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MEASUREMENTS OF THE FLYING QUALITIES OF A SUPERMARINE SPITFIRE VA AIRPLANE

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Langley Field, Va.

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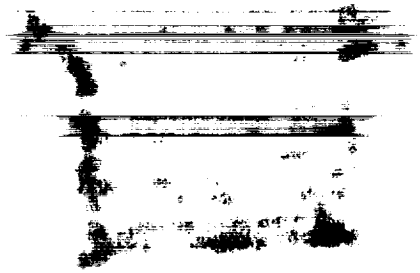

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

ADVANCE CONFIDENTIAL REPORT

MEASUREMENTS OF THE FLYING QUALITIES OF A SUPERMARINE SPITFIRE VA AIRPLANE

By William H. Phillips and Joseph R. Vensel

INTRODUCTION

The flying qualities of the Supermarine Spitfire airplane were measured at the request of the Army Air Forces, Materiel Command. These measurements form part of a program to determine quantitatively the flying qualities of many airplanes of different types. Similar tests have been carried out previously on four American types of pursuit airplanes and on one British fighter, the Hawker Hurricane. A comparison of the results of these tests should lead to a better knowledge of the flying qualities necessary in a fighter-type airplane.

The tests were conducted at Langley Field, Va., during the period from December 30, 1941 to January 29, 1942. Sixteen flights and approximately 18 hours flying time were required to complete the tests.

DESCRIPTION OF THE SUPERMARINE SPITFIRE AIRPLANE

The Supermarine Spitfire is a single-place, single-engine, low-wing, cantilever monoplane with retractable landing gear and partial-span split flaps (figs. 1, 2, 3, and 4). The general specifications of the airplane are as follows:

Name and type	Supermarine Spitfire VA (Air Ministry No. W3119)
Engine	Rolls-Royce Merlin XLV
Rating:	
Take-off	1170 brake horsepower at 3000 rpm
Normal	1200 brake horsepower at 2850 rpm at 15,500 feet
Maximum	1210 brake horsepower at 3000 rpm at 18,250 feet

Supercharger . . . . . single stage, single speed
  Supercharger gear ratio 9.10:1
Propeller . . . . . Rotol constant speed
  Diameter . . . . . 10 feet, 10 inches
  Number of blades 3
  Gear ratio 0.477:1
Fuel capacity . . . . . 85 gallons (imperial)
Oil capacity . . . . . 5.8 gallons (imperial)
Weight, empty . . . . . 4960 pounds
Normal gross weight . . . . . 6237 pounds
Weight as flown for tests . . . . . 6184 pounds
Wing loading, normal gross weight . . . 25.8 pounds per square foot
Power loading, normal gross weight . . . . 4.78 pounds per horsepower
Over-all height (datum-line level) . . 10 feet, 11 inches
Over-all length . . . . . 29 feet, 11 inches
Wing:
  Span . . . . . 36 feet, 11 inches
  Area . . . . . 242 square feet
  Airfoil section root . . . . . NACA 2212
  Airfoil section tip . . . . . NACA 2208
Aspect ratio 5.62
Mean aerodynamic chord . . . . . 7 feet, 1 inch
Location of mean aerodynamic chord
  (approx.). . . 4.8 inches back of leading-edge wing root
    Plan form . . . . . elliptical
    Dihedral (leading edge of wing) . . . . . 6.0°
    Incidence measured from thrust axis:
      Root . . . . . 2.1°
      Tip . . . . . 0°
Wing flaps (split trailing-edge type):
    Total area . . . . . 15.6 square feet
    Flap span . . . . . 17 feet, 10 inches
    Travel . . . . . 85°
Ailerons (metal-covered):
    Length (each). . . . . 6 feet, 10¼ inches
    Area (total area, each). . . . . 9.45 square feet
    Balance area (each) . . . . . 2.45 square feet
Stabilizer (fixed):
    Maximum chord . . . . . 2 feet, 6.2 inches
    Area (including 2.15 sq ft fuselage) 20.1 square feet
    Incidence from thrust axis . . . . . 0°
Elevator:
    Span . . . . . 10 feet, 6 inches
    Maximum chord . . . . . 1 foot, 6.2 inches
    Area (aft hinge line, except for horn balance) . . . . . 13.26 square feet
    Trim tab area . . . . . 0.84 square foot
    Balance area (horn balances) . . . . 1.16 square feet

