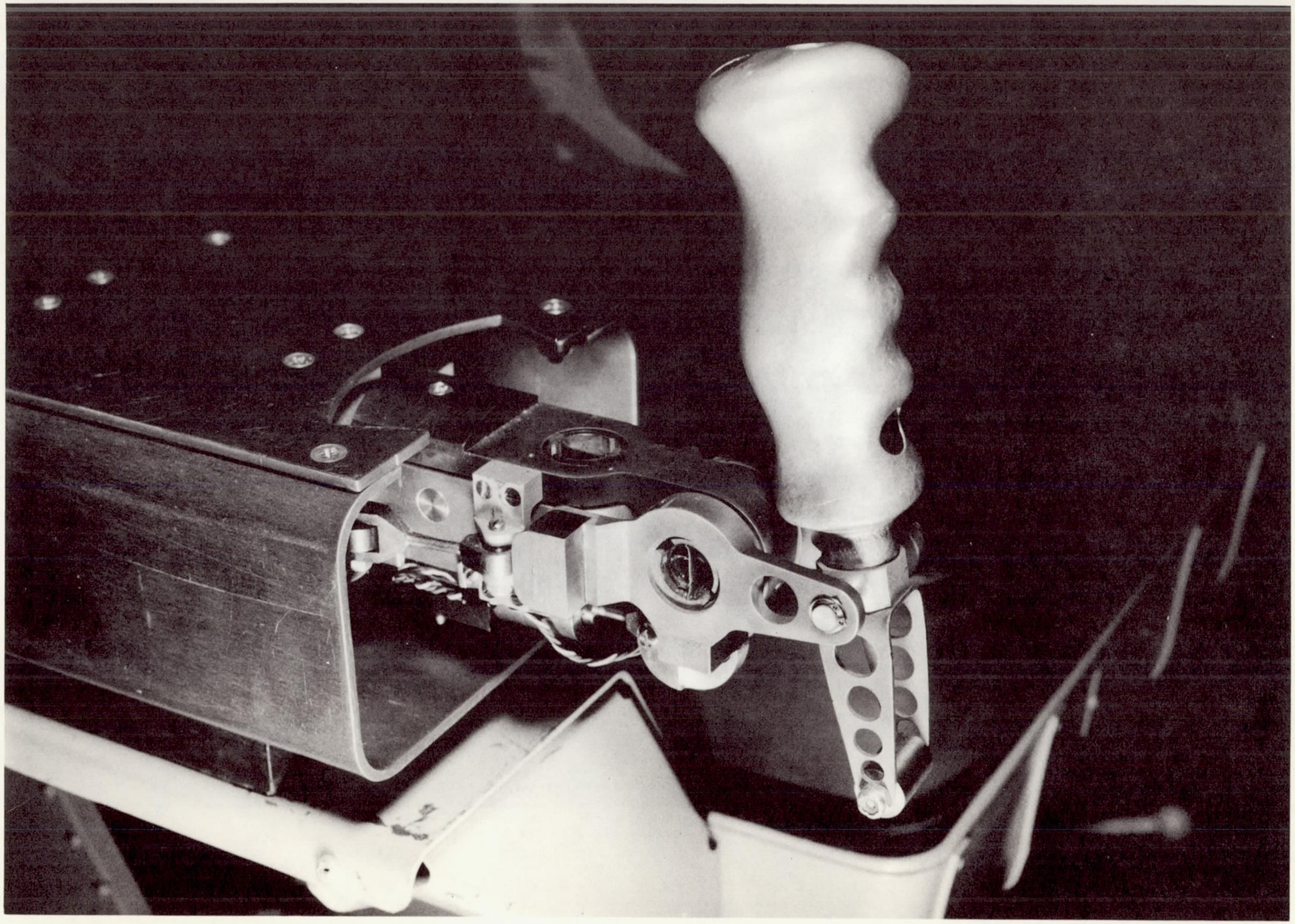

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Figure 10.- Ames' rotational hand controller.

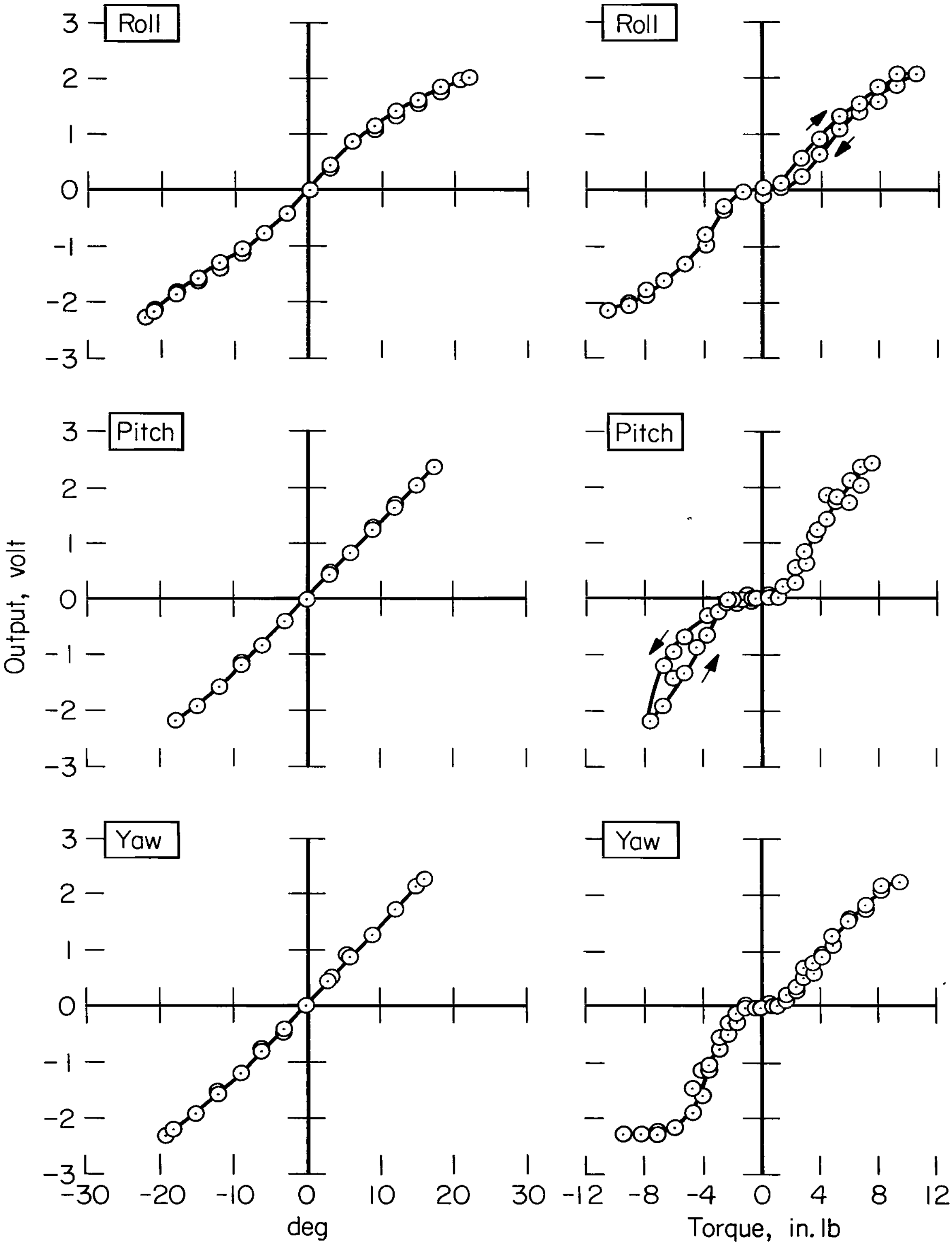

Figure 11.- Ames' hand controller characteristics.

