N63-11123
code 1

50p

NASA TN D-1402

NASA TN D-1402

TECHNICAL NOTE

D-1402

OPERATIONAL EXPERIENCES AND CHARACTERISTICS OF THE X-15 FLIGHT CONTROL SYSTEM

By Robert A. Tremant

Flight Research Center
Edwards, Calif.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

WASHINGTON December 1962

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

TECHNICAL NOTE D-1402

OPERATIONAL EXPERIENCES AND CHARACTERISTICS OF THE X-15 FLIGHT CONTROL SYSTEM

By Robert A. Tremant

SUMMARY

The first 45 flights of the X-15 airplane demonstrated the adequacy of the manual aerodynamic control system and the stability augmentation system to meet the operational control requirements for flight to a Mach number of approximately 6.04 and an altitude of about 217,000 feet. The airplane was lightly damped in all axes with dampers off and encountered regions of lateral-directional uncontrollability. Although the pilots considered the control system to be adequate for the X-15 design flight envelope, some improvement in the system was necessary for extension beyond this envelope.

A limit cycle, or residual oscillation, primarily in roll was observed at flight conditions of high dynamic pressure with high stability-augmentation-system gains. Also, structural frequencies of the airplane control surfaces were excited in flight, and the vibrations were sustained by the stability augmentation system with phase-lead shaping. These phenomena dictated that corrective modifications be made to the flight control system to provide the necessary safety and reliability for flight.

Malfunctions of the stability augmentation system affected 25 percent of the X-15 free flights. Seventy percent of these malfunctions were the result of human error. Although the overall flight reliability of the stability-augmentation-system components does not meet specifications at this time, current trends indicate that the system will approach adequate reliability during the flight period now in progress.

INTRODUCTION

Early in the design of the X-15 research airplane, it became obvious that stability augmentation would be required over much of the flight envelope. A cooperative design team of personnel from North American Aviation, Inc., the U. S. Air Force, the U. S. Navy, and the National Aeronautics and Space Administration specified, therefore, that the X-15 flight control system would include a stability augmentation system for all three axes, with emphasis on simplicity, reliability, and versatility for research purposes.

North American Aviation, which was directly responsible for the development of the airplane and its systems, conducted many of the design tests reported herein. Flight tests were conducted by the NASA Flight Research Center at Edwards, Calif. This paper presents the characteristics of the basic flight control system and the stability augmentation system and discusses the operational performance and reliability of the flight control system during the period from December 1, 1958, to January 1, 1962. The data were obtained in 30 flights with the interim LR11 engines and 15 flights with the XLR99 engine, and extend to a Mach number of about 6.04 and an altitude of approximately 217,000 feet. Simulator data cover the entire design flight envelope of the X-15, that is, to a Mach number of 6 and an altitude of 250,000 feet.

A brief résumé of the flight history of the X-15 stability augmentation system is presented in the appendix.

SYMBOLS

d	duty cycle (mission duration), hr
F_c	lateral stick force, lb
F_r	rudder pedal force, lb
F_s	longitudinal stick force, lb
f	frequency, cycles/sec
h	radar altitude, ft
K_p	roll-damper gain-selector-switch position
K_q	pitch-damper gain-selector-switch position
M	Mach number
m	mean time to failure, hr
p	rolling velocity, deg/sec or radians/sec
q	pitching velocity, deg/sec
$\bar{q}$	dynamic pressure, lb/sq ft
R	reliability, $e^{-d\left(\frac{1}{m_1} + \frac{1}{m_2} + \dots + \frac{1}{m_n}\right)}$, percent
r	yawing velocity, deg/sec
t	time, sec

α	angle of attack, deg
β	angle of sideslip, deg
Δ	incremental value
δ_a	total aileron angle, $\left(\delta_{h_L} - \delta_{h_R}\right)$, deg
δ_c	center-stick deflection, in.
δ_h	horizontal-stabilizer angle, $\frac{\left(\delta_{h_L} + \delta_{h_R}\right)}{2}$, deg
δ_r	rudder-pedal displacement, in.
δ_s	side-stick deflection, in.
δ_v	vertical-tail deflection, deg
φ	phase angle, deg

Subscripts:

a	aileron
h	horizontal
L	left
max	maximum
R	right
u	upper

X-15 AIRPLANE

General Description

The X-15 is a single-place rocket-powered research vehicle designed for flight at hypersonic speed and extreme altitude. A three-view drawing of the airplane is shown in figure 1. The physical characteristics and detailed design information are included in reference 1.

Control is provided through conventional aerodynamic surfaces, except that the horizontal tail provides both pitch and roll control. Yaw control is provided by upper and lower vertical surfaces. The movable portion of the lower vertical surface is jettisonable for landing ground-clearance. All aerodynamic control surfaces are actuated by two independent irreversible hydraulic systems.

The pilot's aerodynamic controls consist of the conventional center stick and rudder pedals, and a side stick located on the pilot's right (fig. 2). In addition, a controller positioned on the pilot's left activates the nonaerodynamic reaction control rockets.

The X-15 aircraft utilized in the early investigations were powered by two LR11 rocket engines which produced a sea-level thrust of 13,000 pounds. Later flights were made with the XLR99 rocket engine which produces 57,000 pounds of thrust at sea level.

Basic Aerodynamic Control System

Although the basic X-15 control system has several unusual features, the design concepts are conventional. Figure 3 shows the basic pitch and roll control system as an irreversible hydraulic system with artificial feel. The mechanical linkages, which are mass balanced, couple the control sticks and rudder pedals to the hydraulic actuator-control valves.

The horizontal stabilizers are used for both pitch and roll control, deflecting conventionally for pitch control and differentially for roll control. The pitch and roll linkages are integral and are designed to transmit both control modes simultaneously by mechanical summing in the left and right mixer mechanisms located in the cockpit area. The basic-control-system force and displacement characteristics are presented in figures 4(a) to 4(n). The longitudinal-control mode employs a nonlinear gearing to give a lower-surface-to-stick displacement through neutral to minimize sensitivity, overcontrol, and pilot-induced oscillations. The basic control system employs cable-tension regulators which are necessary because of changes in cable length as a result of variation in temperature during flight.

The side-located stick is provided for use during periods of high acceleration. This stick is located forward of the right-hand seat armrest and is mechanically linked to the center stick through two dualized hydraulic boost actuators; thus, the side stick tracks the center stick at all times. The boost actuators reduce the side-stick pilot-control forces and synchronize the side- and center-stick displacements. The current X-15 configuration has boost (force) ratios of 4:1 in pitch and 2.7:1 in roll. The side-stick neutral position is adjustable longitudinally through a range of ±1 inch. Both sticks have a common feel-force bungee and have longitudinal trim capability, which is achieved by shifting the zero-force position of the feel bungee to a stick position corresponding to the desired horizontal-stabilizer position.

The pilot has control authority over the range from 15° to -35° stabilizer deflection in pitch, ±15° differential-stabilizer deflection in roll, and ±7.5° vertical-stabilizer deflection in yaw. The pilot-controlled longitudinal trim range is from 5° to -20° horizontal-stabilizer deflection. Roll and yaw trim range is ±2° and is ground-adjustable only.

Stability Augmentation System

The major components of the stability augmentation system (SAS) are: three rate gyros, two pitch-roll servocylinders, one yaw servocylinder, electronic-case assembly (ECA), gain-selector-switch assembly and function-switch assembly (GSSA). Figure 5 shows the relative location of the components within the airplane, and figure 6 is a functional block diagram of the SAS. Basically, the system consists of an electronic network or channel for each axis. This network senses the aircraft rate of change of pitch, roll, and yaw and automatically provides signals to the respective servocylinders that cause the surface actuators to move the horizontal and vertical stabilizers to oppose the airplane angular rates. Individual servocylinder outputs and the pilot's manual inputs are combined to form a single input to the surface actuators. The pitch and roll channels operate singly or in combination at the pilot's discretion. Since the horizontal stabilizers are used for both pitch and roll control, the left and right servocylinders control the stabilizers for both pitch and roll damping. The yaw channel operates independently of the pitch and roll channels. In addition, a signal from the yaw gyro proportional to yaw rate is fed into the roll channel. This is termed the "yar" channel. Therefore, the left and right servocylinder outputs at any given time are an algebraic sum of pitch, roll, and yaw signals, when all channels are operating. Yar-damper off does not affect roll or yaw damping, but roll-damper off makes the yar damper inoperative. The yaw-servocylinder output is a result of yaw-rate input only.

The authority of the stability augmentation system is equal to the pilot's authority in pitch and yaw and to twice the pilot's authority in roll. Figure 7 shows the pilot and damper command envelope for the horizontal stabilizers which provide pitch and roll control and damping. The pilot has on-off and feedback gain control of the SAS, which enables him to vary the gains throughout the flight envelope. Table I lists the stability-augmentation-system gain settings in terms of servocylinder stroke and surface deflection.

To provide fail-safety, the SAS contains dual channels in all modes. The working channel drives the servocylinders. The monitor channel operates electronically simulated servocylinders, and compares the outputs to those of the working channel (see fig. 6). When the difference between the servocylinder position and the simulated servocylinder position exceeds 10 percent (0.10 in.) in any channel, a failure is signaled and the servocylinder centers and locks, disengaging the SAS. Differences can occur because of electrical or mechanical malfunction. A warning light for each channel is provided to indicate to the pilot that the system is in "standby" (steady light) or that a failure (blinking light) has occurred. The pilot may reset each channel by switching the function switches to "standby" and returning them to the "engaged" position. If the malfunction no longer exists, the failure light remains out and the channel is engaged. Electrical power and hydraulics are monitored, and the stability augmentation system is disengaged when either or both fall below preset operational limits.

In the pitch and roll channels, a failure in one channel does not interfere with normal operation of the other channel if the failure is forward of the point at which the signals are combined. If a failure occurs in both channels, the servocylinder automatically centers and locks. In the yaw channel, a single failure causes the yaw servocylinder to center and lock by a spring-loaded action. The yar channel, exclusive of the yaw gyro, is considered functionally to be a part of the roll channel. A failure in the yar channel, with the exception of the yaw gyro, does not disengage the yaw channel. Any failure in the yar circuits is monitored by the roll-channel monitor and indicated by the roll-channel warning light.

The working and monitor channels, exclusive of power supplies, have no common electrical components, and all major electronic networks are molded into individual potted modules. Fail-safety dualization of the SAS does not exist for the rate gyros, servocylinders, and hydraulics, but the rate gyros utilize dual pickoffs and the servos utilize dual-position feedback pickoffs. The SAS electrical load can be carried by either of the two main power units in the airplane. Hydraulic system No. 1 supplies the SAS yaw servocylinder, and system No. 2 supplies the left and right servos.

The fail-safe characteristics of the SAS were determined and analyzed by using the X-15 six-degree-of-freedom flight simulator at North American Aviation, Inc. Systematically programed single and dual failures were introduced during various phases of simulated flight. Single failures within the dualized areas of the SAS created the unbalance required for tripout and servocylinder locking of the affected damper mode with little stick or airplane transient motion and little servocycling prior to locking. In the nondualized areas, only gyro failures of a mechanical nature produced potentially dangerous conditions. Only dual failures of a reinforcing type, which occurred simultaneously and caused hard-over or severe oscillatory servocycling signals, were found to be potentially dangerous. In all of the failures studied, it was determined that the pilot could retain aircraft control or manually disengage the affected dampers before catastrophic conditions were reached.

X-15 FLIGHT SIMULATOR

The X-15 flight simulator is a full-scale, ground-based reproduction of the X-15 cockpit and control systems. All instruments and systems are electrically and hydraulically actuated, with the aerodynamics and performance provided by analog computers. Flight system study and flight missions can be evaluated from piloted flight in six degrees of freedom.

A detailed description of the simulator is presented in reference 2.

OPERATIONAL EXPERIENCES

Basic Control System

Extensive operational testing of the manual flight control system was made with the flight control simulator and the X-15 aircraft. Abrupt inputs in each of the three axes and piloted maneuvers were used to determine the adequacy of the control system. The results revealed no instabilities in the control-stick system, and the side-to-center-stick tracking was found to be excellent. However, in roll, when excessive restraint was applied to the center stick and the side stick was then abruptly disturbed, a sustained side-stick oscillation was induced as long as the center stick was restrained. Because of the high magnitude of center-stick restraint necessary to obtain this condition, no modification has been considered necessary.

No significant deterioration of the flight components of the system was noted through usage.

The design and the actual force and displacement characteristics of the basic X-15 control system are compared in figure 4. Nonlinear gearing in the pitch mode is apparent. The slight upturn occurring in pitch stick force at 20° trailing edge up (fig. 4(b)) is caused by a boost bungee which becomes effective at this point to provide a more linear force characteristic for the nonlinear gearing. Breakout forces are considered by the X-15 pilots to be satisfactory for the entire X-15 program, although all control forces are lower than the design curve.

Development.- During the development and flight testing of the basic control system, several deficiencies were noted and modifications were initiated. One major deficiency still exists in the pitch-roll manual control system. Since the same horizontal-stabilizer linkages effect pitch and roll control, the artificial feel-force bungees are, of necessity, located in the cockpit area. This forward location of ground points allows the pivot points of the actuator-input walking beams to move under the influence of SAS servocylinder outputs, because of the large number of linkages involved. Attempted mechanical modifications, such as preloading, have not eliminated the problem.

A nonlinear gearing for the roll control (figs. 4(k) to (n)), similar to that used in pitch, has been installed in the X-15 aircraft to reduce lateral-control sensitivity about zero and minimize pilot tendency to overcontrol. The ratio of stick displacement and stick force to surface deflection was increased about 100 percent by the addition of the nonlinear gearing. Data from the three flights in which the modified gearing was used show no significant change in handling qualities, and the pilots report no apparent difference in the control characteristics.

The flight simulator has been used to evaluate modification to both the pitch and roll side-stick boost-cylinder control-valve centering bungees. The bungees have been stiffened to effect a force increase proportional to the rate of stick deflection. The effect was felt in both sticks and was similar to rate limiting rather then to the desired viscous damping of the control sticks. The pilots reported that the effect of the bungee alteration was desirable, but they felt that the restraining effect was excessive. Further tests are being made in an attempt to attain an optimum design.

A modification which increases the pilot-controlled longitudinal trim range from -20° to -25° trailing-edge-up stabilizer deflection has been used in two flights. The modification has proved to be desirable, inasmuch as it allows the pilot to trim to higher angles of attack.

Pilot evaluation.- Seven pilots have flown the X-15 airplanes. Based on flight and simulator experience, all the pilots agreed that control was improved by increasing the horizontal-surface rate from the original 15 deg/sec to 25 deg/sec and increasing the longitudinal trim limit from -20° to -25°. The pilots have had no major difficulty in performing the required control tasks, nor have they reported a lack of positive control of the airplane. They reported varying degrees of control sensitivity when using the side stick, especially in roll. Early in the program, the center stick was preferred to the side stick; however, after the pilots gained more experience with the side stick, they rated it as equal to the center stick, and, in many cases, even indicated a preference for the side stick. (An inflated pressure suit interferes with normal center-stick operation, for example.) The pilots agree that center-stick operation is satisfactory and that rudder control is too coarse for use under marginally controllable flight conditions. While extreme rearward side-stick deflection is being maintained, the pilots report experiencing a form of wrist lock when lateral-control movements are attempted. This creates both a sense of awkwardness and imprecise control inputs. Lessening the force by increasing the trim range has helped to alleviate the problem. The pilots report difficulty and inadvertent control inputs when holding forward force on the side stick and operating the thumb trim wheel. General improvement in side-stick performance could be made by relocating the lateral-control pivot point from a position below the forearm to a position on the axis of the forearm. The pilots indicated that a stepped-type trim button would be preferable to a continuous-type trim button.

Stability Augmentation System

Data obtained in X-15 flights from December 1958 to January 1962 were analyzed to determine the operational performance of the SAS. The study revealed that the system has performed consistently at all gain settings tested, shown no apparent deterioration, and operated within design tolerances. With the SAS operating at nominal gain settings of 8-6-8 (pitch, roll, and yaw, respectively), speed and altitude missions in excess of the X-15 design envelope may be made (ref. 3). This single-setting potential was not anticipated during the X-15 design and early development periods.

The dynamic effects of the damper on airplane motions were investigated by making pilot-initiated pulses about all three axes with the dampers on and off. Figures 8(a) to 8(c) present typical time histories which show the increase in damping when SAS is used in all three axes. The data were obtained by operating the X-15 flight simulator so that the SAS-on and SAS-off responses could be compared at exactly the same flight conditions. The damping shows an obvious improvement in all three axes with the SAS on. The handling qualities are, of course, also greatly improved about all three axes because of the increased damping. Damping is especially important in roll because it serves to reduce the lateral-control sensitivity.

Longitudinal pulses were made and evaluated in the Mach number range between 0.6 and 4.5. With dampers off, the airplane motion was found to be lightly damped, and the damping decreased with increasing Mach number ($M > 1$).

With the pitch damper at the minimum gain setting, damping of the airplane oscillations was rated as acceptable by the pilots. For all conditions investigated, the longitudinal oscillations were readily controlled by the pilots.

The lateral motions of the airplane were investigated in the Mach number range between 0.6 and 4.5. For all conditions investigated with roll dampers off, the airplane oscillations were poorly damped, with coupling in roll and yaw. Although the pilots reported high roll-control sensitivity with dampers off, all conditions were controllable except for Mach numbers greater than 2.2 at angles of attack greater then 8°. At these conditions, a pilot-airplane lateral-directional divergence was encountered that was uncontrollable when normal piloting techniques were used (ref. 3). This phenomenon has made the use of the SAS a necessity at angles of attack greater than 8°. With a nominal roll-damper gain setting, the airplane rolling motions were well damped. A small-amplitude residual (limit cycle) oscillation was apparent at high dynamic pressures and high roll-damper gain settings. This phenomenon is discussed subsequently. The pilots reported a satisfactory sense of positive lateral control with the roll damper on.

Directional airplane motions were investigated in the Mach number range between 0.6 and 5.6. In general, the airplane exhibited good directional stability in the speed and altitude regions investigated with the yaw damper off, but at the higher Mach numbers and high angles of attack, the yawing oscillations were neutrally damped to slightly divergent. With the yaw damper on, these airplane motions were well damped.

The yar-damping mode is provided to counteract the rolling moment produced by deflection of the vertical tail at elevated angles of attack. At these angles, reduced effectiveness of the upper vertical tail and increased effectiveness of the lower vertical tail create a rolling moment proportional to the vertical-tail deflection.

Additional information on the aerodynamic effects of the stability augmentation system are presented in reference 4.