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Vertical fin:	
Area	4.61 square feet
Offset	0°
Rudder:	
Vertical span	5 feet, $4\frac{1}{2}$ inches
Maximum chord (aft hinge line) . . .	1 foot, $9\frac{3}{4}$ inches
Total area	8.38 square feet
Balance area (horn balance).	0.34 square foot
Trim tab area	0.35 square foot
Distance from elevator hinge line to leading edge of wing	21 feet, $10\frac{1}{2}$ inches
Distance from rudder hinge line to leading edge of wing	22 feet, 4 inches
Maximum fuselage cross-sectional area (excluding radiator) approximate	10.8 square feet

The relation between the control-stick position and the angles of the controls is shown on figures 5 and 6. Figure 5 also shows the unbalance and friction in the elevator system as measured with the airplane on the ground. A stick force of 2 pounds to the right and 3 pounds to the left was required to overcome aileron friction. The friction in the rudder linkage varied from 7 pounds near the neutral position to 20 pounds near the limits of the rudder travel.

INSTRUMENT INSTALLATION

Items measured	NACA instruments
Time	timer
Airspeed	airspeed recorder
Positions of the three control surfaces	control-position recorder
Rolling velocity	angular-velocity recorder
Normal, longitudinal, and lateral acceleration . . .	three-component accelerometer
Angle of sideslip	recording yaw vane
Angle of bank or pitch	recording inclinometer
Rudder or elevator force	control-force recorder

The airspeed recorder was connected to a swiveling pitot-static head, which was free to rotate in pitch but not in yaw, located on a boom extending a chord length

ahead of the right wing tip. The yaw vane was located at the end of a similar boom on the left wing tip, as shown on figure 4. It was believed that angularity of the flow at this point might cause some error in the recorded sideslip angles. For this reason, another recorder was mounted on the right wing tip and the angles of the two yaw vanes were recorded simultaneously in flight throughout the speed range under various flap and power conditions. Because of symmetry of the airplane, one-half the difference between the readings of the two yaw vanes was taken to represent the correction to apply to each yaw vane. This correction has been applied to all of the recorded values of sideslip angle. These values are therefore believed to represent the actual angles of sideslip of the thrust axis. The difference between the readings of the two yaw vanes was about 3° with level flight power and 2° with power off. The vanes showed the flow to be converging toward the fuselage.

All the recording instruments were synchronized by the timer and the records were obtained photographically. Elevator and rudder forces were determined by recording the tension in the control cables. Aileron forces were measured by means of a visual control-force indicator that rested against the top of the control stick.

The instrument recording the angular position of the three control surfaces was attached to the control linkages near the cockpit. Tests made on the ground showed that errors in the recorded angles due to stretch in the control system were small enough to be negligible in the case of the elevator and rudder controls. A slight amount of flexibility was noticeable in the aileron system but, inasmuch as no simple means was available for determining the error introduced, no correction was applied to the recorded aileron angles.

AIRSPEED CALIBRATION

The readings of the pilot's meter as compared to the correct indicated airspeed in the cruising, gliding, and landing conditions of flight are plotted on figure 7. The correct speed was determined by flying in formation with another airplane. The calibration of the airspeed recorder in the latter airplane was made by the use of a trailing airspeed head. The installation of the airspeed indicator in the Spitfire consisted of a pitot-static head

located below the left wing slightly ahead of the aileron hinge, as shown on figure 4. The installation gave almost correct measurements at high speeds but showed a speed about 10 miles per hour too low near minimum speed. In addition to this error, the reading was affected by the angle of sideslip. The indicator read too low a speed in left sideslips and too high a speed in right sideslips. This fact was determined by comparison of the pilot's indicated speed in sideslips with that recorded by the pitot head located on the boom ahead of the right wing tip and depends on the assumption that the airspeed head on the boom was affected in the same way by sideslip to either side.