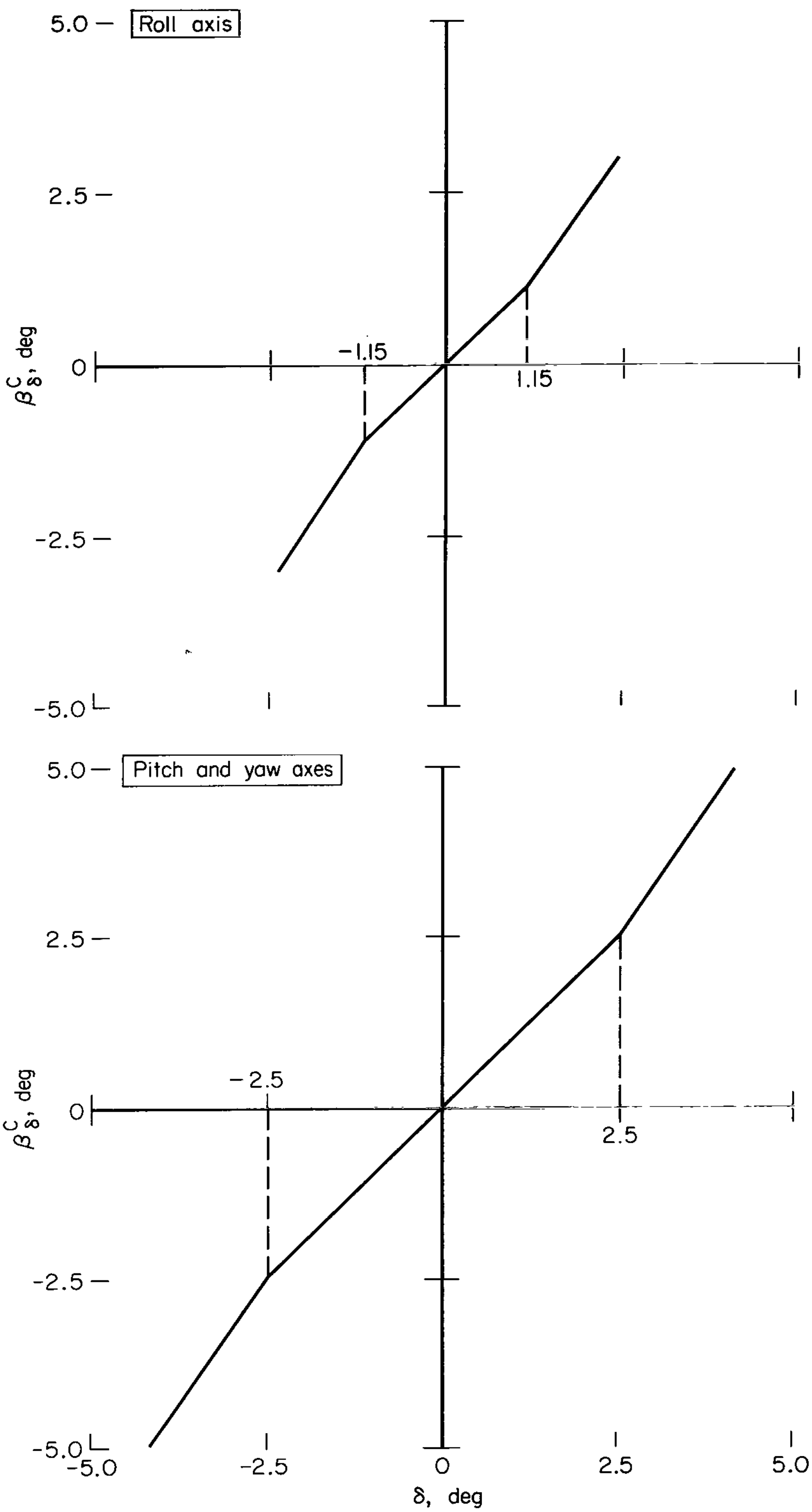

Figure 12.- Nonlinearity used with Ames' hand controller.

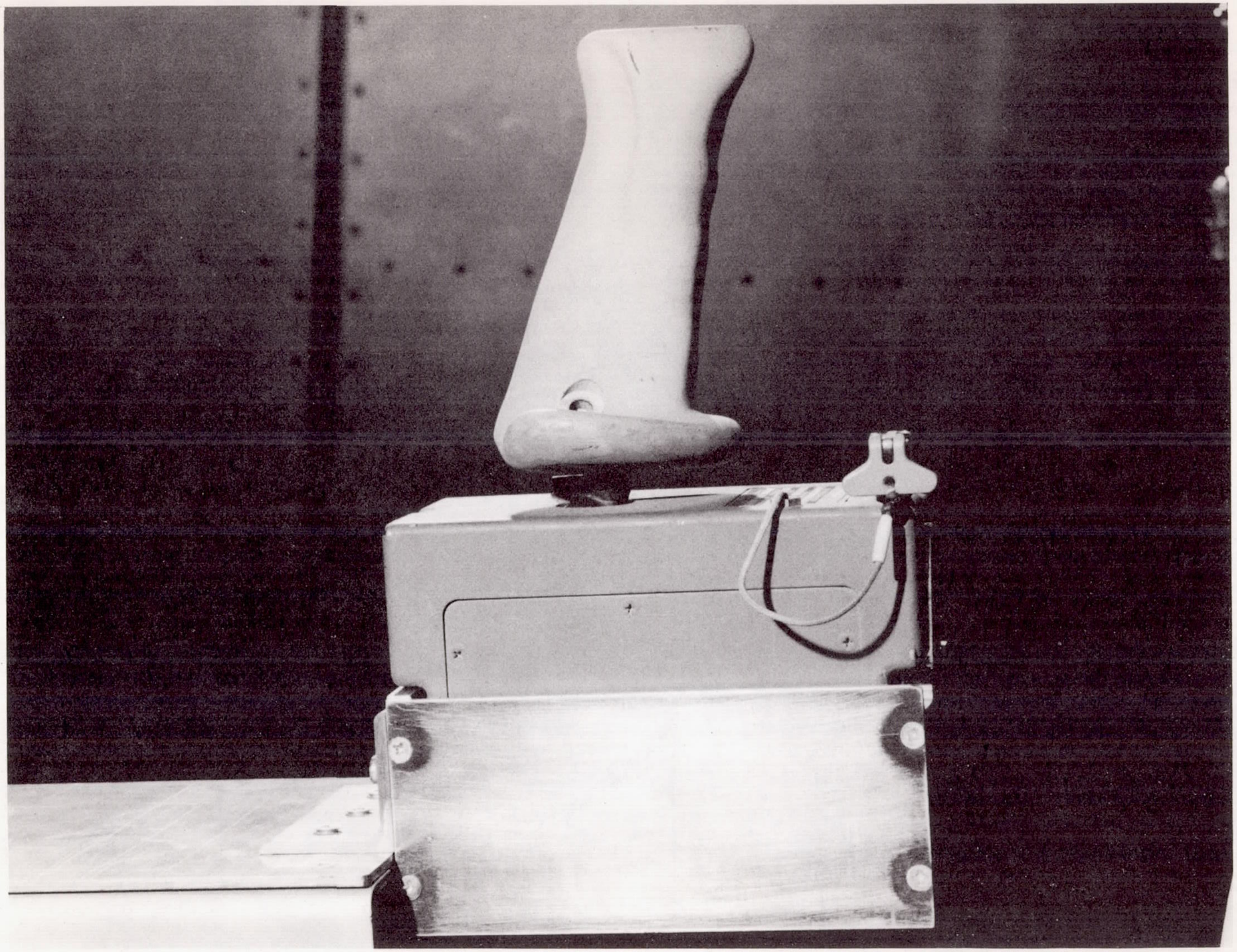

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Figure 13.- Apollo Block I rotational hand controller.