Problems.- Limit cycles, or residual oscillations, caused by the stability augmentation system were encountered first in the X-15 flight simulator. Phase lag produced by the hysteresis and dead band at very small amplitudes caused the limit cycles, which existed in all three axes when the damper gain and control power were high. The hysteresis is the result of the free play in the control linkages between the bungees and the walking beams of the basic control system. In the early flight program, limit cycles of small amplitude were observed in the flight records, but were unnoticed by the pilots. In subsequent flights to higher dynamic pressure, the pilots reported limit-cycle oscillations in roll. Generally, the limit-cycle appeared when the roll-damper gain setting was greater than 6 and the dynamic pressure greater than 500 lb/sq ft. These limit cycles are a function of SAS gain and control effectiveness (ref. 4). The pitch limit cycle occurred at a higher gain than in roll at a given dynamic pressure. Although the pilots have been aware of the yaw limit cycle on occasion during the higher performance flights, it has not been a problem. The roll limit cycle produced the largest amplitude; however, this amplitude was less than 1° change in bank angle at frequencies of 1 cps to 3 cps.

Figure 9 compares the limit-cycle characteristics of the X-15 airplane ground tests (aerodynamics were simulated by an analog computer), the X-15 flight simulator, and the available X-15 flight data. In general, these three types of data are comparable, but the limit-cycle characteristics are more severe for the X-15 flight simulator than for actual X-15 flight at the same given conditions. In an initial effort to alleviate the limit-cycle problem, phase lead was increased in the system by removing the original shaping which had provided system lag. Subsequent flights with this configuration resulted in a limit cycle of lower amplitude at higher frequency. Although this was a more acceptable system in regard to limit cycles, the SAS then became susceptible to a high-frequency (12 to 13 cps) vibration, discussed in the following paragraph. Additional analysis and discussion of the limit-cycle problem are presented in reference 4.

The X-15 horizontal surfaces are very lightly damped as a result of their rigid, welded construction. Excitation at their natural frequencies results in a closed-loop vibration, which couples the surface vibration and the SAS gyros through the inertial reaction of the fuselage. A resonance of 13 cps, corresponding to the first bending mode of the horizontal stabilizer, is caused by the SAS. Actuator rate-limiting confines the output to 2° peak-to-peak differential surface deflection. The vibration has occurred in both pitch and roll.

Development.- A vibration at high SAS gains during the first X-15 captive flight was eliminated by relocating the gyro package from the instrument compartment to the center-of-gravity compartment, thus removing the gyro from a point influenced by fuselage bending.

In an effort to alleviate both the limit-cycle and vibration problems, a high-response "notch filter" was investigated to lower the system response at the structural-vibration frequency of the horizontal surfaces. Phase characteristics of the filter were chosen so that phase shifts in the frequency range of the limit cycle were more tolerable. The filter was mechanized on an analog computer and ground tested on the actual aircraft individually in each mode and simultaneously in all modes. The problem of the sustained vibration appears to be eliminated. Another solution being investigated is a pressure-feedback main-actuator control valve which will provide damping to the actuator at the natural frequency of the horizontal surface. Initial tests indicate that the valve is highly desirable, but further development is required to finalize the configuration and characteristics.

To provide operational redundancy, an independent pitch-roll SAS backup system has been developed. This package operates at a fixed gain level (ground preset), contains its own sensors, has minimum electronics, and feeds directly to the existing SAS servos. The system contains no fail-safe features, is "pilot-elect" switch-controlled, and designated for emergency use only. When the pilot elects to use this system, it will automatically function upon failure of the normal SAS roll mode.

Inasmuch as the final SAS preflight check is made 3 to 15 days prior to flight, a pilot "in-flight" test system was provided for a prelaunch test of all SAS electrical functions (both working and monitor channels), except the gyros, at one ground preset level.

Flight simulator studies of SAS fail-safe features disclosed a need for dualization of servocylinder feedback wiring to enable the malfunction detector to detect electrical failure. The dualization was accomplished during the demonstration flight program. Also incorporated into the hydraulic system was an auxiliary SAS hydraulic package comprised of a self-contained reservoir and a hydraulic motor/pump assembly driven by the No. 1 main hydraulic system. This package provides hydraulic power directly to the left and right SAS servocylinders in the event of a failure of the No. 2 auxiliary power unit, but retains the independence between the main hydraulic systems. The package contains no electrical components and enters operation automatically by a pressure-priority-selector valve.

Pilot evaluation.- As a result of flight or simulator experience, or both, all X-15 pilots generally agree that the SAS functions well as a damper. They believe that the maximum gain capability of the SAS is sufficient for the entire X-15 flight envelope, and that the dampers reduce the lateral-control sensitivity. Roll damping is required to fly the X-15 maximum-performance flights, particularly during atmospheric entry at high angles of attack when the airplane is uncontrollable (ref. 3) without dampers. Because of this need for dampers, all the pilots desired hydraulic duality or hydraulic package backup for the pitch-roll system and a redundant or other backup system for the roll damper. The pilots rated the importance of the SAS modes as roll, pitch, and yaw. They found the roll limit cycles to be annoying during higher performance flights and believed they should be eliminated or reduced. No pilot wished to make another flight with the phase-lead shaping which sustained the high-frequency vibration.

All of the pilots desire improvement in the SAS gain-selector knobs, since the heavy flight gloves provide a poor knob grip and cover the switch-position numbers when the switches are being operated. The pilots find the SAS console difficult to reach under the normal seat and head restraints, and practically impossible to reach in an inflated pressure suit.

Reliability

Reliability tests.- Design specifications of the stability augmentation system required that: (1) The probability of no failure of the stability augmentation system would be at least 0.995 during any (each) flight throughout the life of the airplane. Airplane life is defined as 100 one-half-hour flights plus 5 hours between flights, for a total of 550 hours. (2) Servocylinders would operate 10 missions or 50 hours under thermal cycling, plus 50 ground hours without maintenance and 100 missions before complete replacement. (3) All components would be capable of a minimum of 650 hours of operation before complete unit replacement, and the entire system (including servocylinders and ship's wiring) would have a minimum mean-time-to-failure (MTF) of 100 hours.

Reliability tests of the stability augmentation system were conducted using the finalized electronics-case assembly (ECA). Prior service-life testing for design deficiencies showed that the ECA represented the mean-time-to-failure of the entire SAS. The ECA was subjected to a series of duty-cycle tests of

85 minutes duration at all environmental conditions. The SAS operated for 53 minutes of each test cycle. A total of 568 duty cycles, representing 500 hours of ECA operation, was required for the test. During these tests, 3 modules failed, which resulted in an acceptable MTF of 167 hours.

The following tabulation, which compares the module failure rates under three different environmental conditions, indicates that the failure rate is affected by temperature cycling:

Operation	Temperature profile	Module failure
Service-life test	Severe (185° F to -60° F)	10 in 461 hours
Reliability test	Moderate (140° F to -20° F)	3 in 500 hours
Flight simulator	Room temperature	0 in 1,000 hours

Operational reliability.- The X-15 program offers the opportunity for accurate documentation of the SAS failures over the entire operational life of the system. The data presented in this paper cover the 37 months following delivery of the first of four flight-qualified systems which have been used with two X-15 aircraft. SAS malfunctions have affected 25 percent of the X-15 free flights (20 percent of the total captive plus free flights). Of the X-15 free flights, 8 were elected to be flown with a known SAS failure. The total malfunctions incurred in flight were 7 in pitch, 7 in roll, and 1 in yaw. In only one flight did more than one damper mode simultaneously malfunction.

Reliability data are presented herein in the categories of (1) flight operation, covering power-on time, from close-out for flight (essentially, B-52 engine start) through landing stop of X-15; (2) field service operation, extending from SAS preflight for the first captive X-15 flight through the final flight, including only power-on hours used in preflighting and actual flight; and (3) total operation, from delivery of the first SAS to January 1, 1962, including all ground and flight power-on system operation.

During the total operational period of 1,610 SAS hours, 107 individual SAS components failed. Of these, 100 failed during ground servicing, and 7 failed after the system was closed out for flight. Figure 10 is the complete failure record for the X-15 program. As expected, most of the failures occurred early in the program when operation was at a maximum and technical experience at a minimum. Of the components, the modules failed most frequently. Detailed study of the failed modules revealed that transistor malfunction was the most common cause of the failures.

Table II divides SAS categories into significant periods. The "Projected flight operation" category consists of the current trends projected to completion of the X-15 program. The flight operations categories are broken down into take-off to landing (as they affect cost, time, and program progress) and launch to landing (affecting safety-of-flight and mission success). The "Total field service operations" category is broken down into the LR11 engine period, which was the development period for systems, and the XLR99 engine period, which was the flight-data research period. For comparison, the design specification, environmental service life, and reliability tests are listed. From the standpoint of component failure, the overall operational reliability R has been: total flight, 91.5 percent; free-flight phase, 97.6 percent; field operation, 35 percent; and total operation, 26.4 percent.

Figure 11 presents faired curves of component failure for the field service period. Also included, for comparison, is the flight-failure curve. It can be readily seen that the failure rate is decreasing, as evidenced by the slope of the curves approaching zero.

Malfunctions are defined as any interruption of normal operation; therefore, it is possible to encounter more component failures than malfunctions if the failure does not affect component operation. "Permanent" malfunctions occur when a damper mode trips out and the pilot cannot effect reengagement; "intermittent" malfunctions occur when a damper mode trips out, but the pilot can immediately effect reengagement. Figure 12 is the complete flight-malfunction history of the SAS. Most of the malfunctions occurred early in the program. If the X-15 had been an unmanned vehicle, such as a missile, and if mission success had depended on no system malfunction after system close-out, 7 of the 10 maiden X-15 flights would have failed.

Table III presents a breakdown of the malfunctions affecting flight and the period in which the malfunction occurred, regardless of the time of discovery. Permanent failure was the predominant malfunction and occurred, in all but one instance, during the captive flight phase. The overall flight reliability in relation to system malfunction has been 86.1 percent.

Table IV lists the average failure and malfunction rates during the various phases of the X-15 program. As can be seen, the rates decrease by an average factor of 4 between the initial and final periods.

Future X-15 flights should attain the extremes in speed and altitude of the flight envelope. At the system's current overall performance level, the probability of the success (that is, no component failure during free flight) of five projected missions is 88.4 percent. The probability of success (no system malfunction during free-flight phase) of all five missions is 24.7 percent.

It is evident from the data trends that the SAS has entered the period (of undetermined duration) of adequate reliability before true wearout failure becomes predominant. This indicates that the system should be used at X-15 maximum performance in the immediate future.

Eighty percent of all malfunctions affecting flight occurred during the first 40 percent of the flight program. During the first 50 percent of the flight program, 80 percent of all component failures occurred. Eighty-six percent of the component failures affecting flight and 70 percent of the malfunctions affecting flight were directly attributable to human error in servicing and handling. Thirty-five percent of the total components which have failed to date were the result of human error during ground maintenance operations. It is also noteworthy that all but one of the seven component failures affecting flight were the result of human error which was not detected prior to flight, and that 50 percent of these failures were external of the actual SAS in the form of a break in the ship's wiring. Failures attributable to human error have decreased with time because of increased familiarity with the system and improved maintenance and inspection procedures. Sixty-five percent of the failures did not involve the electronics-case assembly, which was expected to be the area of least reliability. Only once did a damper channel permanently malfunction after launch; however, in all instances, the flights could have been terminated before the committed portion.

CONCLUDING REMARKS

Flight and simulator studies of the X-15 control system encompassing the flight envelope of the X-15 airplane show that the manual aerodynamic control system and the stability augmentation system are adequate for the X-15 flight envelope.

A stability-augmentation-system limit cycle, primarily in roll, occurred during flight at dynamic pressures greater than 500 lb/sq ft and roll gain settings equal to or greater than 6. With high-response lead shaping, the limit cycles were reduced to an acceptable level but gave rise to a high-frequency sustained vibration of the horizontal stabilizers. A "notch filter" is being incorporated to minimize both problems.

There has been no apparent deterioration of the control systems through usage.

The pilots expressed the need for redundancy in the roll damper for extension of the flight envelope. Modifications were made during the current flight program to improve the reliability, fail-safety, and operational characteristics of the systems.

Transistor failure was found to be the most common cause of module failures. The free-flight reliability of the stability-augmentation-system components has been 97.6 percent. Malfunctions have affected 25 percent of the free flights, but the malfunction rate has decreased with experience and technical familiarity with the system. Eighty-six percent of the component failures affecting flight were the result of human error. Seventy percent of the malfunctions affecting flight were caused by human error. Although the overall flight reliability does not meet specifications at this time, the current trend indicates that the system will approach adequate reliability during the flight period now in progress.

Flight Research Center,
National Aeronautics and Space Administration,
Edwards, Calif., September 19, 1962.

APPENDIX

X-15 FLIGHT HISTORY WITH THE STABILITY AUGMENTATION SYSTEM

A brief résumé of the flight history of the X-15 stability augmentation system is presented. In 80 attempts, 24 flights of X-15-1 and 21 flights of X-15-2 have been made successfully. Fifteen of the flights were performed with the XLR99 engine.

Of the seven X-15 pilots, only three have experienced SAS failure or malfunction. One pilot has had four damper failures (all prelaunch), three flight nuisance tripouts, and 18 seconds of high-frequency vibration after landing. Another pilot has had one damper failure (prelaunch) and one flight nuisance tripout. The third pilot has had two damper failures (one prelaunch), two flight nuisance tripouts, and 57 seconds of high-frequency vibration during flight. All pilots have had flight experience with various damper modes intentionally inoperative and extensive flight-simulator time under normal and SAS-failed conditions.

X-15-1

On flight 1-C-1*, SAS checks during flight revealed system vibration at high (8-8-8) gain settings. The gyro package was subsequently relocated to the center-of-gravity compartment.

The first free flight (1-1-5) was made with the pitch damper failed; this was known prior to launch. The pilot experienced severe longitudinal pilot-induced oscillations on landing approach when using the side stick for control. Postflight inspection revealed that a pin connector in the pitch modulator-demodulator module was broken by a technician in securing the system after the final preflight check.

Flight 1-2-7 was made with the pitch damper failed; this was not known prior to launch. Postflight inspection revealed a dual failure in the pitch mode. A broken input lead in the pitch working channel (ship's wiring) from the gain-selector switch to the SAS electronics-case assembly (ECA) was found in the pilot's console. Also, the gyro return ground lead in the pitch monitor channel had been omitted during earlier factory rewiring of the ECA. These malfunctions caused failure of both the monitor and working channel and prevented a malfunction-light indication. The failures were not detected because of inadequate ground check procedures. The pilot experienced moderate longitudinal pilot-induced oscillations at landing.

Flight 1-3-8 was made with the roll damper failed; this was known prior to launch. Postflight inspection revealed a broken roll-channel signal lead in the

*In the flight-designation system adapted for the X-15, the first digit indicates the airplane number; the second is the free-flight number or captive (C) or abort (A) letter designation; and the third is the flight-attempt number.

pilot console. This lead is part of the aircraft wiring and was broken when other work was done in the pilot's console after the final preflight stability-augmentation-system check. The console was rewired to improve its reliability.

Flight 1-4-9 was made with all stability-augmentation-system channels functioning satisfactorily. However, postflight inspection revealed a major leak in the hydraulic line to the stability-augmentation-system yaw servo-cylinder. Loss of the No. 1 hydraulic system would have resulted if the flight had been longer than the estimated 3-minute duration. This was not interpreted as a flight failure of the SAS.

Flight 1-6-11 was made with roll damper failed; this was known prior to launch. Postflight inspection revealed a SAS roll-shaper module failure. A pitch servoamplifier module was also found to be below minimum specification, but not failed.

Flight 1-A-14 was aborted because of a pitch-damper failure prior to launch. Subsequent ground check revealed an improperly set malfunction-trip level.

Flight 1-19-32 was made with all dampers operating normally. The pilot turned the pitch damper off for data maneuvering and neglected to reengage it upon completion. Unaware that the pitch damper was off, the pilot made a normal landing; no pilot-induced or other longitudinal oscillations were noted.

SAS operation on all other X-15-1 flights has been satisfactory, except for small-amplitude limit cycles in roll experienced in most flights.

X-15-2

Flight 2-C-1 was made with all stability-augmentation-system channels operating normally except for one tripout of the yaw channel. The channel was immediately reengaged. Subsequent checks revealed high malfunction-detector voltages. These high voltages were caused by poor adjustment, which probably resulted from the potentiometer nuts being tightened after alinement and not rechecked for possible setting change.

Flight 2-2-6 was made with the roll damper failed; this failure occurred at launch. Postflight tests revealed normal SAS operation. Preflight checks for the subsequent flight revealed a failed roll-malfunction-detector module which was assumed to be intermittently defective. The cause of failure in flight 2-3-9 (see following paragraph) was also the cause of this flight failure.

Flight 2-3-9 was made with the roll damper failed; this failure occurred at launch. Postflight checks revealed an incorrectly wired ground lead (ship's wiring) to the plug of the SAS electronics-case assembly. The error had not been detected because of inadequate preflight test techniques. Before the cause of the failure was discovered, a new SAS was installed and the pilot's console was rewired to improve reliability.

Flight 2-5-12 was made with all stability-augmentation-system channels operating normally except for one nuisance tripout of the pitch mode when the pilot actuated the fuel-jettison switch. Since induced signals were believed to be the cause of the tripout, diodes were placed across the liquid-nitrogen-valve solenoid.

Flight 2-6-13 was made with all stability-augmentation-system channels operating normally except for a roll-mode nuisance tripout experienced during a three-revolution maximum-aileron rolling maneuver. Postflight checks revealed no SAS discrepancies, and the tripout remains unexplained.

Flight 2-8-16 was made with all stability-augmentation-system channels operating normally. The pilot deliberately landed with all the channels inoperative. No unusual oscillations were noted.

Flight 2-9-18 was the first flight made with the modified SAS shaping (high-response filter). The pilot used high gain settings and evaluated the roll limit cycle. SAS operation was normal except for one unexplained pitch-damper tripout when the engine master switch was turned off. Upon landing impact, a high-frequency vibration (mentioned previously) was experienced for 18 seconds until the pilot realized the source and turned off all the dampers. The vibration was so severe that the pilot first thought he had lost the nosewheel. A SAS gain-reduction switch was subsequently installed on the main landing gear in the belief that only ground contact would provide sufficient excitation.

Flight 2-14-28 was flown with the modified SAS shaping. The system operated normally except for 57 seconds of high-frequency vibration triggered by a series of rapid, large (stop-to-stop) control inputs made by the pilot while evaluating the side stick. At the beginning of the vibration, the pilot thought the phenomena were buffeting. He later sensed that the vibration was primarily in the pitch mode and made gain reductions in both pitch and yaw. The vibration ceased. Flight records, however, showed the vibration to be primarily in the roll mode. As a result of this flight experience, the SAS was restored to its original shaping (low-response filter).