TESTS, RESULTS, AND DISCUSSION

All of the flying-qualities tests were made with the center of gravity at a distance of 31.4 inches behind the leading edge of the wing at the root. The mean aerodynamic chord of 85 inches was computed to be 4.80 inches back of the leading edge of the wing at the root. The center of gravity was therefore at 31.4 percent of the mean aerodynamic chord. Because no accurate drawings of the Spitfire were available, the calculated location of the mean aerodynamic chord may be somewhat in error.

The center-of-gravity location with full military load is not known. The airplane, however, as weighed with a 140-pound pilot and all known items of military equipment except ammunition in place had a weight of 6014 pounds and a center-of-gravity location 31.1 inches behind the leading edge of the wing. The addition of ammunition is not believed to change this center-of-gravity location appreciably. The weight of the airplane as flown in the tests with instruments and ballast added to retain the desired center-of-gravity position was 6184 pounds.

Longitudinal Stability and Control

Characteristics of uncontrolled longitudinal motion.- Of the two types of control-free oscillation, only the short-period oscillation is dealt with here, as previous tests have shown that the characteristics of the well-known long-period (phugoid) oscillation have no correlation with the handling qualities of an airplane. The degree of damping of the short-period oscillation was in-

vestigated by suddenly deflecting the elevator and releasing it in high-speed level flight. The subsequent variation of elevator angle, elevator force, and normal acceleration was recorded. A typical time history of this maneuver is shown in figure 8. The variation of elevator angle and normal acceleration completely disappeared after one cycle, and thereby satisfied the requirement of reference 1. The oscillation was satisfactorily damped in spite of the fact that the mass unbalance of the elevator shown on figure 5 would be expected to reduce the damping.

The longitudinal handling characteristics of the Spitfire were observed to be poor in rough air. This behavior was attributed to the airplane's neutral static longitudinal stability and relatively light wing loading, rather than to the characteristics of its control-free short-period oscillation.

Characteristics of the elevator control in steady flight.- The static longitudinal stability of the Supermarine Spitfire airplane was measured by recording the control forces and positions in steady flight at various speeds in the following conditions:

Condition	Manifold pressure (in. Hg)	Engine speed (rpm)	Flap position	Landing-gear position	Radiator-shutter position	Hood position
Take-off	44 (7 lb/sq in. boost)	2850	up	down	open	open
Climbing	44 (7 lb/sq in. boost)	2850	up	up	open	closed
Cruising	37.5 ($3\frac{3}{4}$ lb/sq in. boost)	2650	up	up	flush	closed
Gliding	throttle closed	----	up	up	closed	closed
Landing approach	22 (-4 lb/sq in. boost)	2300	down	down	open	open
Landing	throttle closed	----	down	down	closed	open

The results of these tests are presented in figures 9, 10, and 11. The conclusions regarding the elevator control characteristics in steady flight may be summarized as follows:

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1. The stick-fixed longitudinal stability in the gliding condition was neutral, as shown by the fact that no change in elevator deflection was required to trim throughout the unstalled speed range. The stability was essentially neutral in all flap-up, power-on conditions of flight except at low speeds, where some rearward motion of the stick occurred. This apparent positive stability at low speeds with power on while still well above the stall was caused by the elevator deflection due to sideslip, because some left sideslip was found to occur at low speeds in straight power-on flight with the wings level. Scatter of the points in the plotted data may likewise be attributed to an inconsistent variation of sideslip angle with airspeed. The variation of elevator angle with sideslip will be further discussed under the subject of pitching moment due to sideslip. As the curves of figure 11 show, the Spitfire displayed stick-fixed instability in the flap-down conditions of flight with power on or off. It is concluded that in all flight conditions, the Spitfire failed to meet the requirements for satisfactory longitudinal stability stated in reference 1. The upward travel of the elevator in the power-off conditions near minimum speed resulted from decreased downwash at the tail caused by separation of the flow at the wing root. This phenomenon is explained in the report on stalling characteristics (reference 2).