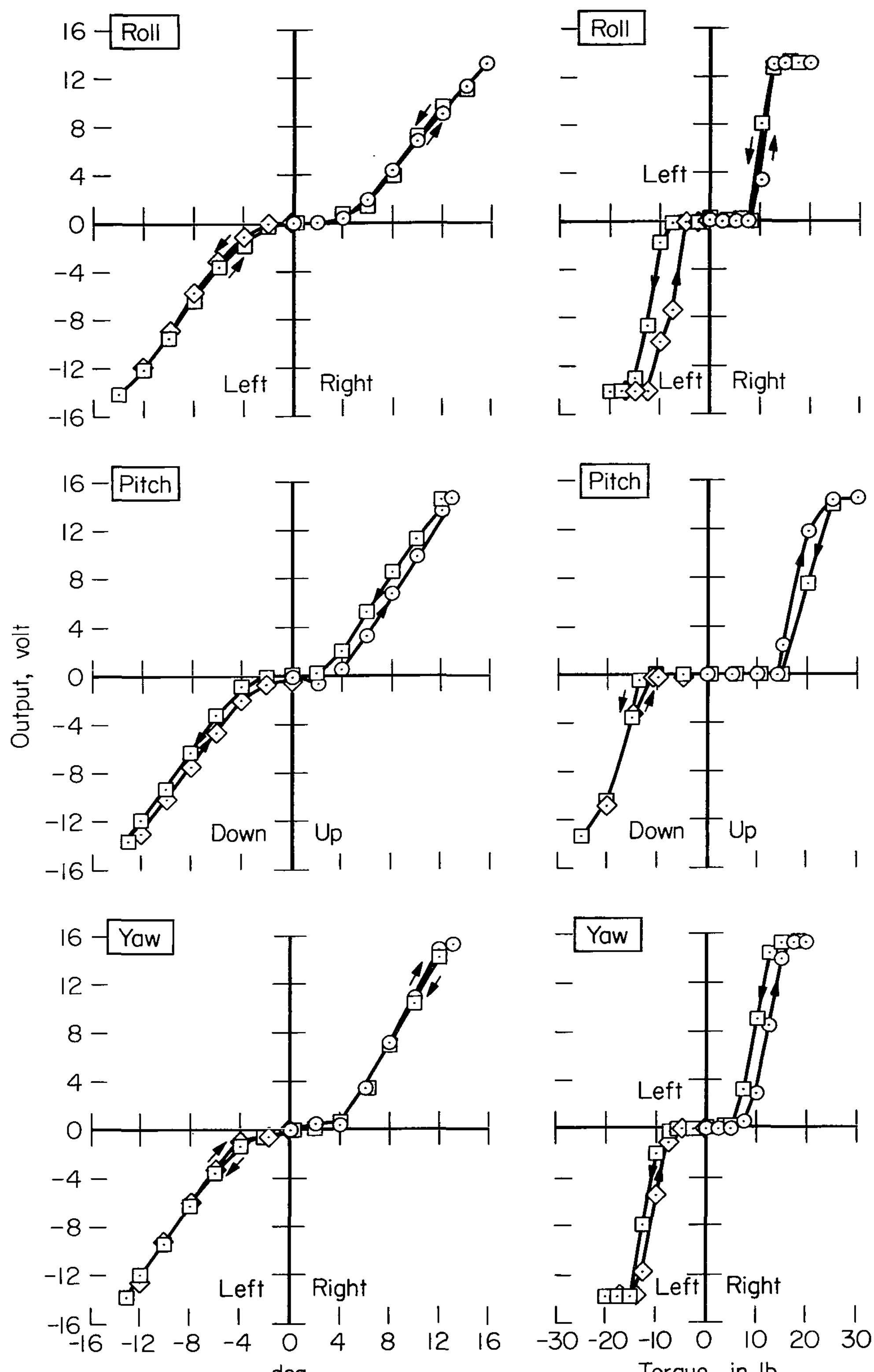
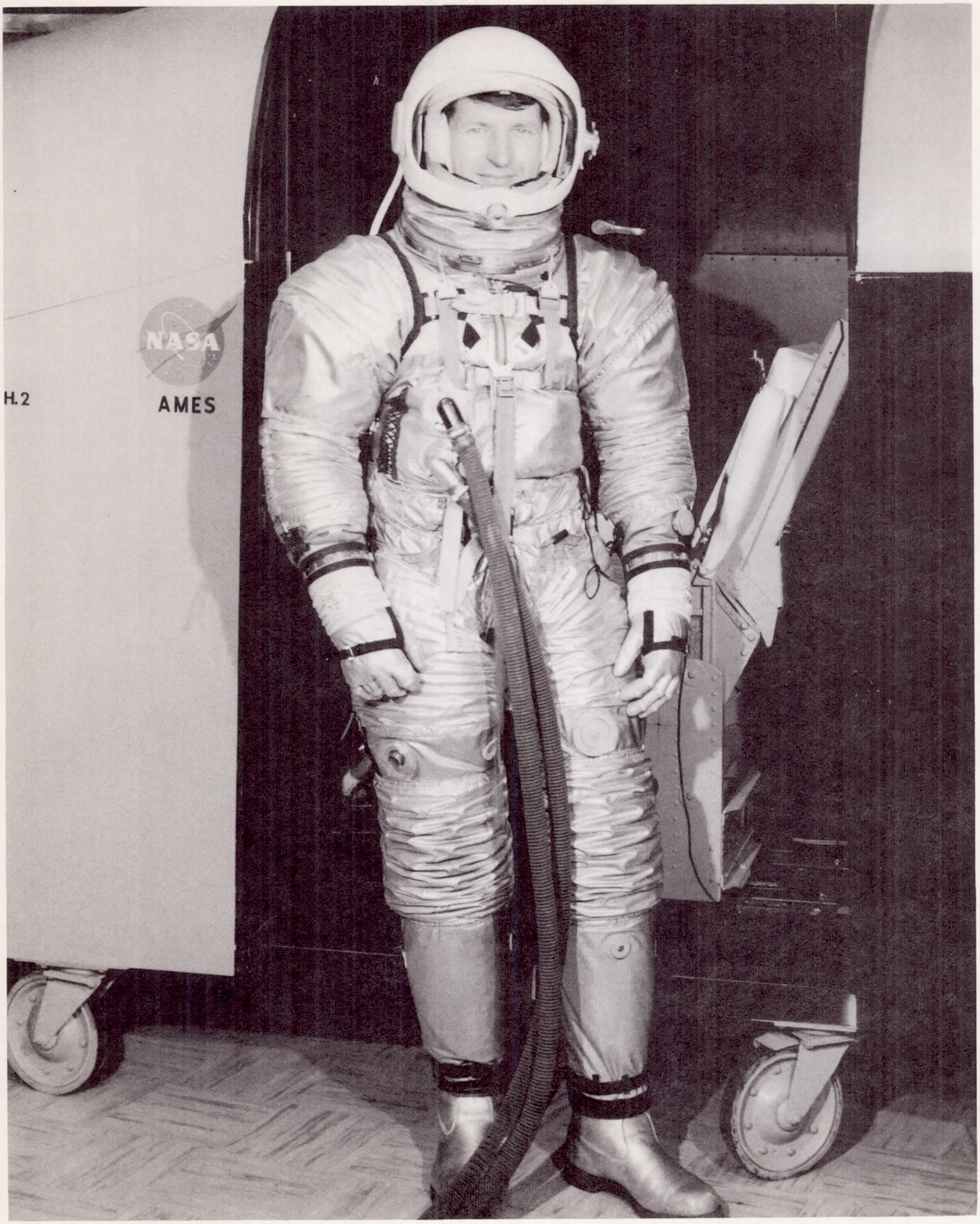
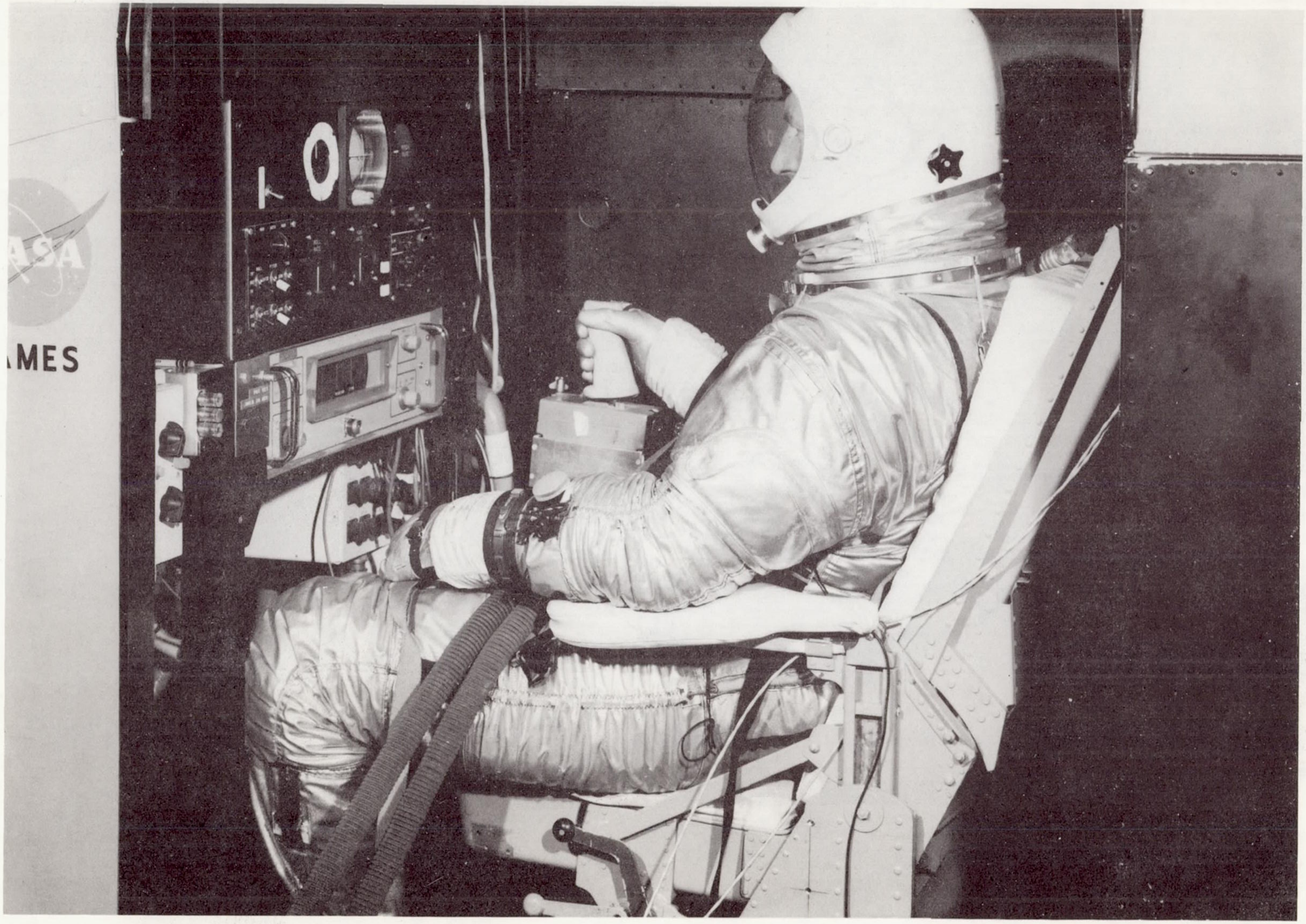