Flight 2-15-29 was flown with normal damper operation except for one nuisance tripout of the pitch mode at engine shutdown. Postflight checks revealed no system discrepancies. Numerous floating and shorted ground shields in the ship's wiring were found and repaired. The shields are believed to have caused earlier unexplained tripouts in flight and ground runs.

Flight 2-16-31 was normal until engine start, when the pitch damper failed. Concurrent with the pilot's second attempt to reengage the pitch mode, the roll damper failed but was immediately reengaged and operated satisfactorily for the remainder of the flight. Postflight checks revealed normal system operation. The entire SAS was removed from the aircraft for extensive bench checks, which revealed broken wafers in the pitch, roll, and yaw gain-selector switches. The switch shafts had been longitudinally drilled and tapped for a new type of knob prior to flight, which caused the severe wafer damage. Engine vibration in flight was sufficient to cause discontinuity. It is believed that the pitch-switch wafers were partially broken prior to flight 2-15-29 and caused a tripout.

An open resistor in the left servoamplifier module was also found during the bench check. This was a noncritical item and did not affect system flight performance. The module had been rejected a year earlier for the same reason and had been returned to the contractor. (The resistor opened after 20 minutes of power-on operation, but performed normally when cold.) This module was erroneously returned to stock and eventually reinserted into the SAS.

SAS operation on all other X-15-2 flights has been satisfactory except for the roll limit cycle experienced during most flights with low-response shaping.

REFERENCES

1. Matranga, Gene J.: Launch Characteristics of the X-15 Research Airplane as Determined in Flight. NASA TN D-723, 1961.

2. Cooper, Norman R.: X-15 Flight Simulation Program. Aerospace Eng., vol. 20, no. 1, Nov. 1961, pp. 16-17, 71-77.

3. Petersen, Forrest S., Rediess, Herman A., and Weil, Joseph: Lateral-Directional Control Characteristics of the X-15 Airplane. NASA TM X-726, 1962.

4. Taylor, Lawrence W., Jr., and Merrick, George B.: X-15 Airplane Stability Augmentation System. NASA TN D-1157, 1962.

TABLE I.- GAIN AND AUTHORITY OF THE X-15 STABILITY AUGMENTATION SYSTEM

Gain setting	Gain							
	Pitch		Roll		Yaw		Yar	
	Servo-cylinder	Surface	Servo-cylinder	Surface	Servo-cylinder	Surface	Servo-cylinder	Surface
	In./deg/sec	Deg/deg/sec	In./deg/sec	Deg/deg/sec	In./deg/sec	Deg/deg/sec	In./deg/sec	Deg/deg/sec
1	0.005	0.075	0.0017	0.051	0.004	0.03	0.003	0.09
2	.010	.150	.0033	.100	.008	.06	.006	.18
3	.015	.225	.0050	.150	.012	.09	.009	.27
4	.020	.300	.0067	.200	.016	.12	.012	.36
5	.025	.375	.0083	.250	.020	.15	.015	.45
6	.030	.450	.0100	.300	.024	.18	.018	.54
7	.035	.525	.0117	.350	.028	.21	.021	.63
8	.040	.600	.0134	.400	.032	.24	.024	.72
9	.045	.675	.0150	.450	.036	.27	.027	.81
10	.050	.750	.0167	.500	.040	.30	.030	.90
Condition	Servocylinder and surface limits							
Normal functioning	Maximum servocylinder stroke = ±1.0 inch or ±15° of horizontal stabilizer		Maximum servocylinder stroke = ±1.0 inch or ±15° of horizontal stabilizer		Maximum servocylinder stroke = ±1.0 inch or ±7.5° of vertical stabilizer		Maximum servocylinder stroke = ±1.0 inch or ±15° of horizontal stabilizer	
Mal-functioning	0.1 inch or 1.5° of horizontal stabilizer		0.1 inch or 3° differential stabilizer		0.1 inch or 0.75° of vertical stabilizer		0.1 inch or 3° of differential stabilizer	

TABLE II.- OPERATIONAL COMPONENT FAILURES OF THE X-15 STABILITY AUGMENTATION SYSTEM

	Flights		System total-			Average duty cycle	Mean time to failure	Reliability, R	Component failures						
									ECA		Other SAS components				
	X-15/ B-52	X-15	Failures	Operating hours	In-flight operating hours				Modules	Other[1]	Gyro	GSSA	Servo-cylinder	Misc.[1]	Ship's wiring
Projected flight operation															
B-52 take-off to X-15 landing	400	200	0	12,000	300	1.5	300	99.5	0	0	0	0	0	0	0
X-15 launch to landing	400	200	0	12,000	50	.25	50	99.5	0	0	0	0	0	0	0
SAS design specification	300	300	0	550	50	.5	100	99.5	0	0	0	0	0	0	0
Service-life test - environmental	300	300	11	635	461	1.75	42	95.9	11	0	0	0	0	0	0
Reliability test - environmental	300	300	3	805	500	1.42	167	99.3	3	0	0	0	0	0	0
Total flight operations				1,380											
B-52 take-off to X-15 landing	80	45	7		180	2.25	25.7	91.5	2	0	0	1	0	1	3
X-15 launch to landing	80	45	1		7	.16	7	97.6	0	0	0	1	0	0	0
LR11 flight operation				1,190											
Take-off to landing	56		6		126	2.25	21	89.8	2	0	0	0	0	1	3
Launch to landing		30	0		4.8	.16	∞	∞	0	0	0	0	0	0	0
XLR99 flight operation				170											
Take-off to landing	24		1		54	2.25	54	95.9	0	0	0	1	0	0	0
Launch to landing		15	1		2.4	.16	2.4	93.5	0	0	0	1	0	0	0
Total field service operations (ground preflight plus flight)	80	45	83	502	180	6.3	6.1	35	27	26	1	3	3	7	16
LR11 period	56	30	73	410	126	7.8	5.6	24.9	25	23	1	0	2	6	16
XLR99 period	24	15	10	92	54	4.8	9.2	59.2	2	3	0	3	1	1	0
Total operations (all ground plus flight since first system delivery)	80	45	107	1,610	180	20	15	26.4	41	31	1	3	3	9	19

[1]Include relays (6), chokes (1), potentiometers (9), transformers (1), resistors (3), cold solder (3), broken wire (8), sockets/pins (4), pick-offs (3), cam bearings (2).

TABLE III.- X-15 SAS MALFUNCTIONS AFFECTING FLIGHT

Operation	Malfunction type	Captive phase	Free-flight phase	Total	Flights affected		Reliability, R		Total reliability	
					B-52/X-15	X-15	B-52/X-15	X-15	B-52/X-15	X-15
Total flight	Intermittent	2	5	7	14	11	91.6	89.0	82.8	86.1
	Permanent	7	1	8			90.5	97.7		
LR11 flight	Intermittent	2	2	4	11	8	93.1	89.0	82.0	93.5
	Permanent	7	0	7			88.0	∞		
XLR99 flight	Intermittent	0	3	3	3	3	88.0	81.9	84.6	76.5
	Permanent	0	1	1			95.9	83.5		

TABLE IV.- X-15 AVERAGE SAS FAILURE RATES

Operation	Initial period	Overall period	Final period
B-52/X-15 flights	First 20 flights	80 flights	Last 25 flights
	1.5 component failures per flight 0.5 malfunction per flight	1 component failure per flight 0.2 malfunction per flight	0.5 component failure per flight 0.1 malfunction per flight
X-15 flights	First 10 flights	45 flights	Last 15 flights
	6 component failures per flight 0.7 malfunction per flight	2 component failures per flight 0.3 malfunction per flight	1 component failure per flight 0.15 malfunction per flight
Field operation period	First 150 hours	502 hours	Last 100 hours
	0.4 component failure per hour	0.15 component failure per hour	0.1 component failure per hour
Total use period	First 1,200 hours	1,610 hours	Last 200 hours
	0.06 component failure per hour	0.07 component failure per hour	0.03 component failure per hour

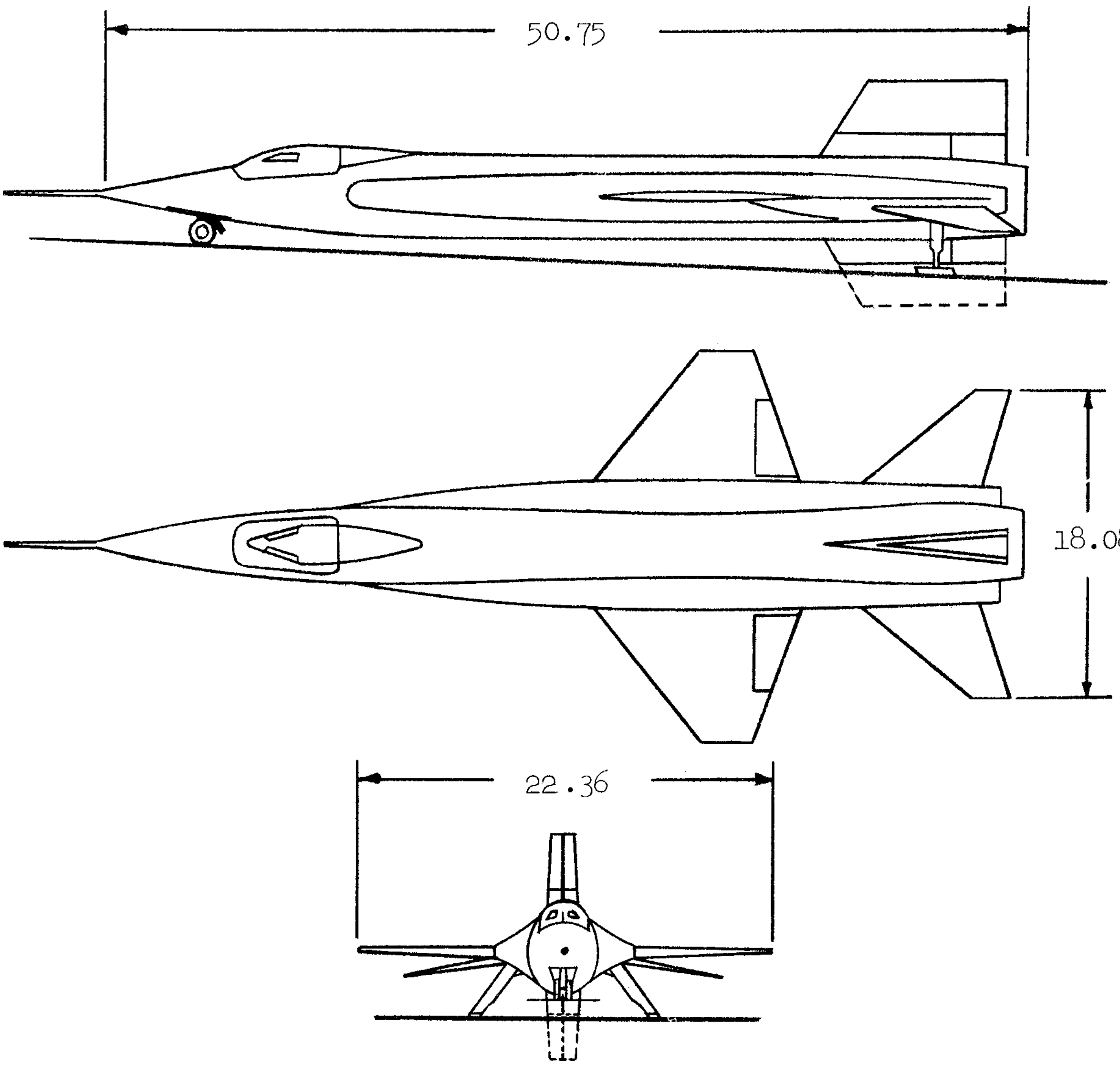
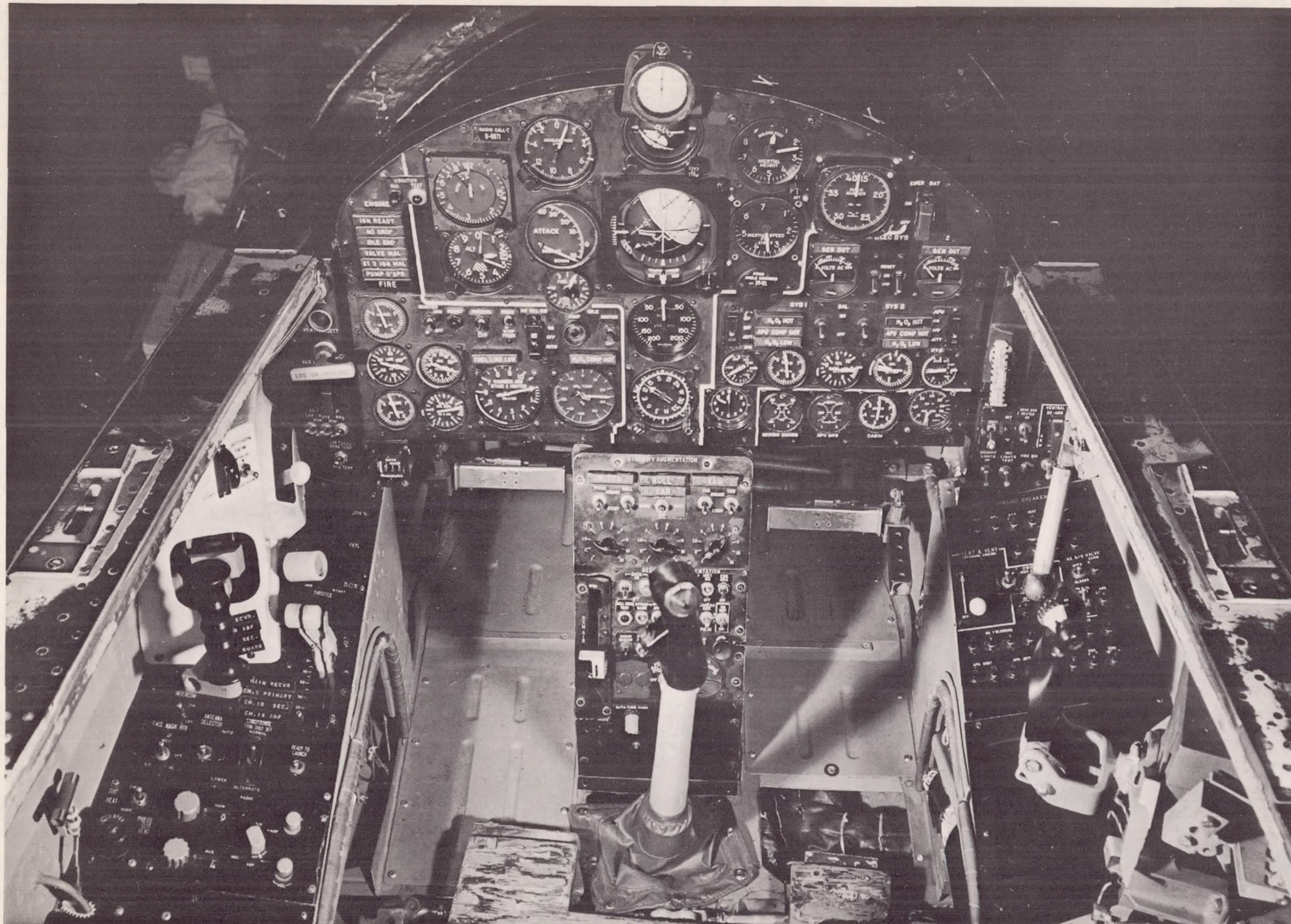


Figure 1.- Three-view drawing of the X-15 airplane. All dimensions in feet.



E-6587

Figure 2.- Photograph of the X-15 cockpit.