A similar increase in elevator angle was required in the power-on conditions of flight near minimum speed. No separation of flow from the wing root was observed in these conditions, but the elevator deflection due to sideslip is sufficient to account for this elevator motion. It is probable that separation of flow over the yawed fuselage was responsible for the large up-elevator angles required in sideslips.

2. In spite of the neutral stick-fixed static stability with flaps up, the airplane had a slightly stable stick-force variation with airspeed throughout the speed range (fig. 9). This stable stick-force gradient is attributed to the unbalanced elevator. If a completely mass-balanced elevator had been employed, the stick-force variation would have been slightly unstable, a condition consistent with neutral stick-fixed stability. In the flap-down condition with power on, the stick-force variation for the trim-tab setting used was unstable and, with power off, the stick-free stability was neutral. If the airplane had been trimmed for zero stick force at low speed, the variation might have been slightly stable with

power off and about neutral with power on. In all conditions an increased pull force on the stick was required near the minimum speed. This increased pull force served as a desirable stall warning. It was associated with the separation of flow at the wing root and the up-elevator angles required at the stall.

3. The friction in the elevator system was such that a force of 2 pounds was required to reverse the motion of the stick, as shown in figure 5. This friction was small enough that, in the flight conditions where a stable stick-force gradient existed, the control would return to its trim position.

The effect of friction is not shown on the force curves of figures 9, 10, and 11, because the vibrations of the airplane largely eliminated the frictional force while the measurements were being taken.

4. The limits of elevator motion were not reached in steady flight from the minimum speed to the highest speeds tested. Figures 9, 10, and 11 show that in all conditions only a few degrees of elevator motion were required to trim throughout the speed range.

Characteristics of the elevator control in accelerated flight.- The characteristics of the elevator control in accelerated flight were determined from measurements taken in pull-ups and in turns. The data obtained in pull-ups are presented in figure 12. Time histories of representative turns are shown in figures 13 to 20.

The elevator control was found to be powerful enough to develop either the maximum lift coefficient or the allowable load factor at any speed. As shown in figure 12(a), less than 5° movement of the elevator was used in reaching maximum lift coefficient in pull-ups from level flight. In these maneuvers, the elevator was abruptly deflected a small amount and then held fixed with the aid of a graduated tape in the cockpit. In pull-ups made at high speed, the elevator was always eased forward before maximum acceleration had been developed, in order to avoid overloading the structure.

The normal acceleration was observed to increase progressively with elevator angle, though the range of elevator motion was so small that no measurements were made of the exact form of this variation.

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The small elevator travel required to reach maximum lift coefficient was evident in turns as well as in pull-ups. The variation of elevator angle with lift coefficient in turns is plotted in figure 21(a). Only 3° up-elevator movement was required to go from level flight at a lift coefficient of about 0.3 to the first sign of the stall. This movement corresponds to a stick deflection of 3/4 inch. This degree of stability is far lower than the 4 inches of rearward stick movement required in reference 1.

The Spitfire airplane had the unusual quality that allowed it to be flown in a partly stalled condition in accelerated flight without becoming laterally unstable. Violent buffeting occurred, but the control stick could be pulled relatively far back after the initial stall flow breakdown without causing loss of control. With the gun ports open, lateral instability in the form of a right roll occurred, but not until an up-elevator deflection of 10° had been reached and unmistakable warning in the form of buffeting had occurred. This subject is discussed more fully in reference 2.