Figure 14.- Apollo Block I hand controller characteristics.



A-37299

Figure 15.- Gemini type pressure suit.



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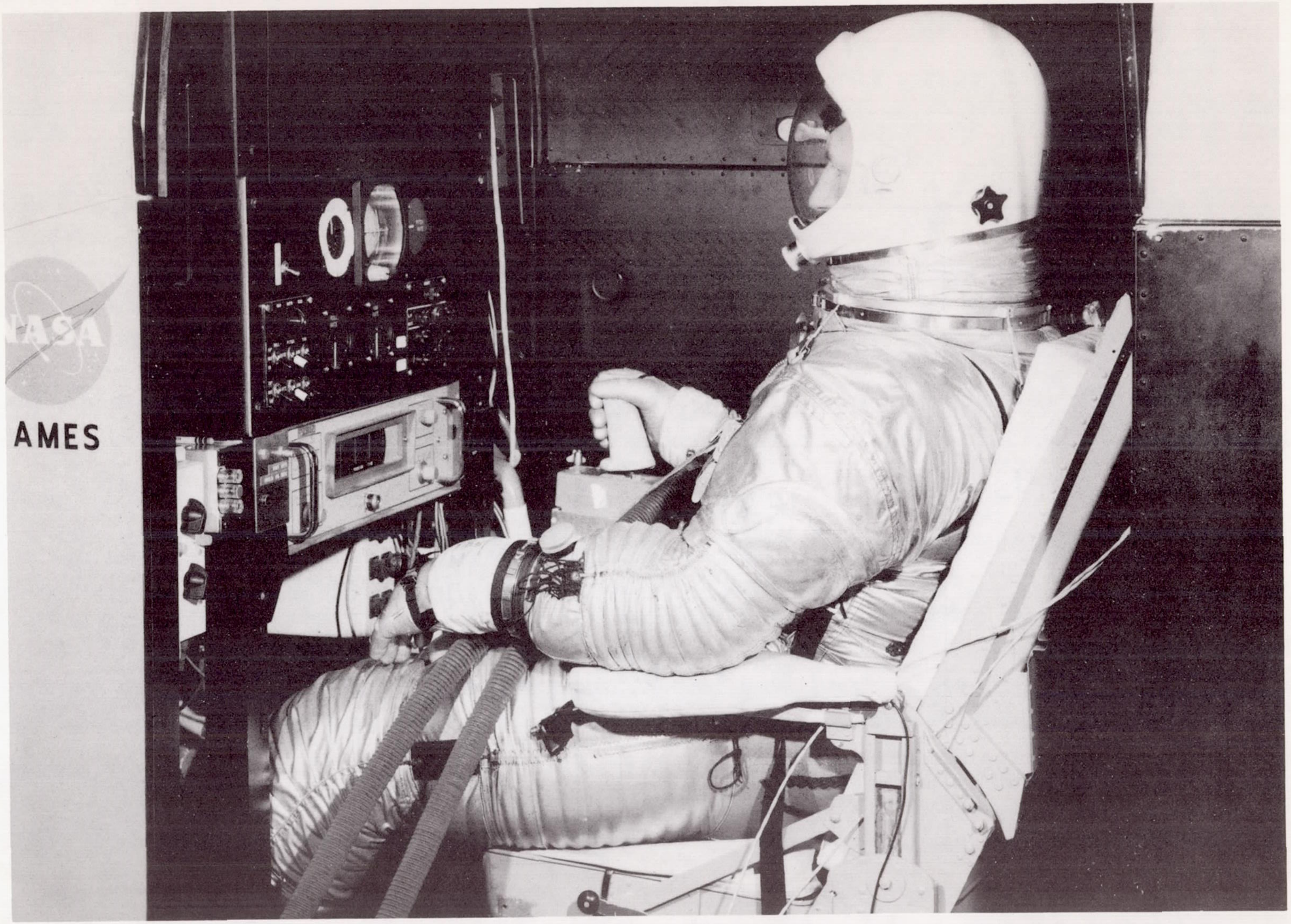
(a) Unpressurized.

Figure 16.- Gemini type pressure suit and Apollo Block I hand controller.



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(b) Pressurized.

Figure 16.- Concluded.