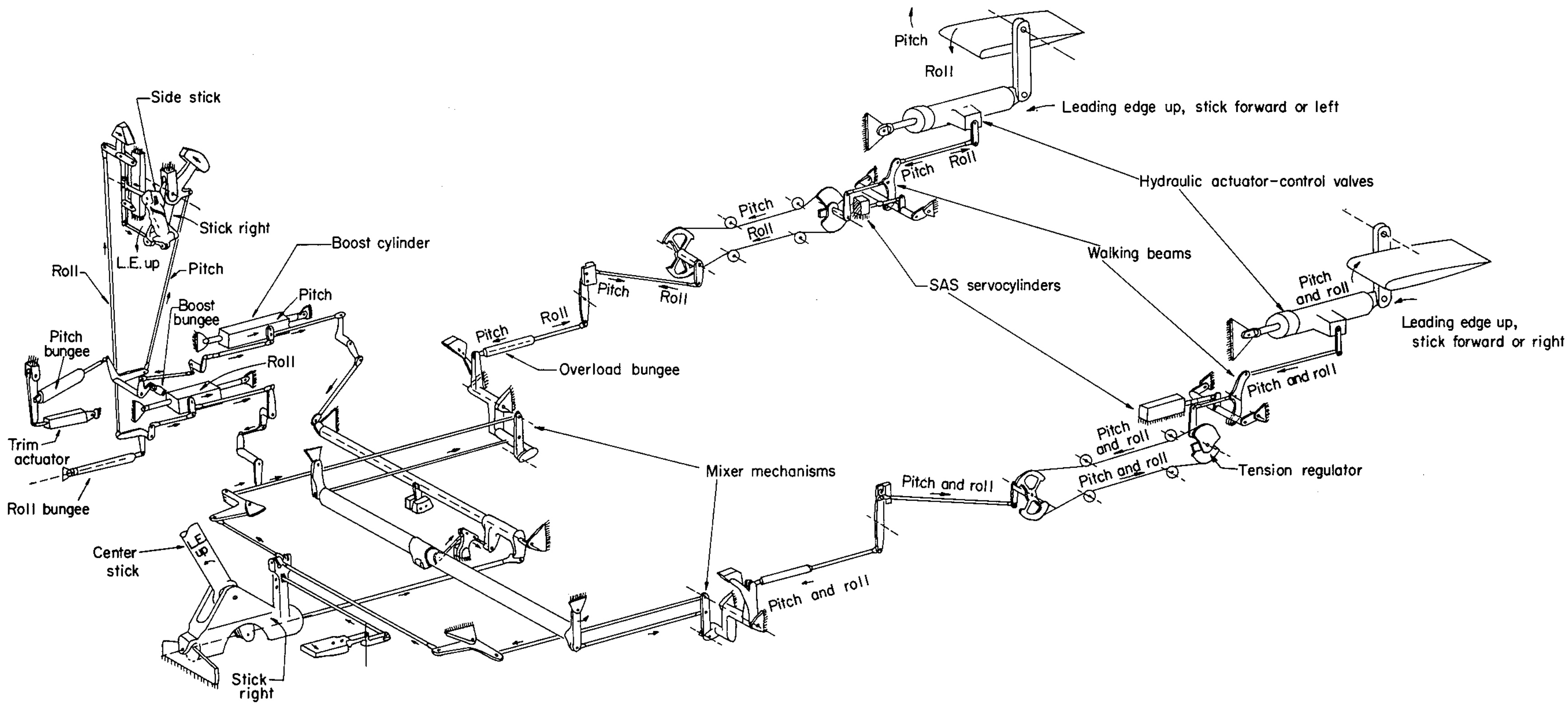
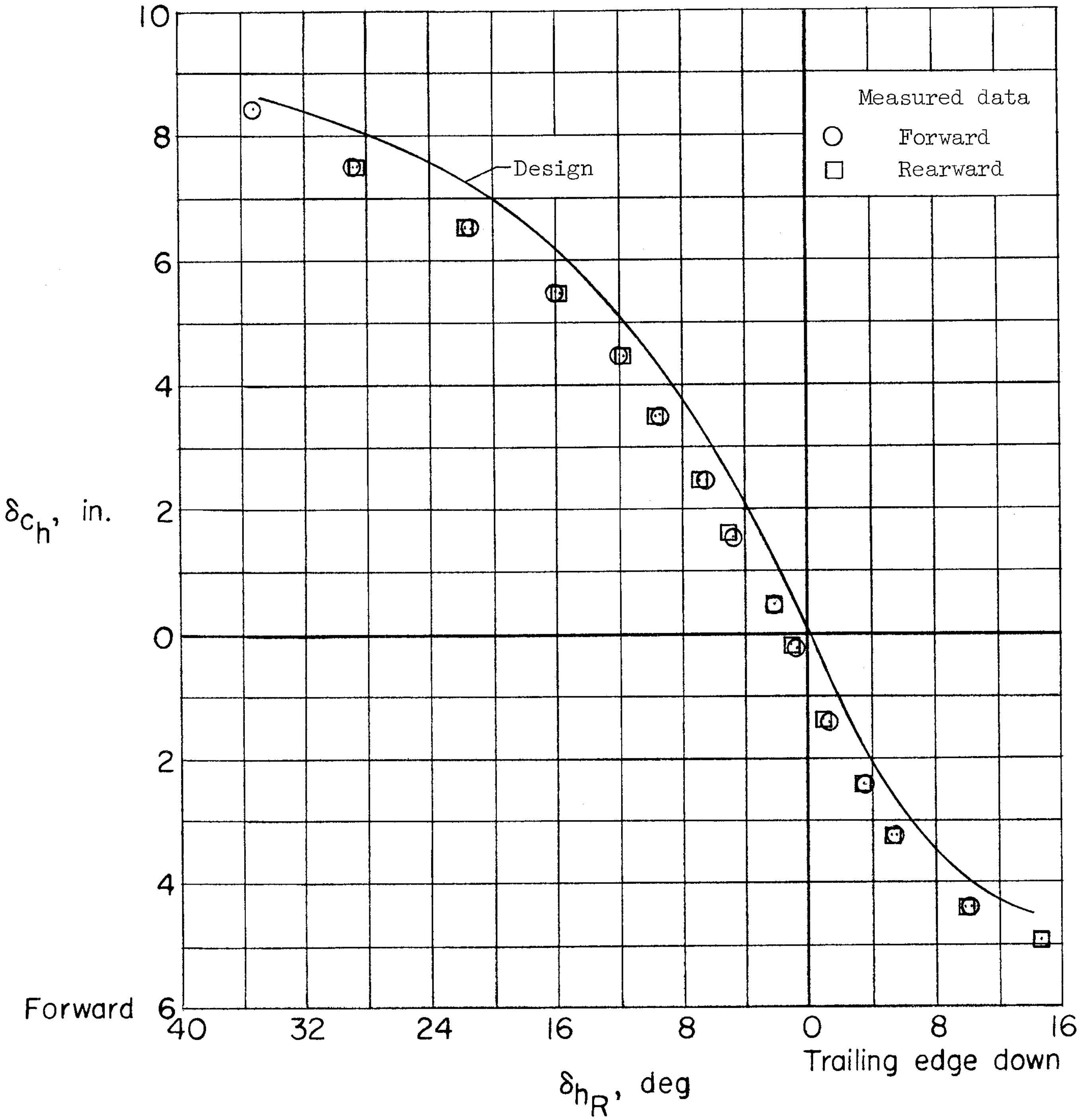
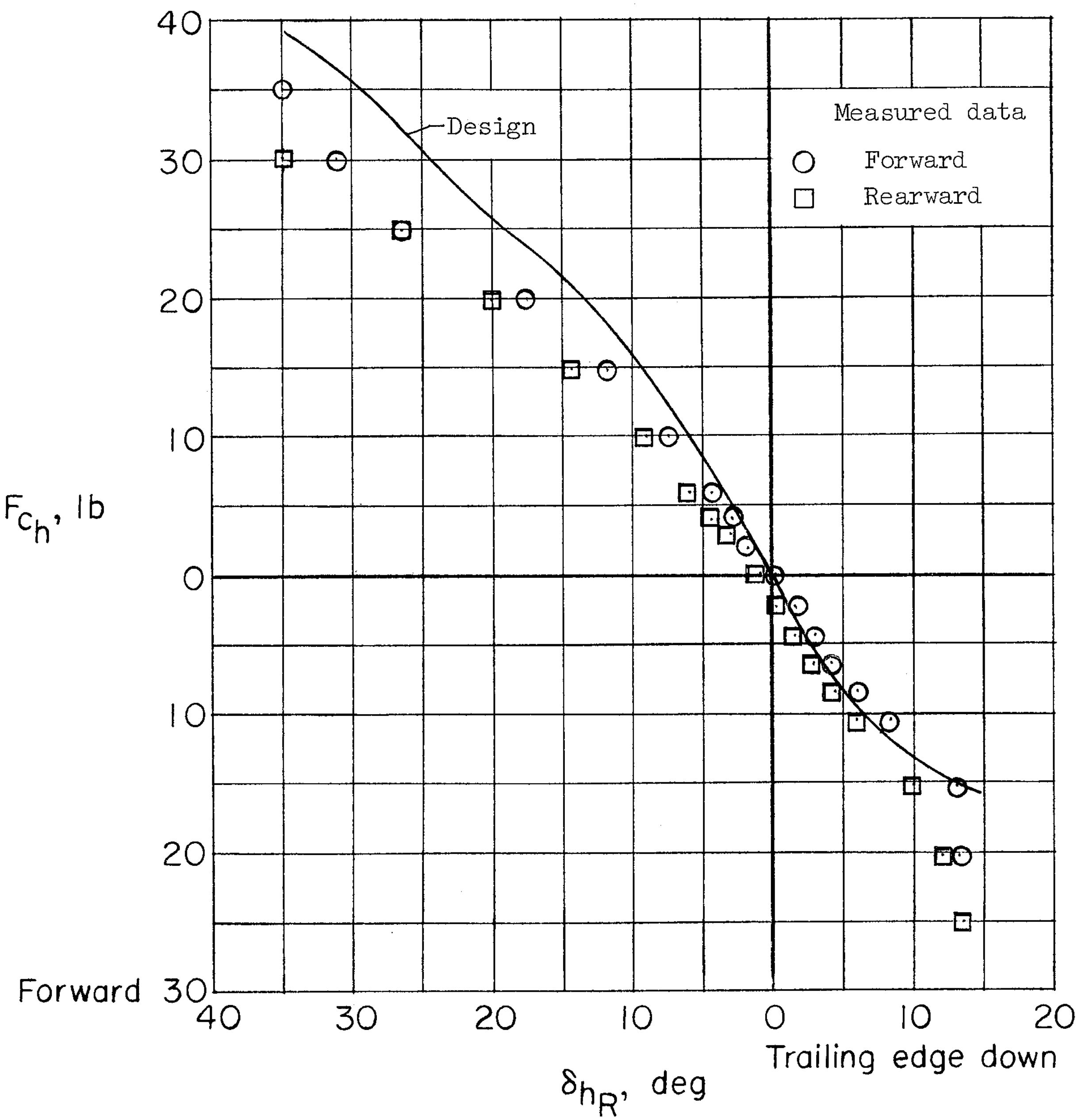


Figure 3.- X-15 basic pitch and roll control system.



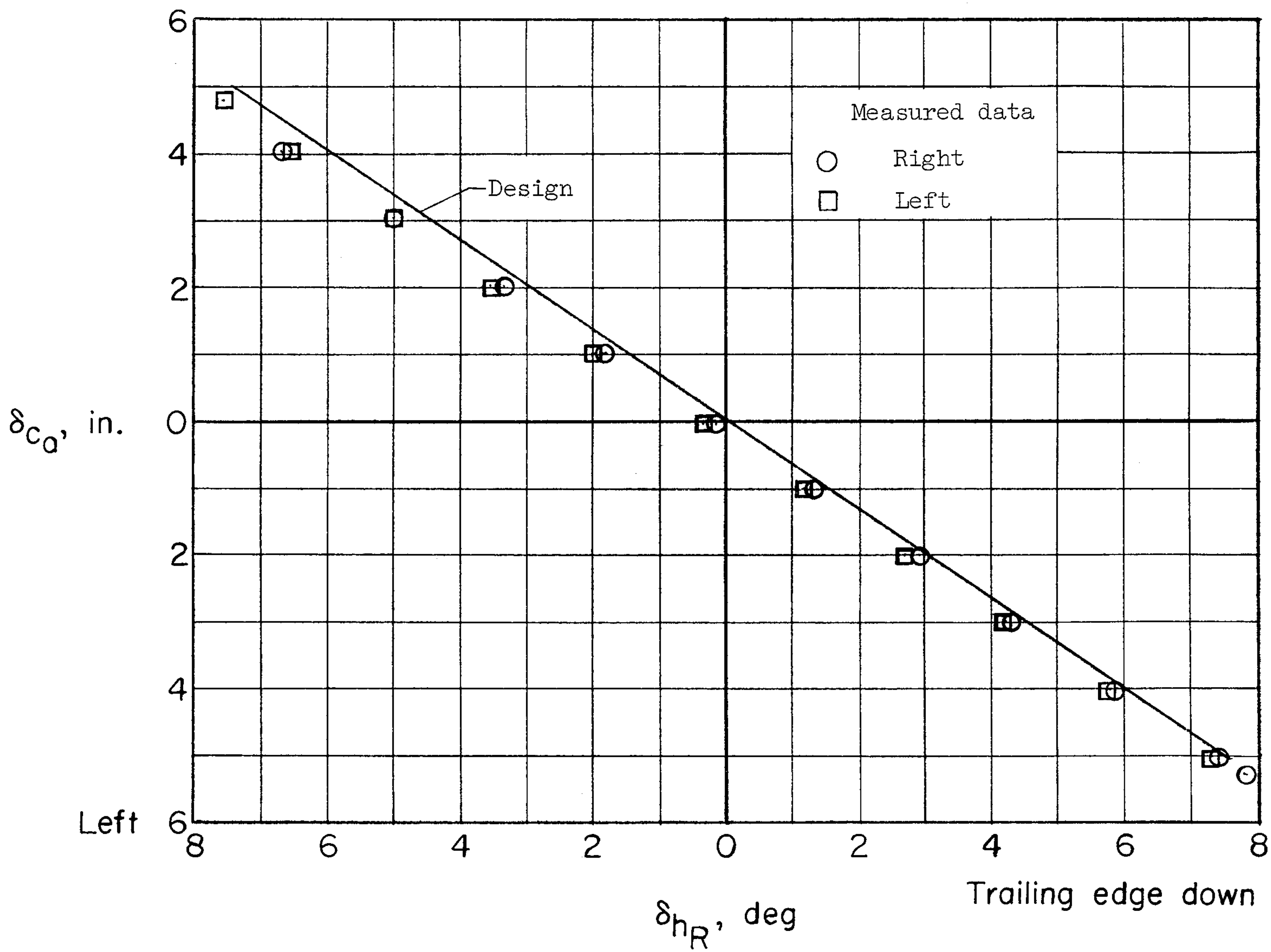
(a) Center-stick pitch displacement measured at 23.25-inch radius.

Figure 4.- X-15 basic-control-system characteristics.



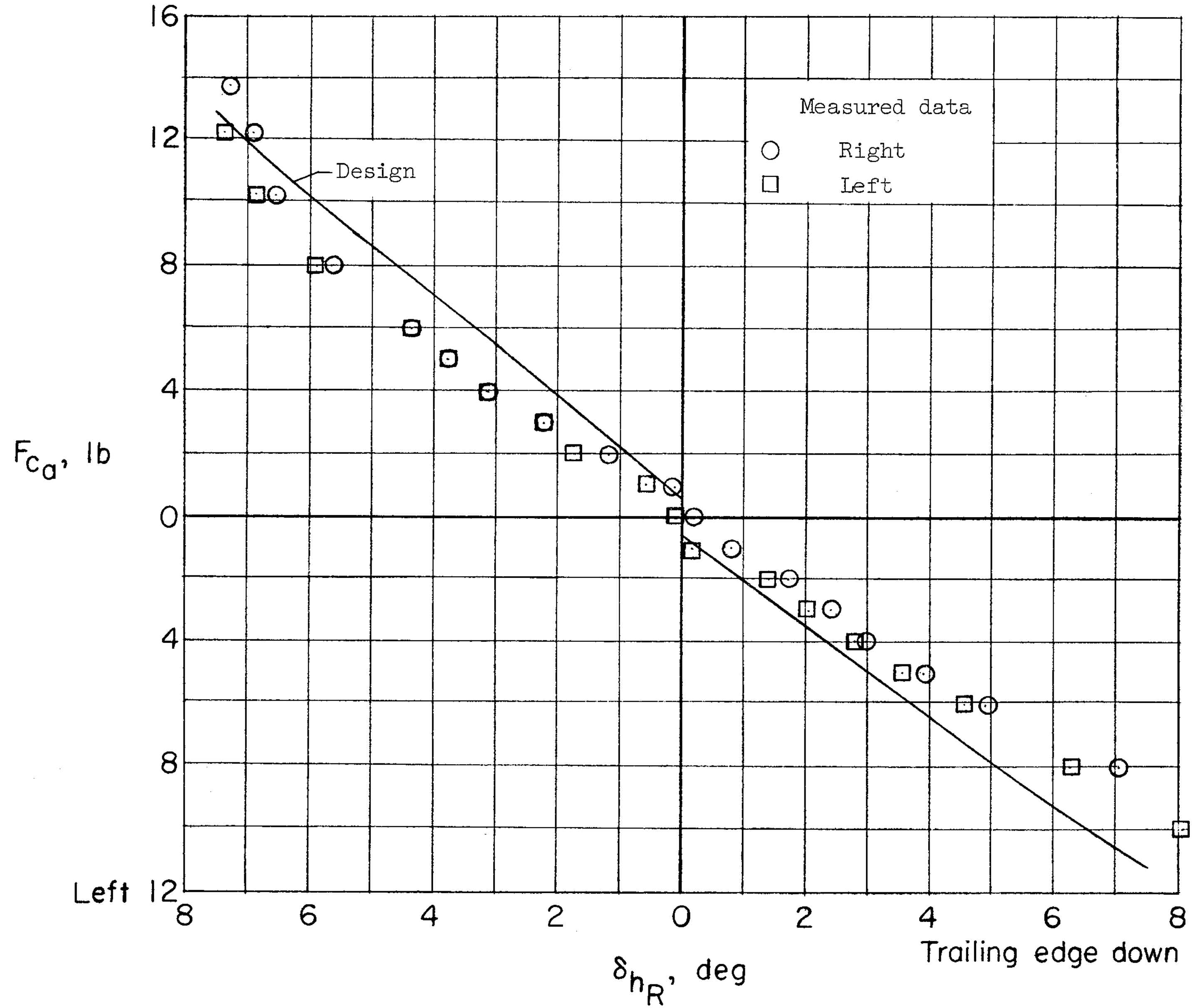
(b) Center-stick pitch force measured at 23.25-inch radius.

Figure 4.- Continued.



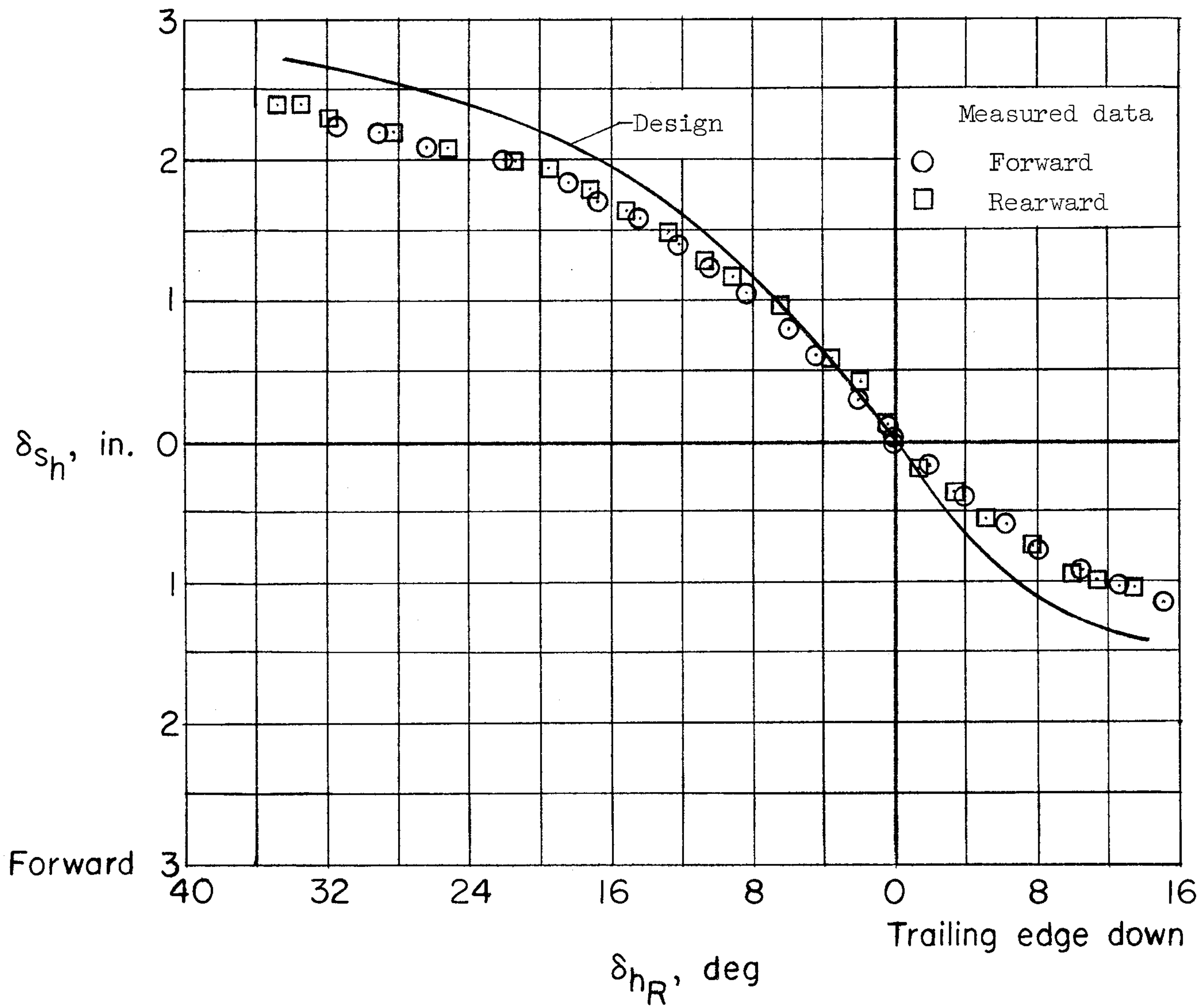
(c) Center-stick roll displacement measured at 28-inch radius, linear gearing.