The excellent stall warning made it easy for the pilots to rapidly approach maximum lift coefficient in a turn so long as the speed was low enough to avoid undesirably large accelerations at maximum lift coefficient.

The excellent stall warning possessed by the Spitfire was obtained at the expense of a high maximum lift coefficient. The maximum lift coefficient in accelerated flight was 1.21, while the average lift coefficient throughout a stalled turn was usually about 1.10.

In turns at speeds high enough to prevent reaching maximum lift coefficient because of the excessive accelerations involved, the small static longitudinal stability of the Spitfire caused undue sensitivity of the normal acceleration to small movements of the stick. As shown by the time histories of high-speed turns (figs. 15 to 18), it was necessary for the pilot to pull back the stick and then ease it forward almost to its original position in order to enter a turn rapidly without overshooting the desired normal acceleration. Although this procedure appears to come naturally to a skillful pilot, flight records from other airplanes show that a turn may be entered rapidly and the desired normal acceleration may be held constant by a single rearward motion of the stick provided

the static stability of an airplane is sufficiently large. By careful flying, the pilot was able to make smooth turns at high speed, as shown by figures 17 and 18. Ordinarily, however, small movements of the stick caused appreciable variations in the normal acceleration, as shown in figures 15 and 20.

The variation of stick force with normal acceleration in turns is plotted in figure 21(b). The stick-force gradient of 5.0 pounds per g was considered a little too light by most of the pilots. It is lower than the value of 6 pounds per g recommended as an upper limit in reference 1. Inasmuch as the elevator mass unbalance under static conditions gave a force of 4.0 pounds on the stick, it is apparent that the stick force required in accelerated flight came almost entirely from the statically unbalanced elevator. Practically no stick force would be required in turns if the elevator were mass balanced. This suggests that the airplane would appear definitely unstable in turns if the elevator were mass balanced, as is required for flutter prevention on American pursuit airplanes.

The stick-force gradient measured in pull-ups, shown on figure 12(b), was in good agreement with that obtained in turns. The motor would cut out when negative acceleration was experienced in the push-downs required to recover from these pull-ups.

Characteristics of the elevator control in landing.- The average elevator angle required to make a three-point landing was about 8.4° up with respect to the thrust axis. The elevator angles used at contact in individual landings varied over a range of 10°, partly because tail buffeting caused the elevator to oscillate and partly because the pilot had continually to apply corrections to the angle of pitch of the airplane because of the lack of longitudinal stability in the landing condition. The elevator angle required for three-point contact was always well within the available range.

The average value of the elevator angles used in three-point landings was 12° higher than the elevator angle required to reach the minimum speed in a gradual stall in the landing condition at altitude. The airplane could be flown, however, in a partly stalled condition at altitude with the stick full back. The reduction in down-

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wash at the tail caused by separation of the flow at the wing root was probably combined with that due to ground effect in the three-point landings. A time history of a three-point landing, shown in figure 22, illustrates the unsteadiness of the airplane and controls as the landing attitude was approached.

The stick force required to make a three-point landing was much less than the value of 35 pounds recommended as an upper limit in reference 1. By use of the trim tab, the force could be reduced almost to zero.

Characteristics of the elevator control in take-off.- The elevator power was adequate to raise the tail or adjust the attitude angle as desired during take-off. Figure 23 shows the time history of a take-off made with 48 inches of mercury manifold pressure. The time required to leave the ground in this case (11.1 sec) does not represent the minimum possible take-off time.

Trim change due to power and flaps.- Trim changes caused by the application of power or flaps were unusually small in the Spitfire. This quality is highly desirable in a fighter-type airplane. The following table shows the stick-force changes with a given tab setting required to maintain trim at 120 miles per hour in various conditions of flight.