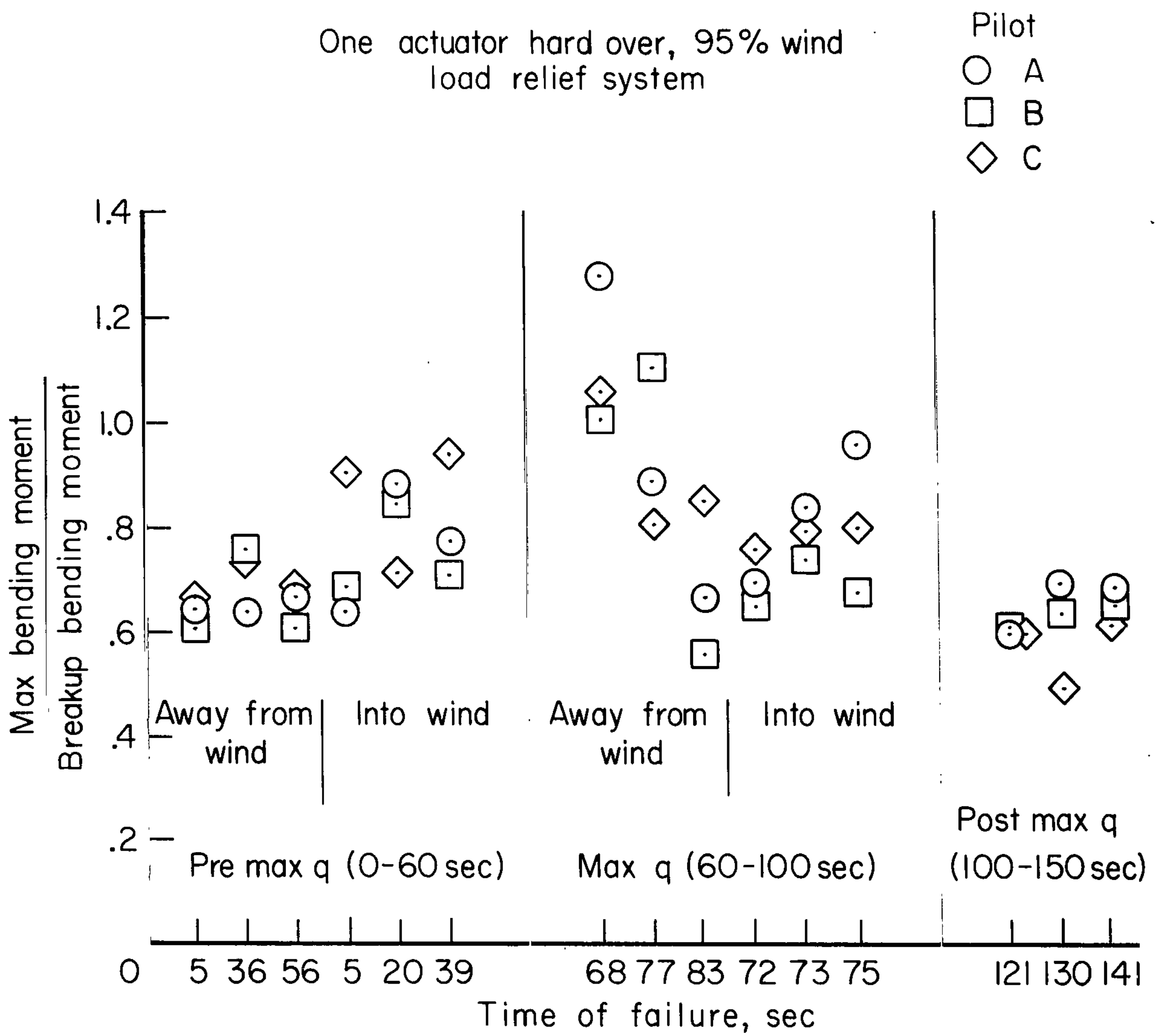

Figure 17.- Typical system failure data using the load relief control system.

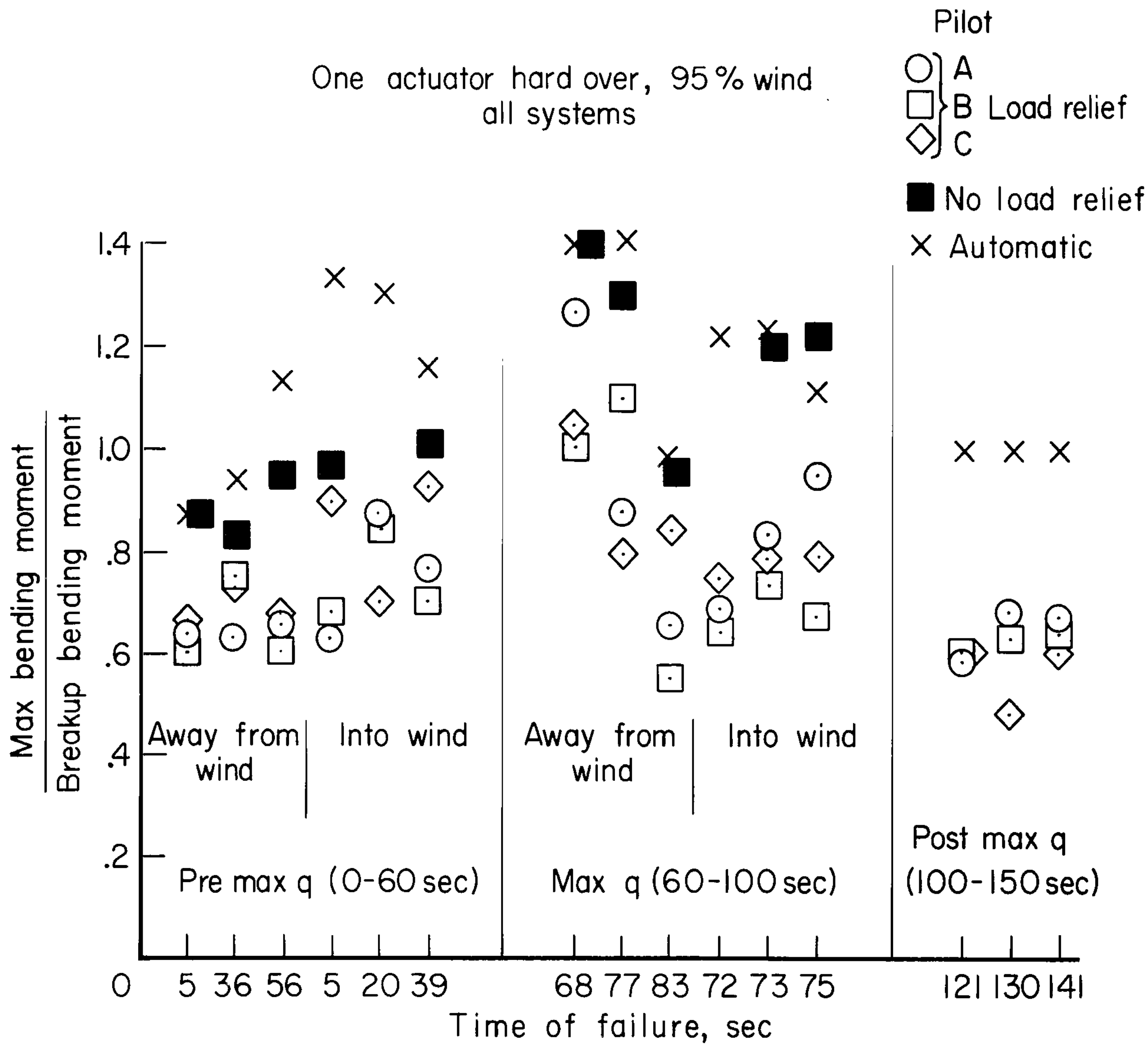

Figure 18.- Typical system failure data for load relief, no lead relief, and automatic control systems.