Figure 4.- Continued.



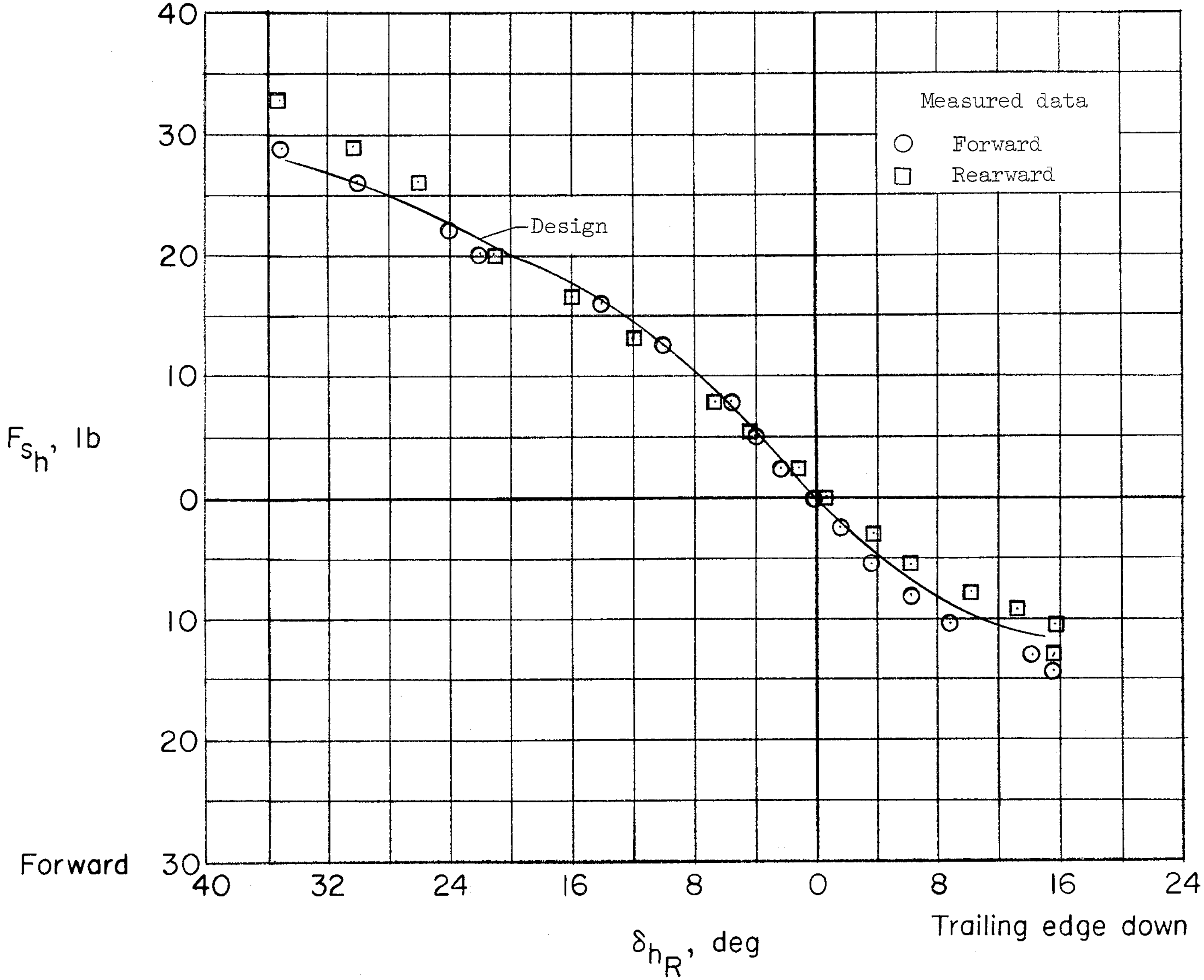
(d) Center-stick roll force measured at 28-inch radius, linear gearing.

Figure 4.- Continued.



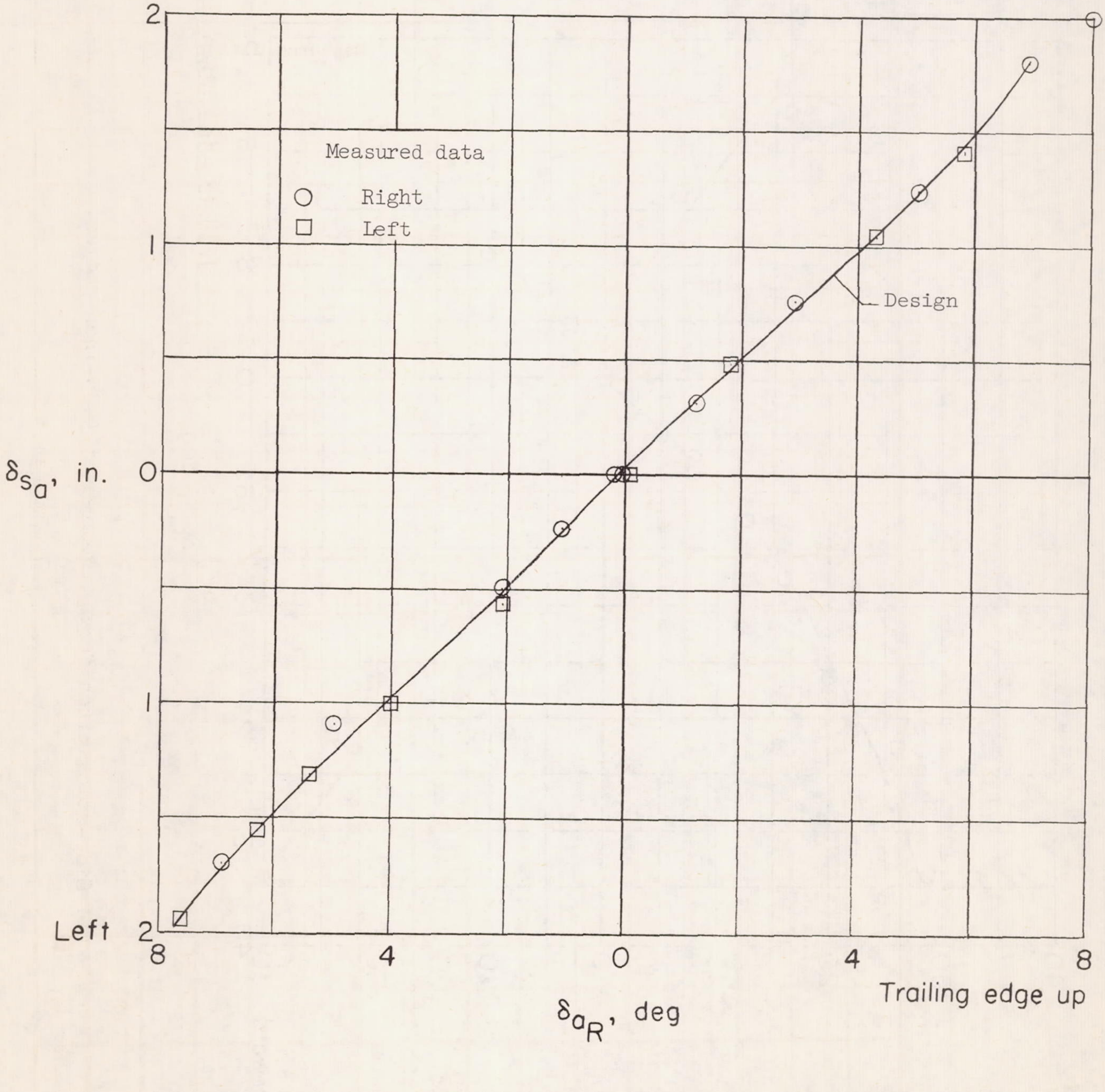
(e) Side-stick pitch displacement measured at 4.25-inch radius.

Figure 4.- Continued.



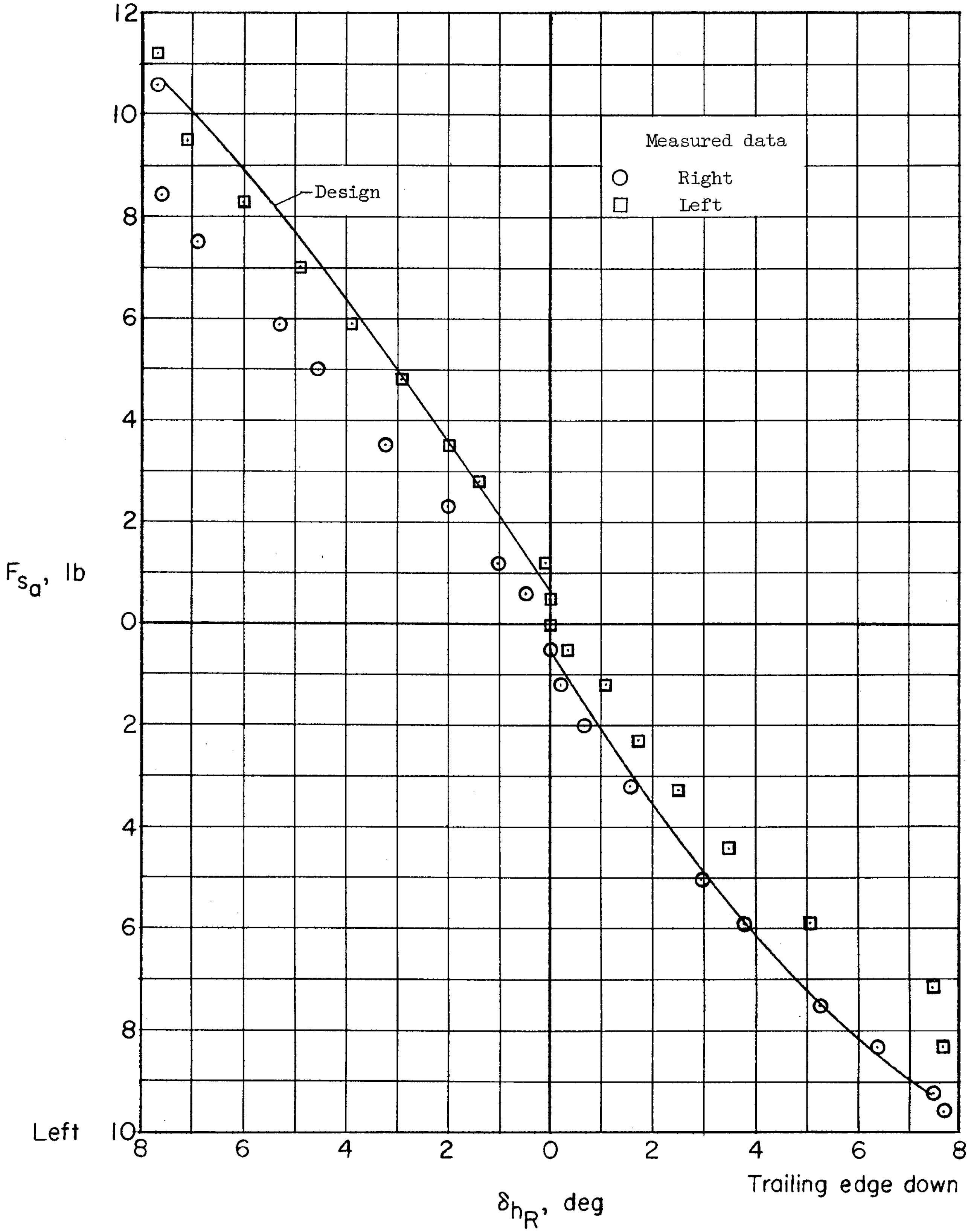
(f) Side-stick pitch force measured at 4.25-inch radius.

Figure 4.- Continued.



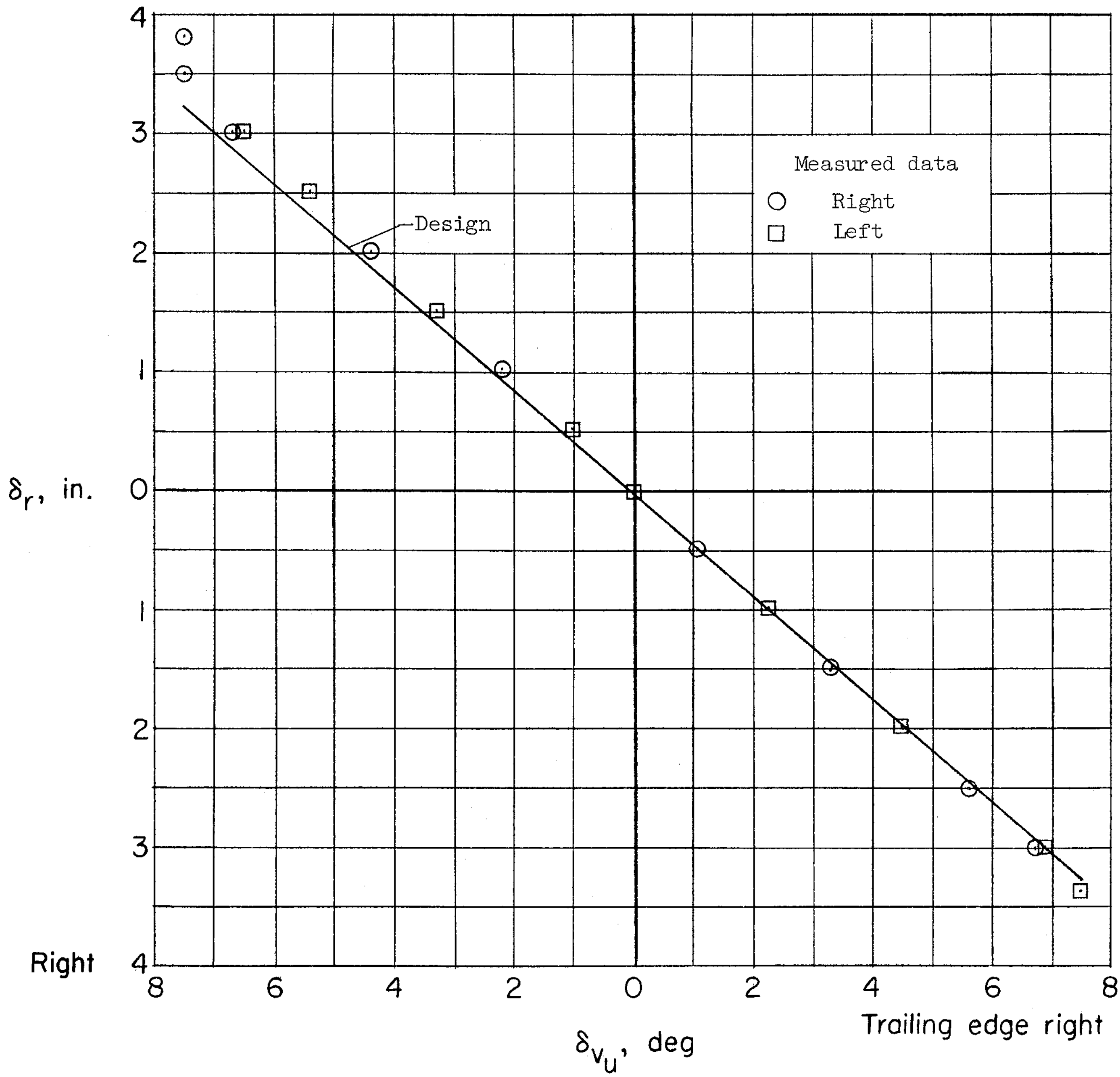
(g) Side-stick roll displacement measured at 3-inch radius.

Figure 4.- Continued.



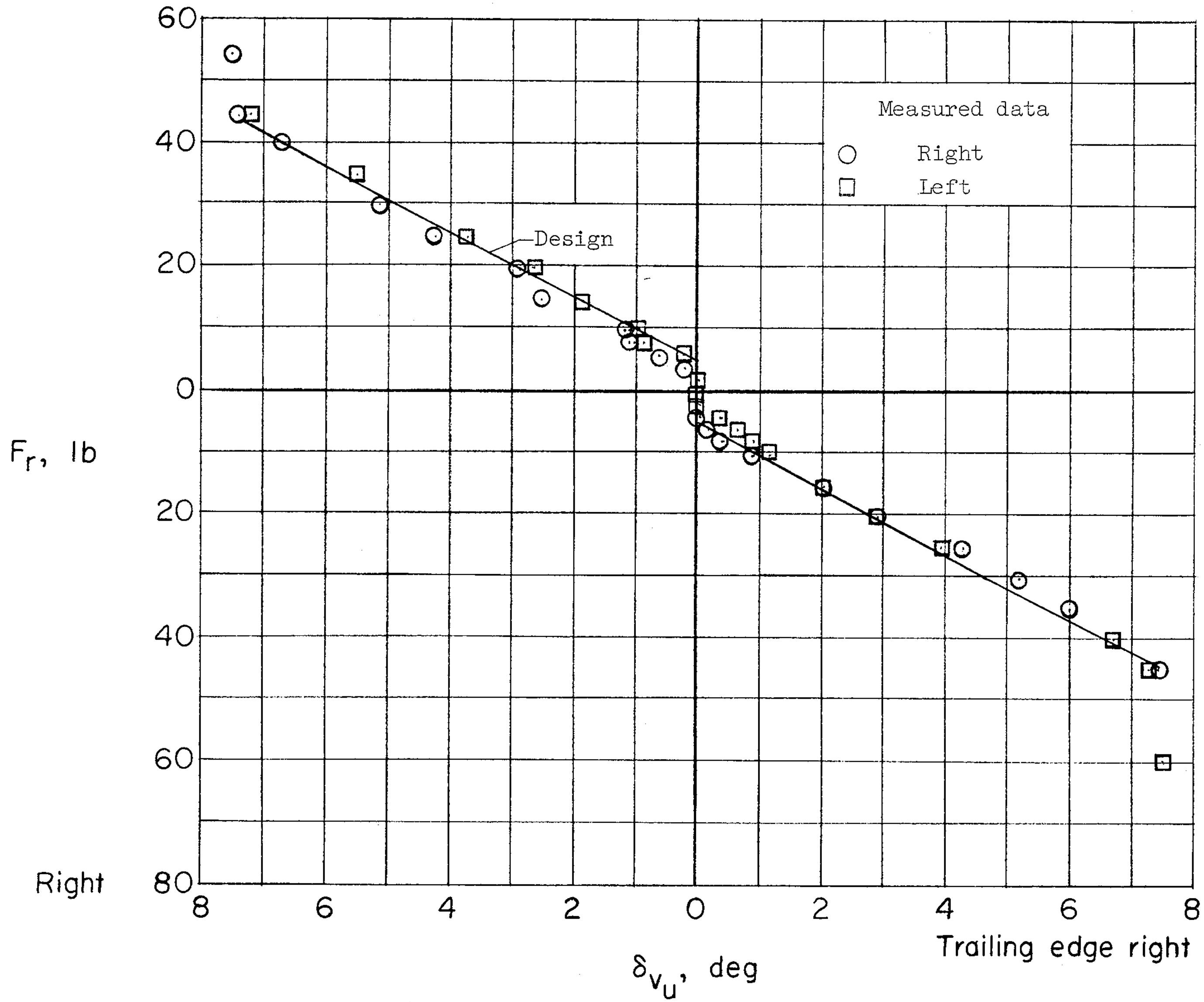
(h) Side-stick roll force measured at 3-inch radius.