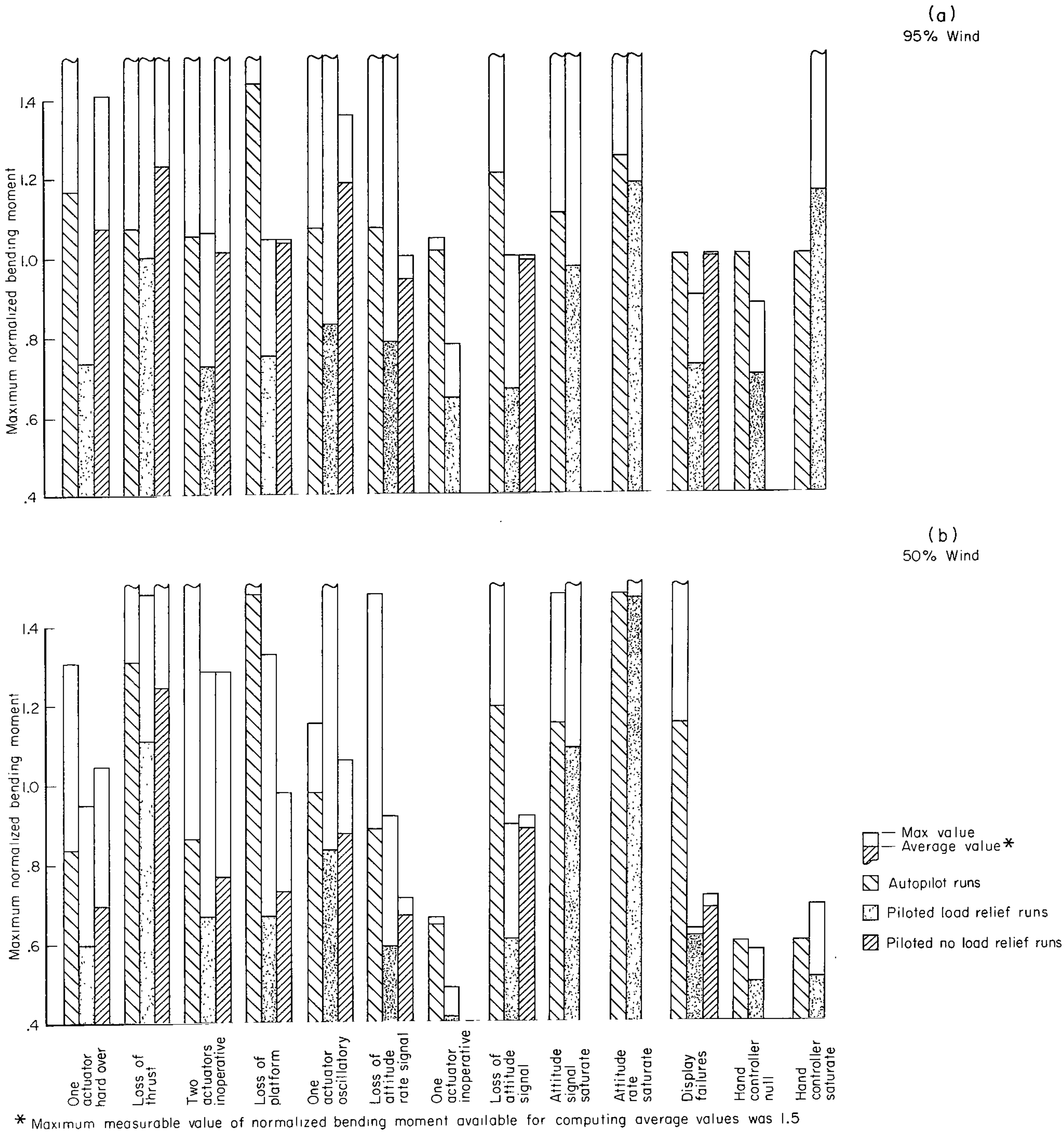

Figure 19.- Structural load data.

No.	Failure	Unreliability No.	System	95% Wind Effectivity	95% Wind Criticality, $\times 10^{-6}$ (1000 2000)	50% Wind Effectivity	50% Wind Criticality, $\times 10^{-6}$ (1000 2000)
1	One actuator hard over	5450×10^{-6}	Load relief No load relief Automatic	0.045 .322 .488		0.0 .045 .155	
2	Loss of thrust (One)	5000		.439 .450 .450		.489 .466 .566	
3	Two actuators inoperative	3100		.011 .084 .392		.039 .133 .225	
4	Loss of platform	1500		.044 .666 .666		.044 .0 .666	
5	One actuator oscillatory	1200		.577 .400 .400		.282 .266 .577	
6	Loss of attitude rate	300		.090 .0 .801		.0 .0 .801	
7	One actuator inoperative	200		.0 .0 .401		.0 .0 .0	
8	Loss of attitude signal	*		.0 .0 .578		.0 .0 .489	
9	Attitude signal saturate	*		.444 .444 .667		.444 .440 .667	
10	Attitude rate saturate	*		.785 .667 1.000		.888 .888 1.000	
11	Attitude display (Lock, jump, drift)	—		.0 .0 N.A.		.0 .0 N.A.	
12	Attitude error display null	—		.0 .0 N.A.		.0 .0 N.A.	
13	Attitude error display saturate	—		.0 .0 N.A.		.0 .0 N.A.	
14	Attitude rate display null	—		.0 .0 N.A.		.0 .0 N.A.	
15	Attitude rate display saturate	—		.0 .0 N.A.		.0 .0 N.A.	
16	Accelerometer display null	—		.0 .0 N.A.		.0 .0 N.A.	
17	Accelerometer display saturate	—		.0 .0 N.A.		.0 .0 N.A.	
18	Hand controller null	50		.0 .0 N.A.		.0 .0 N.A.	
19	Hand controller saturate	50		.440 .0 N.A.		.178 .0 N.A.	

Total		95% Wind	50% Wind
	Load relief	3300×10^{-6}	2990×10^{-6}
	No load relief	5720	3320
	Automatic	7940	6320

* Less than 20
N.A. Not applicable

Figure 20.- Saturn IC criticality study; summary of results.

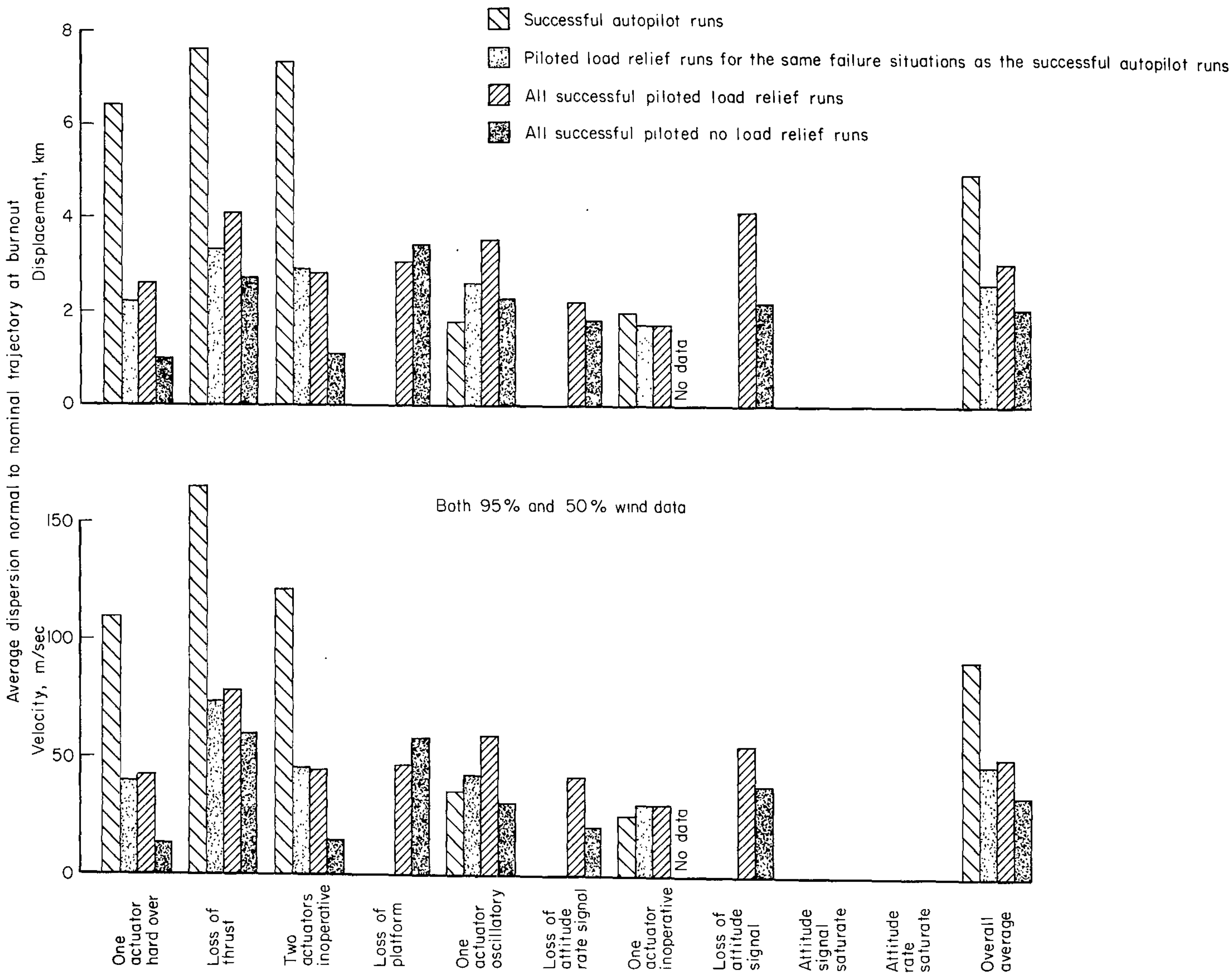

Figure 21.- Summary of trajectory dispersion data.

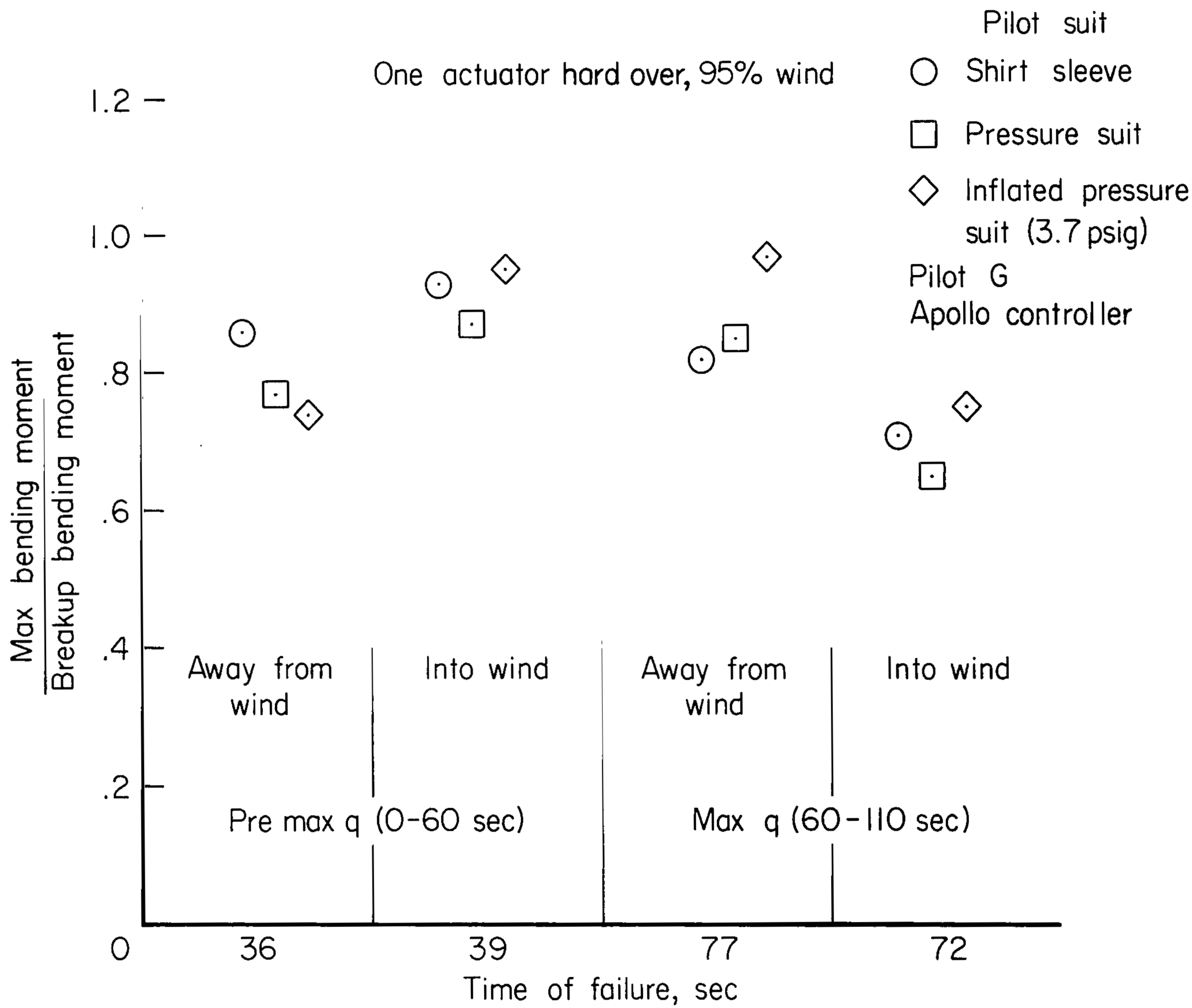

Figure 22.- Effect of pressure suit on performance.

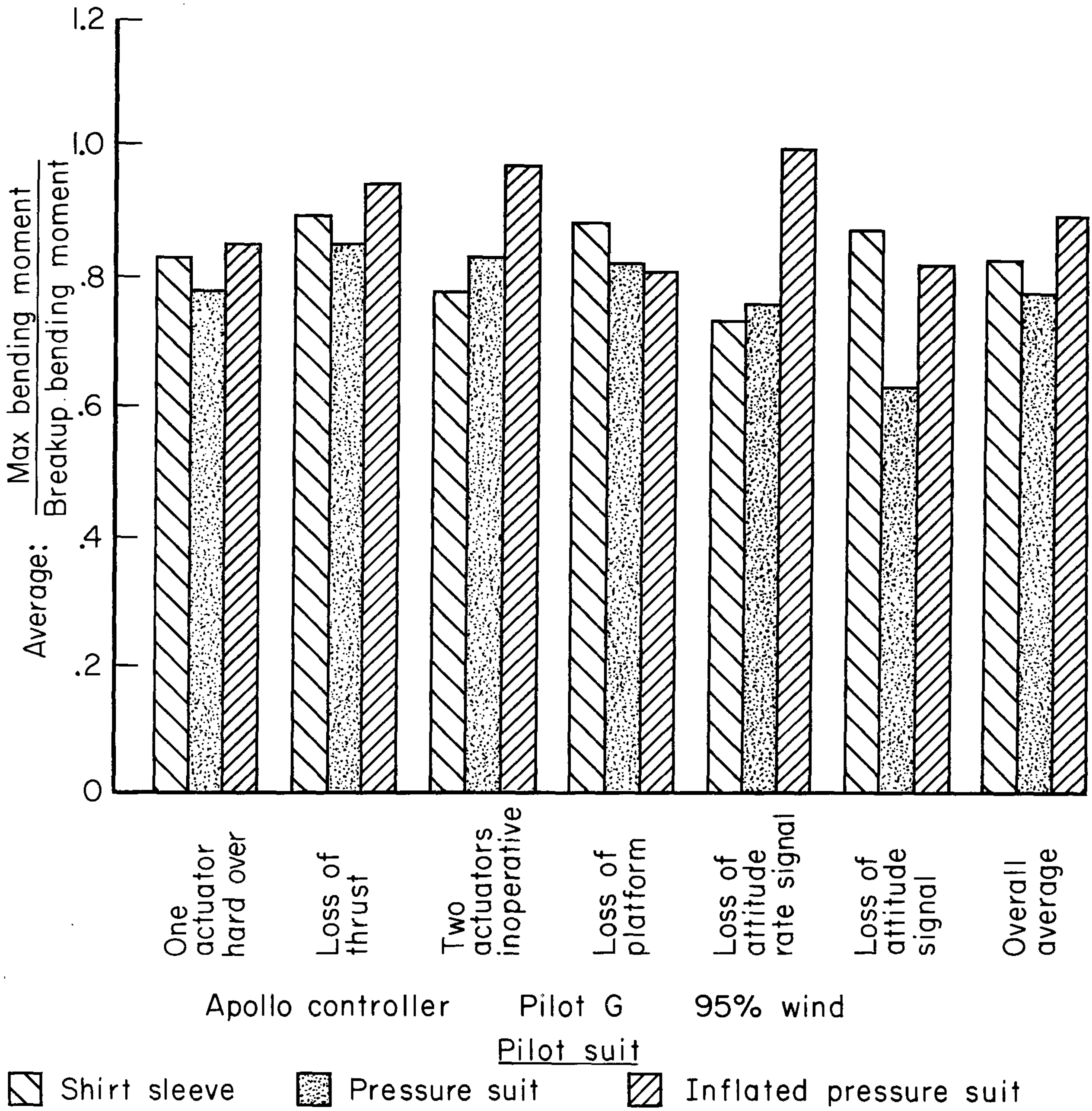

Figure 23. - Summary of effect of pressure suit.