Figure 4.- Continued.



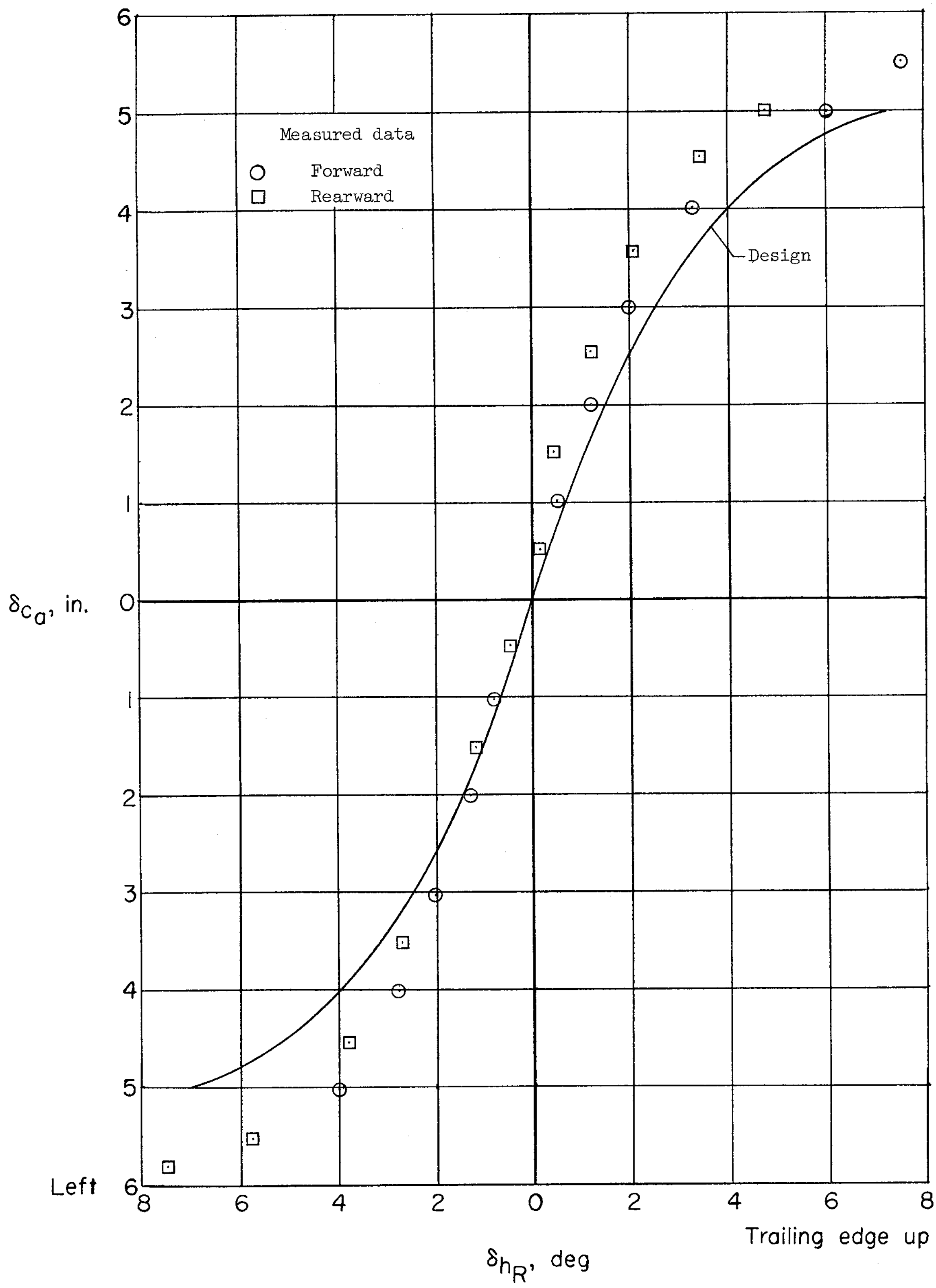
(i) Rudder-pedal displacement.

Figure 4.- Continued.



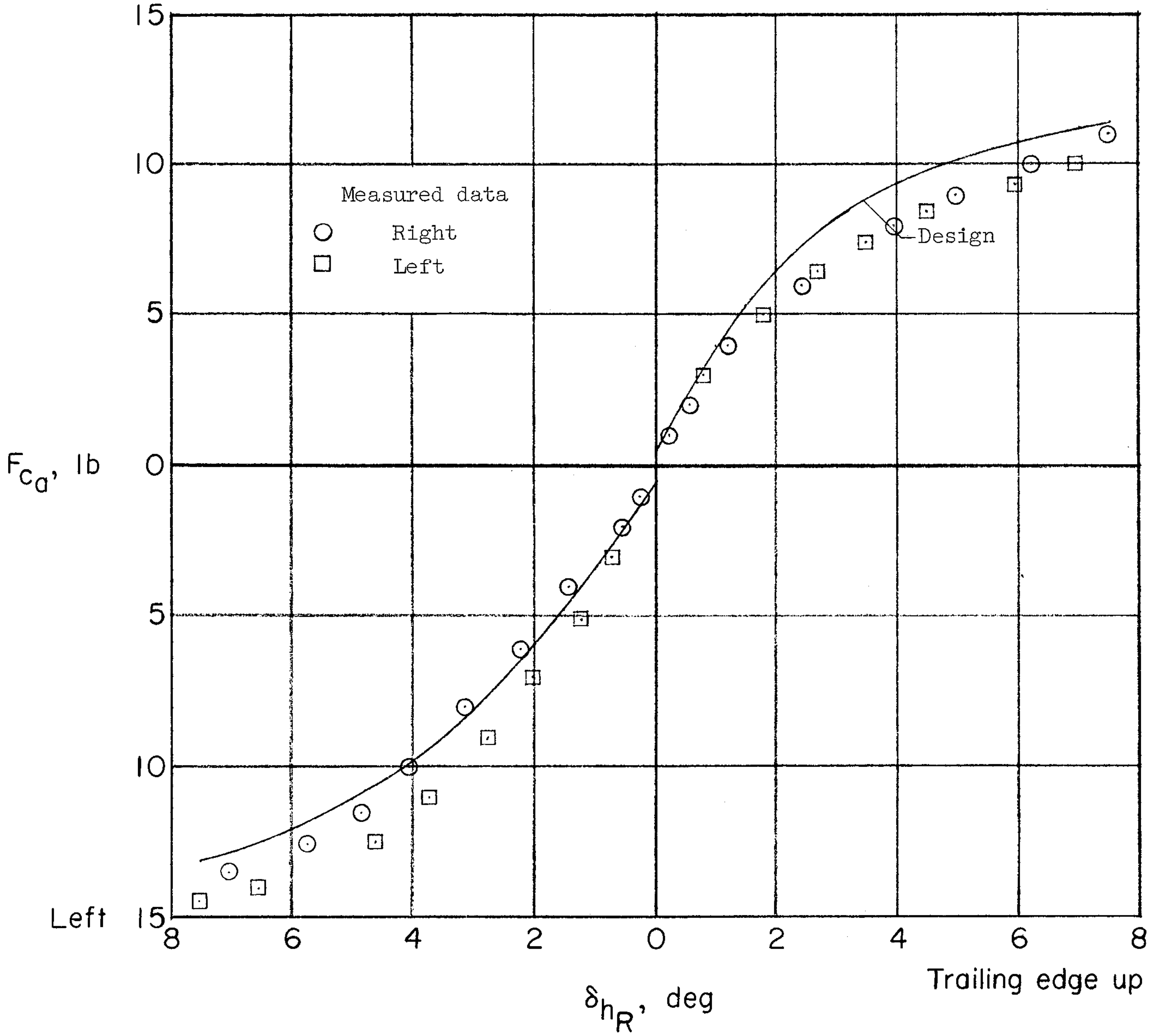
(j) Rudder-pedal force.

Figure 4.- Continued.



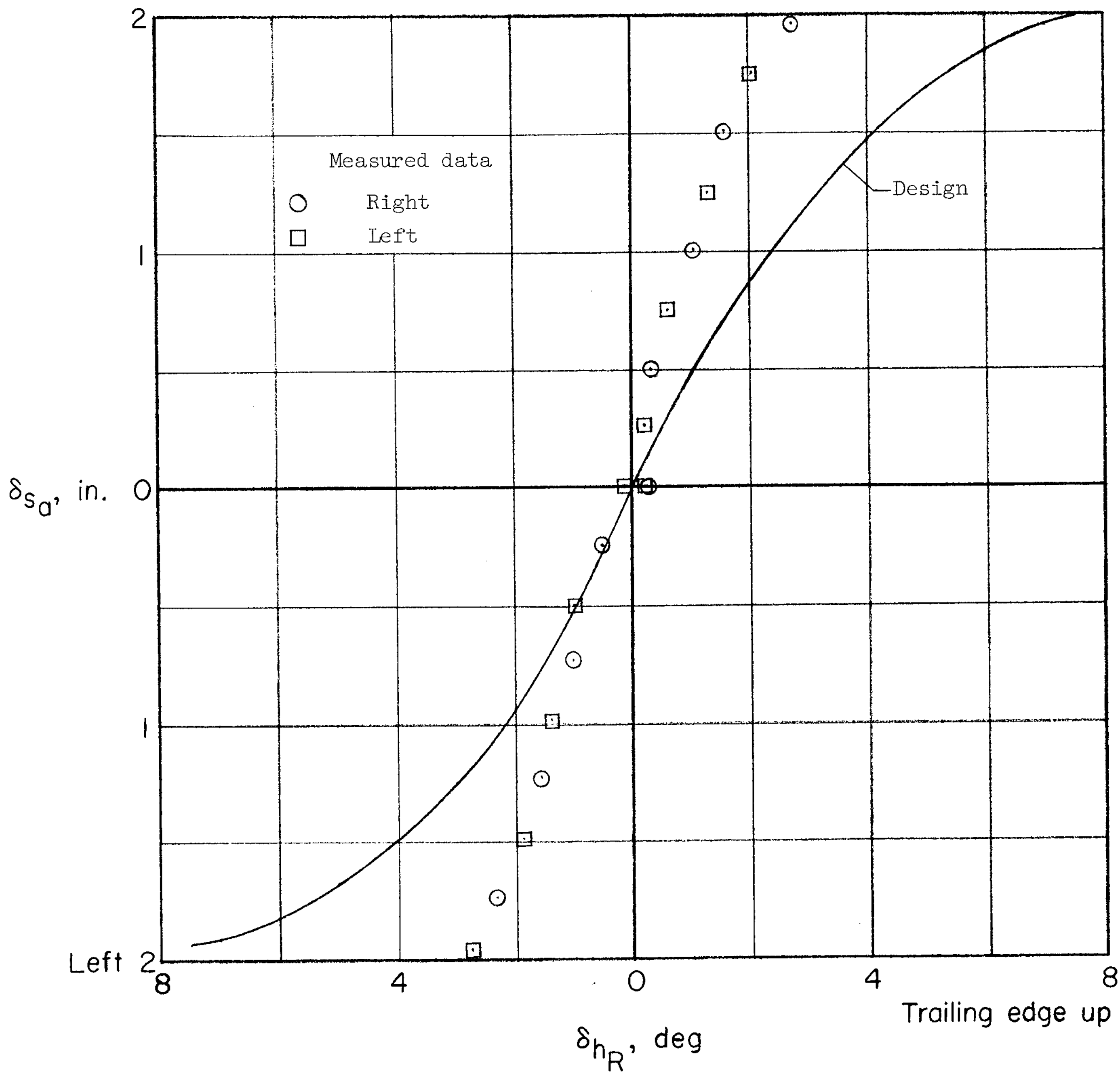
(k) Center-stick roll displacement measured at 28-inch radius, nonlinear gearing

Figure 4.- Continued.



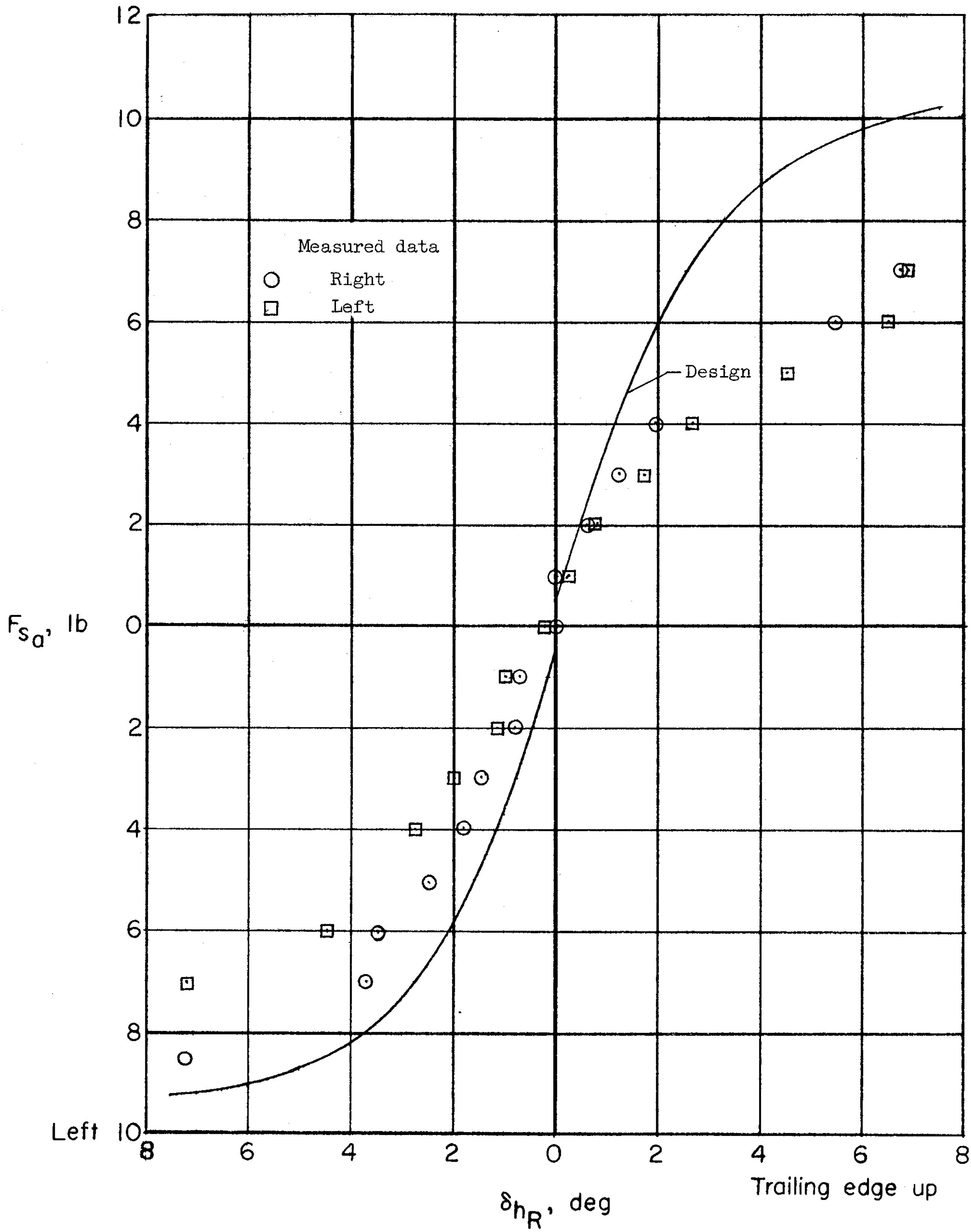
(l) Center-stick roll force measured at 28-inch radius, nonlinear gearing.

Figure 4.- Continued.



(m) Side-stick roll displacement measured at 3-inch radius, nonlinear gearing.

Figure 4.- Continued.



(n) Side-stick roll force measured at 3-inch radius, nonlinear gearing.

Figure 4.- Concluded.