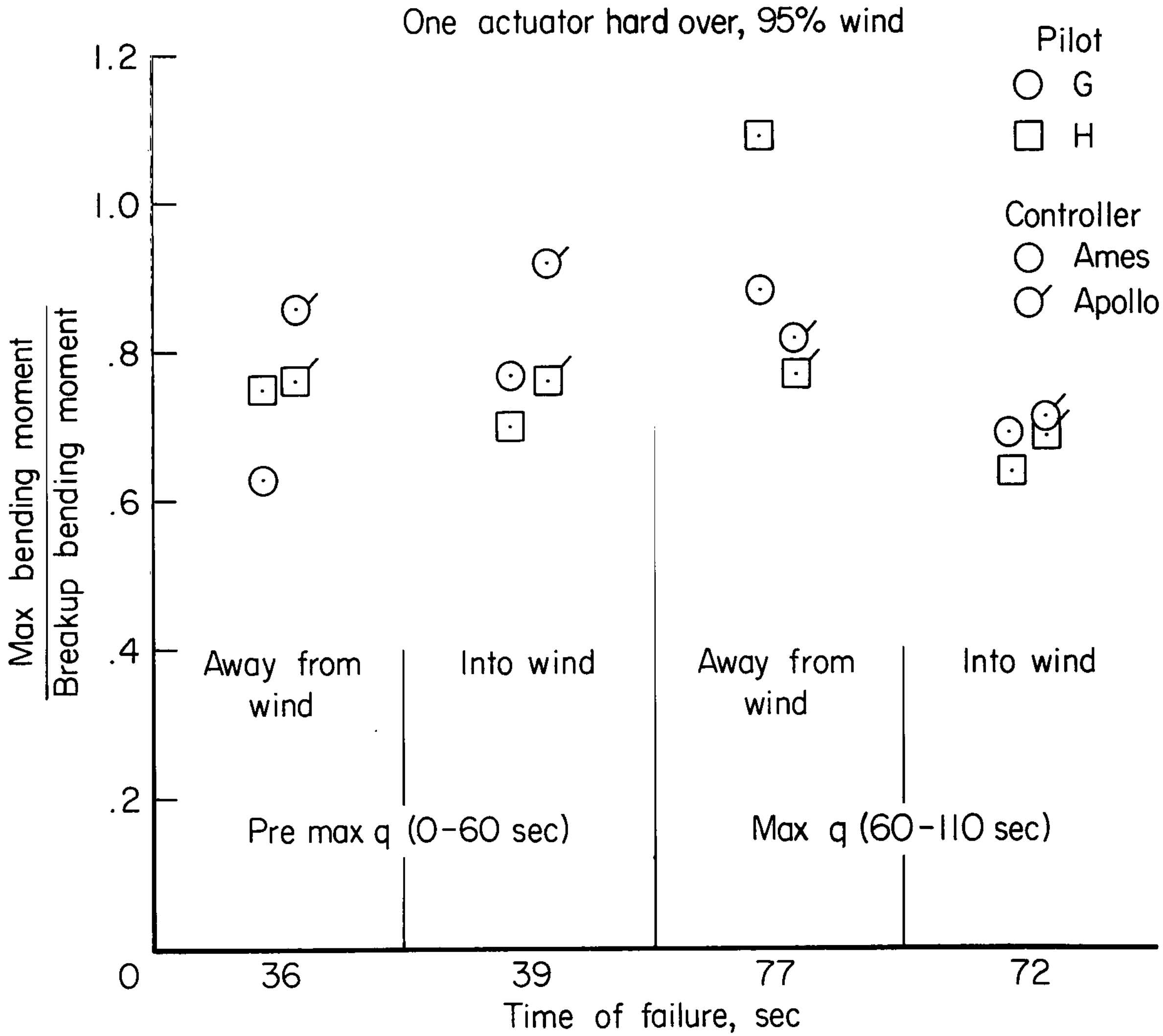

Figure 24.- Comparison of the Ames and Apollo Block I hand controllers.

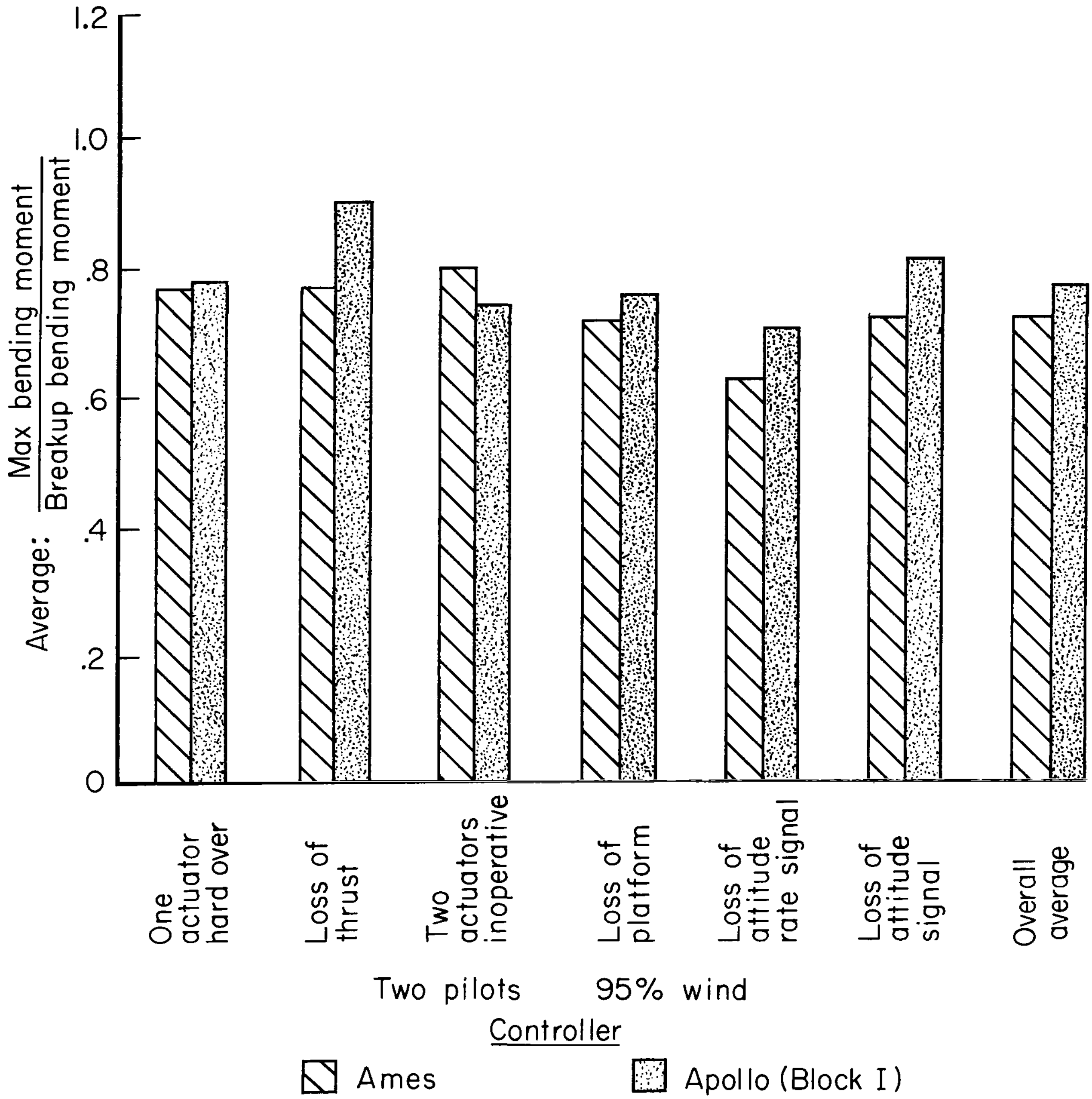

Figure 25.- Summary of hand controller comparison.

NASA-Langley, 1969 —— 21 A-3067

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