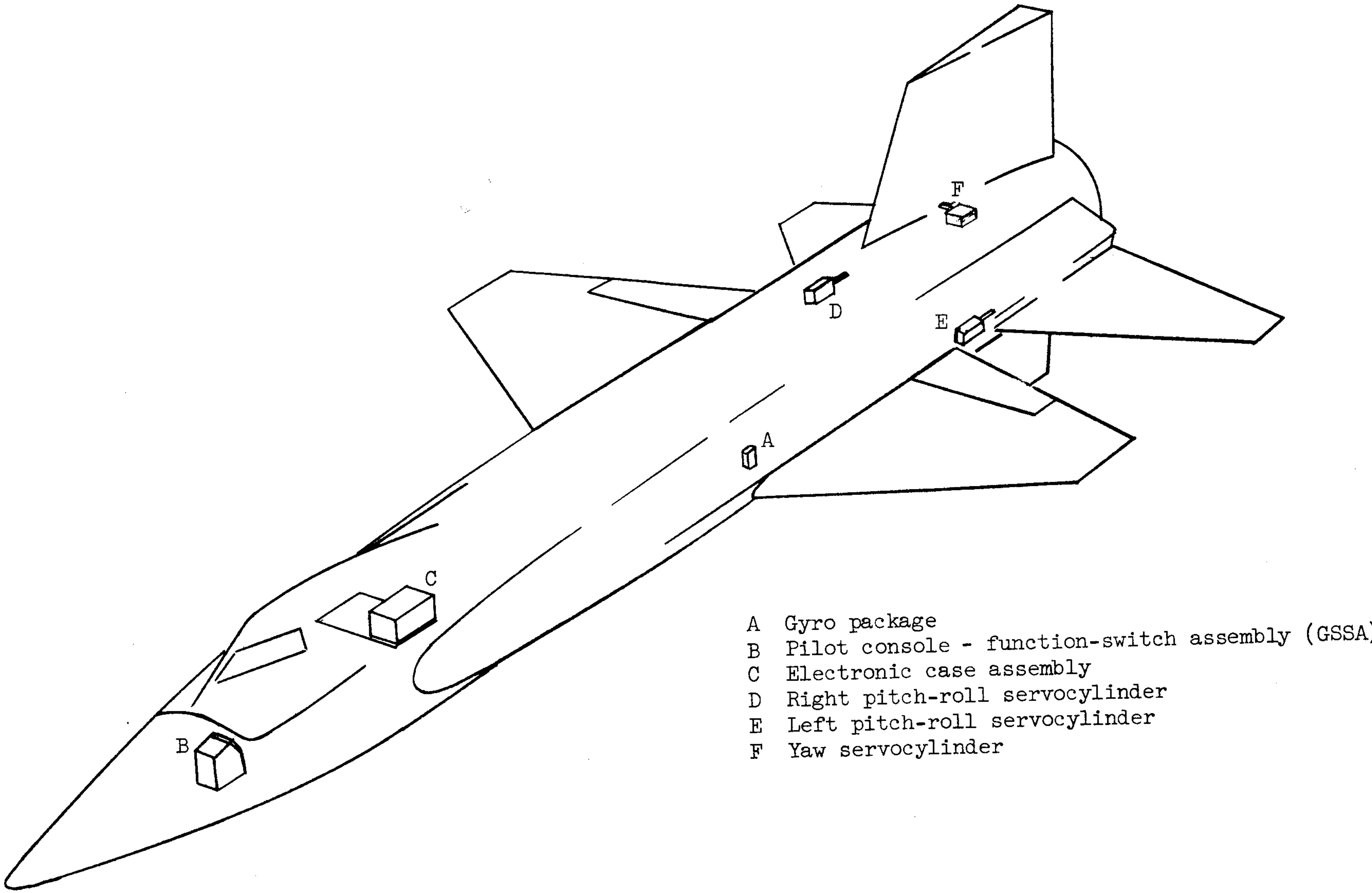


Figure 5.- Relative locations of SAS components in X-15 airplane.

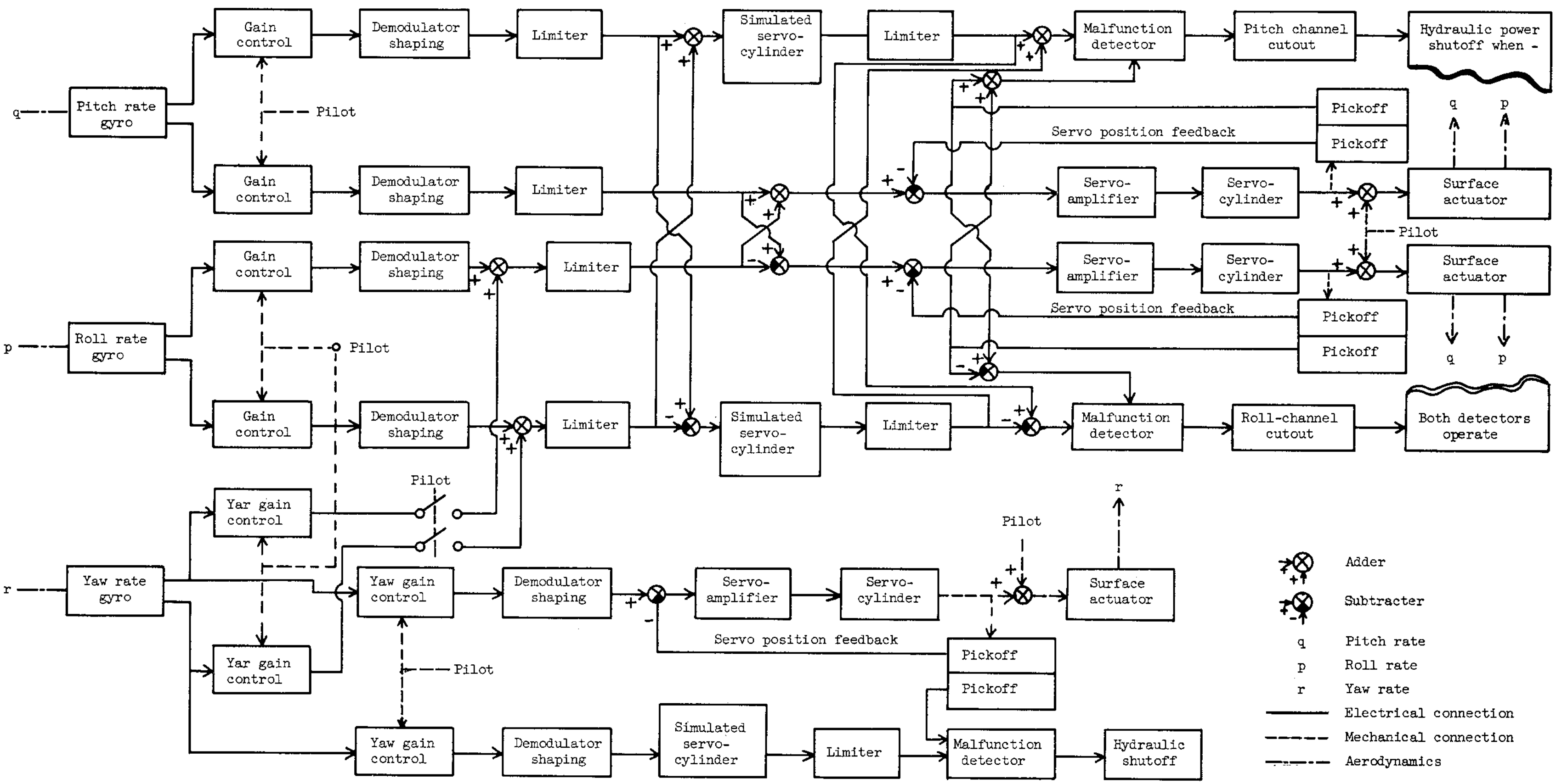


Figure 6.- Functional block diagram of the stability augmentation system.

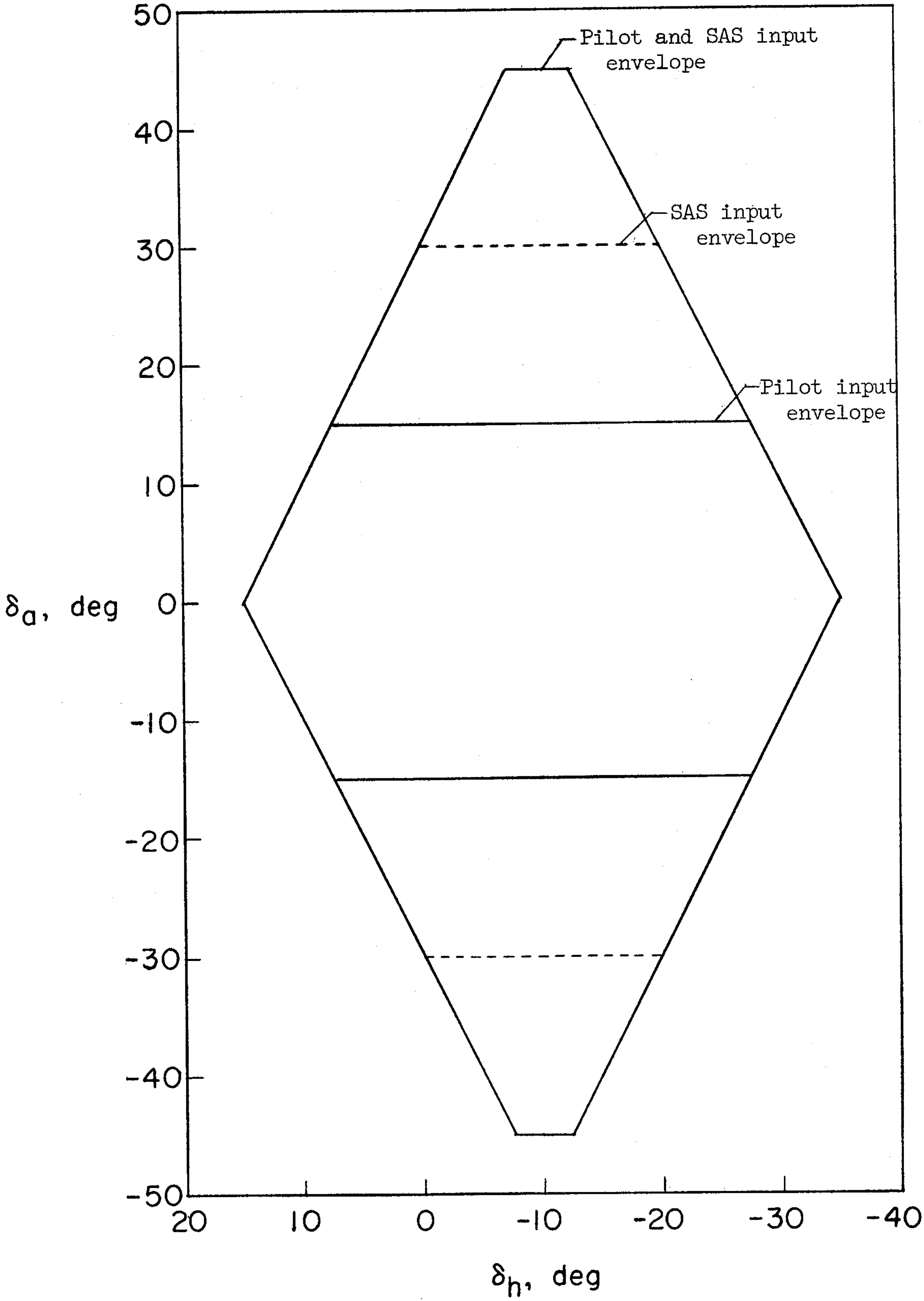
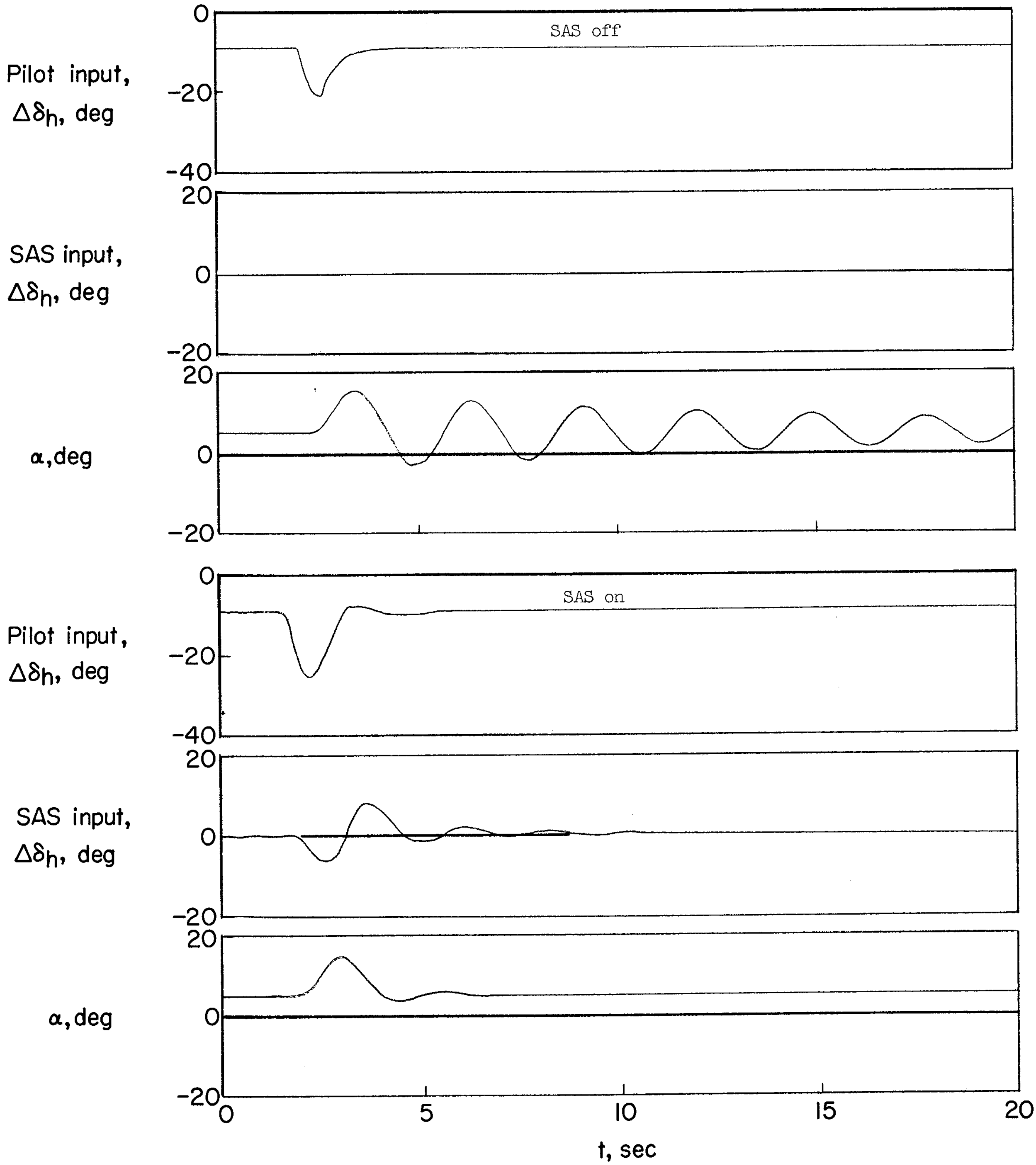
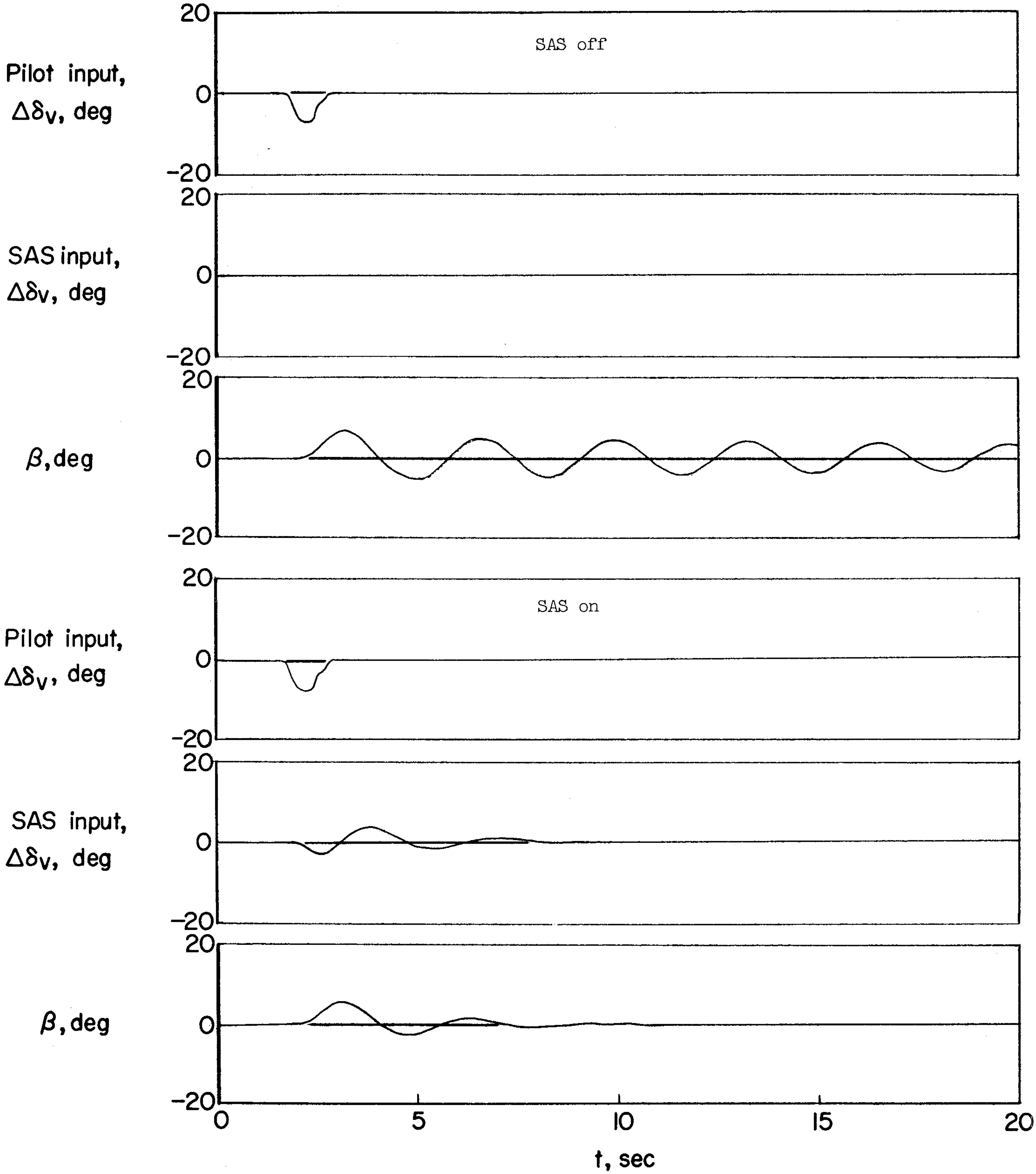


Figure 7.- X-15 horizontal-tail-surface control envelope.



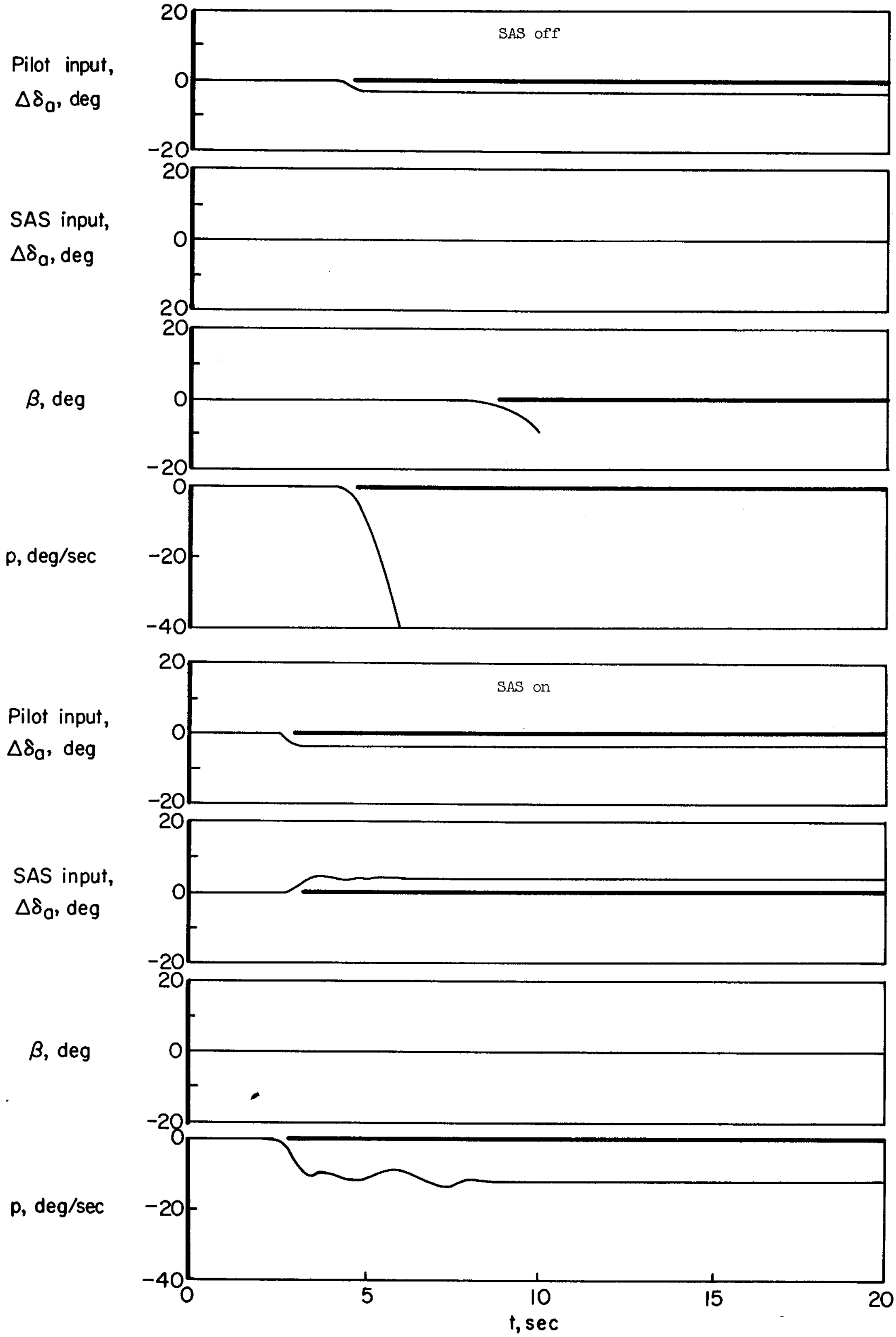
(a) Pitch. M = 3; $\bar{q}$ = 400 lb/sq ft.

Figure 8.- Comparison of transient responses in pitch, yaw, and roll with and without SAS.



(b) Yaw. $M = 3$; $\bar{q} = 400$ lb/sq ft; $\alpha = 5°$.

Figure 8.- Continued.



(c) Roll. $M = 4$; $\bar{q} = 300$ lb/sq ft; $\alpha = 5°$.

Figure 8.- Concluded.

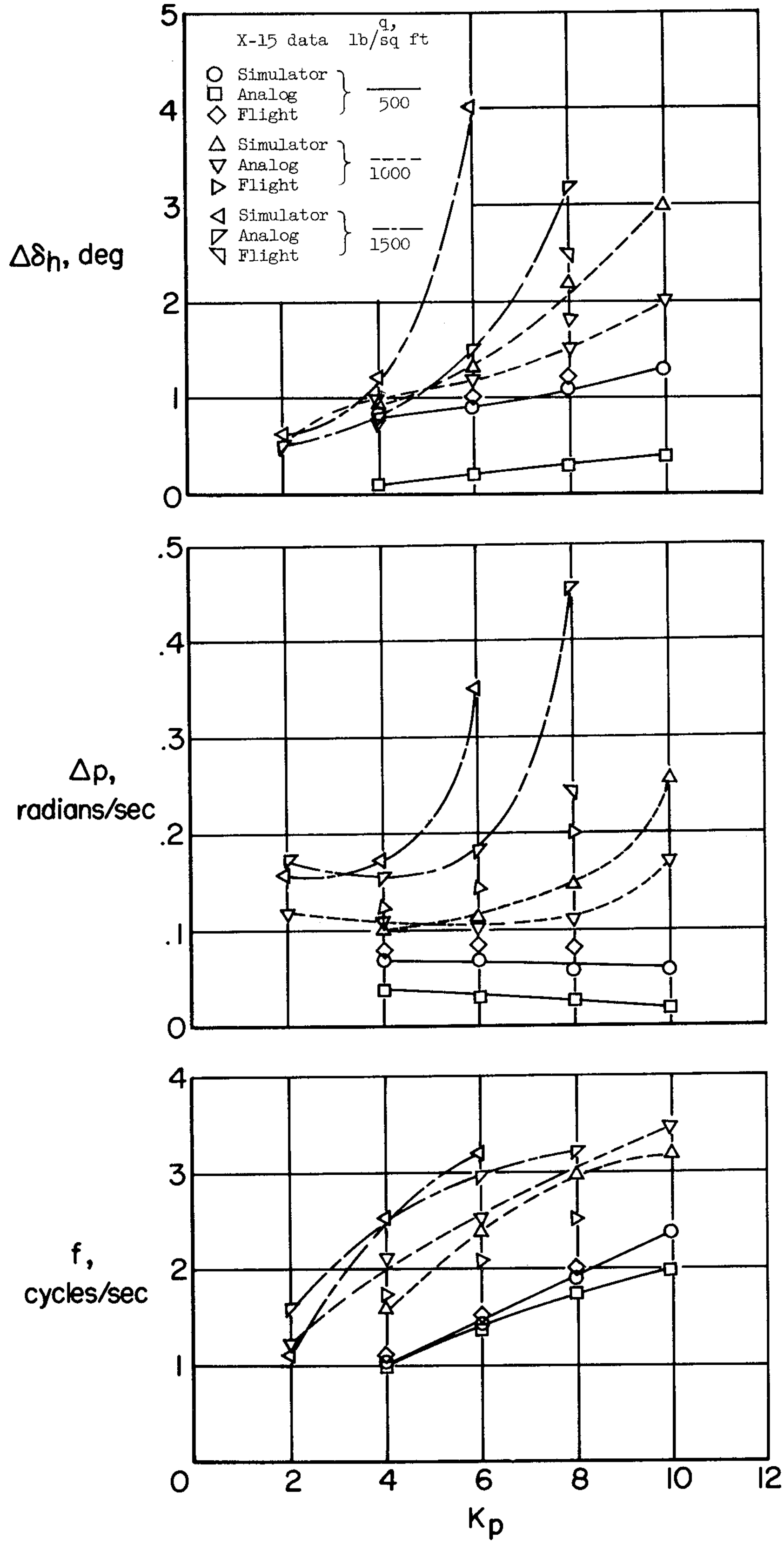


Figure 9.- Limit-cycle characteristics. $M = 2.0$; $K_q = 0$.

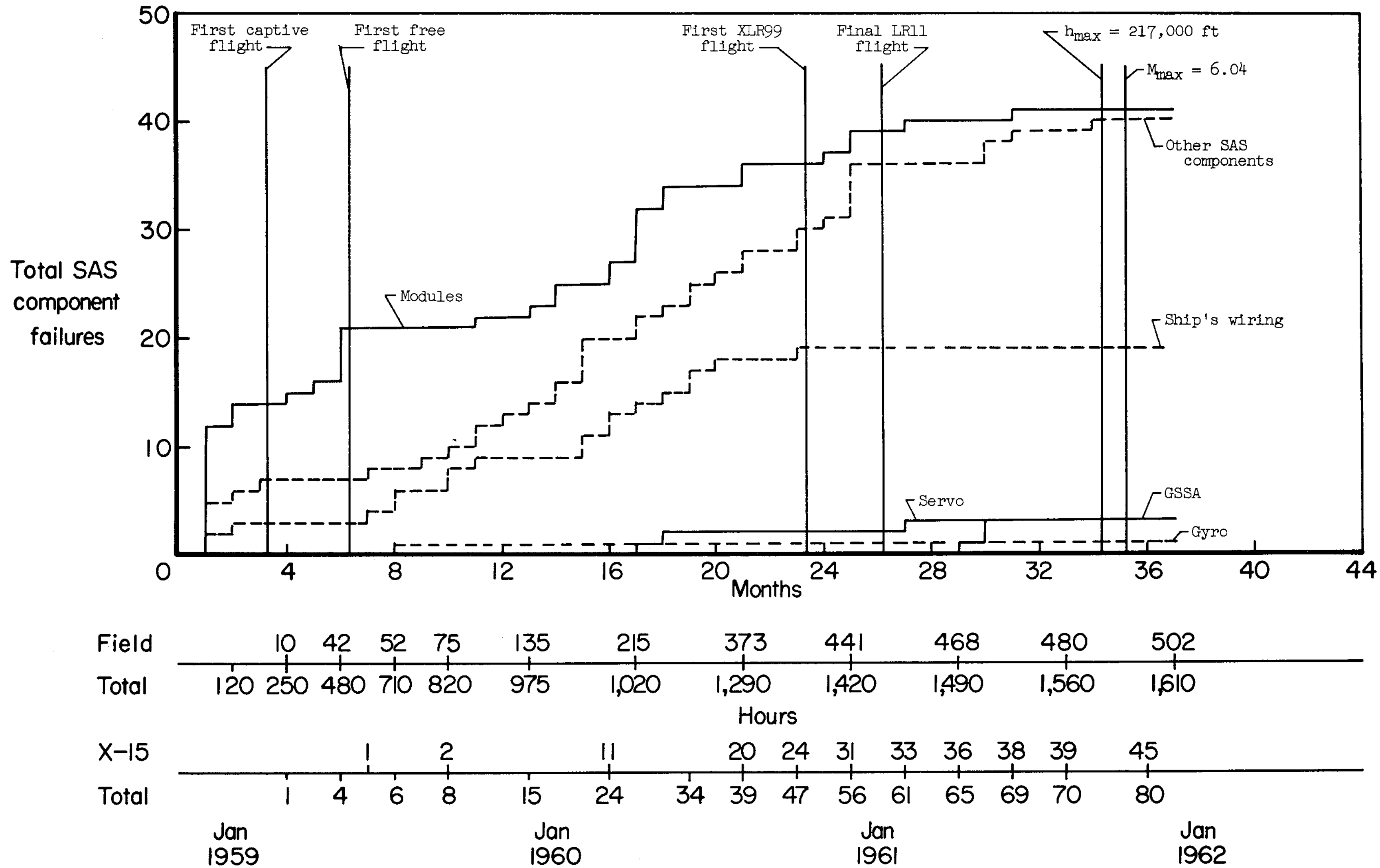


Figure 10.- Failure history of SAS components.

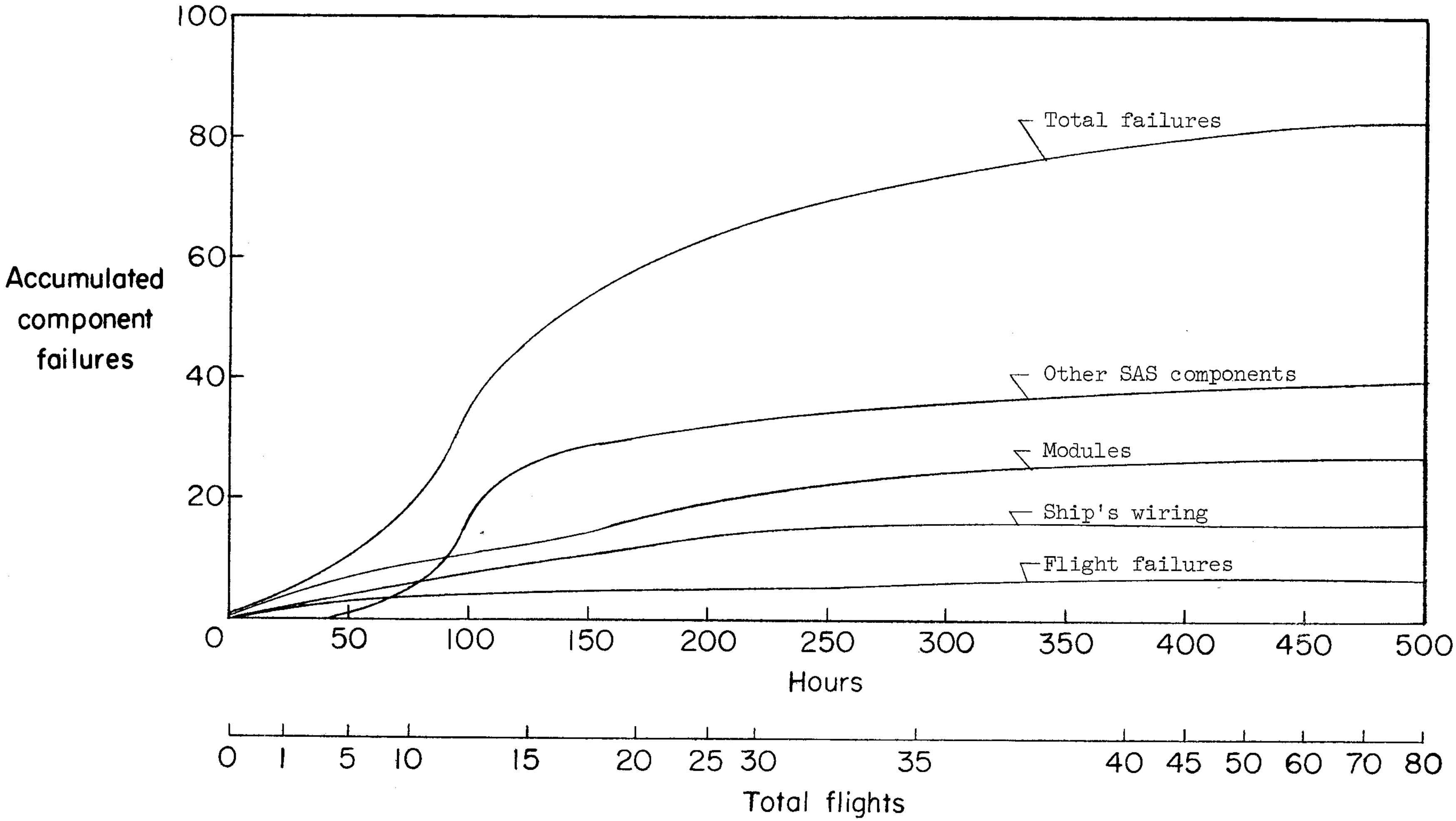


Figure 11.- SAS component failures during the field operation period.

NASA-Langley, 1962 H-288

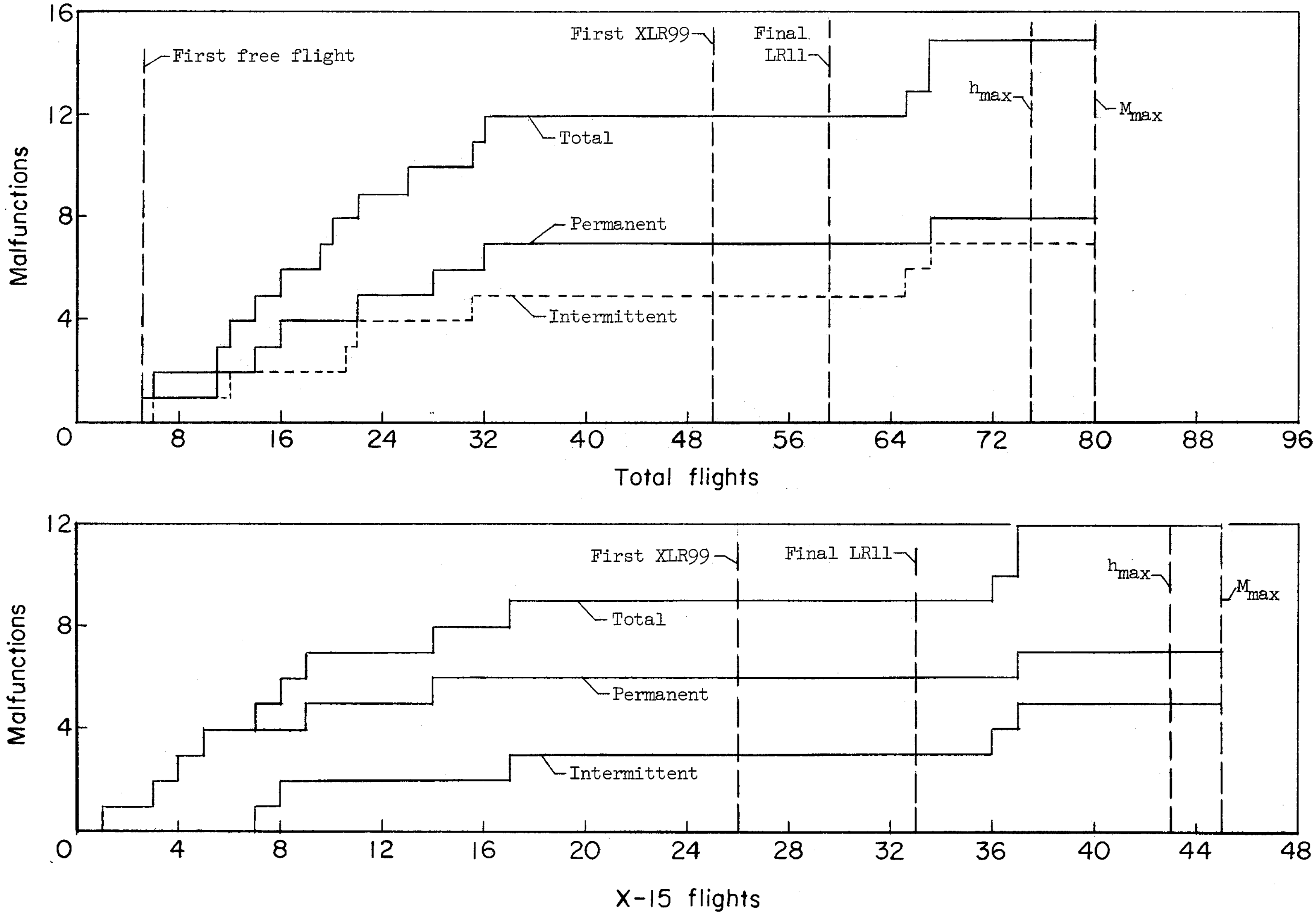


Figure 12.- SAS malfunctions affecting flight.