Pilot's indicated airspeed (mph)	Correct indicated airspeed (mph)	Hood	Flaps	Gear	Engine speed (rpm)	Manifold pressure (in. Hg)	Shutters	Stick force (lb)
120	125	closed	up	up	-----	throttle closed	flush	0
120	128	closed	up	up	2850	44	flush	3 push
120	124	closed	down	up	2850	44	flush	0
120	127	closed	up	down	2850	44	flush	1.3 push
120	123	closed	up	down	----	throttle closed	flush	1 pull
120	123	closed	down	up	----	throttle closed	flush	2 pull
120	123	open	down	down	----	throttle closed	closed	3 pull

The stick force required to maintain trim while the flight condition was changed in any possible manner was much less than the value of 35 pounds set as an upper limit in reference 1.

Characteristics of longitudinal trimming device.- Because the trim changes required for the different flight conditions were so small, the elevator trim tabs had ample power to trim the airplane at any speed in any flight condition for the center-of-gravity location used in these tests. In order to determine the power of the elevator trim tabs, measurements of the elevator forces required for trim with different trim-tab settings were made at various speeds. The change in stick force per degree trim-tab change is plotted as a function of speed for three flight conditions in figure 24. The variation of stick force with speed for any trim-tab setting may be obtained by adding to the forces plotted on figures 9, 10, and 11 the force caused by the change in trim-tab angle.

Lateral Stability and Control

Characteristics of uncontrolled lateral and directional motion.- The characteristics of the control-free lateral oscillation were determined by trimming the airplane for steady flight and then deflecting the rudder and releasing the controls. Records were taken of the subsequent variation of sideslip angle. These measurements were made in the cruising condition at 125 and 200 miles per hour. The damping of the oscillation satisfactorily met the requirement of reference 1. At 200 miles per hour, one oscillation, and at 125 miles per hour, 1.5 oscillations were required for the motion to damp to one-half amplitude. No undamped short-period oscillations of the controls themselves were observed, except for a tendency toward an aileron shake near full aileron deflection. This type of oscillation is not a control-free characteristic and therefore will be discussed under the heading of aileron-control characteristics.

Aileron-control characteristics.- The effectiveness of the ailerons of the Supermarine Spitfire airplane was determined by recording the rolling velocity produced by abruptly deflecting the ailerons at various speeds. The aileron angles and stick forces were measured. It should be noted that the airplane tested was equipped with metal-covered ailerons.

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The results of these tests are presented in figures 25 to 28. Figure 25 shows the variation of $pb/2V$ and aileron force with total aileron deflection in the landing condition, and figure 26 gives these curves for level flight with flaps and gear up at three speeds. Total aileron deflection is defined as the sum of the deflections of the right and left ailerons. The quantity $pb/2V$ is the helix angle in radians described by the wing tip in a roll, where p is the rolling velocity in radians per second, b the wing span in feet, and V the true velocity in feet per second. A complete discussion of this criterion for aileron effectiveness is given in reference 3.

The ailerons were sufficiently effective at low speeds, and were relatively light at small deflections in high-speed flight. The forces required to obtain high rolling velocities in high-speed flight were considered excessive. With a stick force of 30 pounds, full deflection of the ailerons could be obtained only at speeds lower than 110 miles per hour. A value of $pb/2V$ of 0.09 radian in left rolls and 0.08 radian in right rolls was obtained with full deflection. A rolling velocity (at 6000 ft altitude) of about 59° per second could be obtained with 30 pounds stick force at 230 miles per hour indicated speed.

The ailerons were relatively light for small deflections, but the slope of the curve of stick force against deflection increased progressively with deflection, so that about five times as much force was required to fully deflect the ailerons as was needed to reach one-half of the maximum travel. The effectiveness of the ailerons increased almost linearly with deflection all the way to the maximum position. The value of $pb/2V$ obtained for a given aileron deflection was nearly the same in all the speeds and conditions tested. It may be concluded, therefore, that there was very little reduction in aileron effectiveness either by separation of flow near minimum speed or by wing twist at high speeds.

Figure 27 shows the aileron deflection, stick force, and helix angle obtained in a series of rolls at various speeds intended to represent the maximum rolling velocity that could be readily obtained. The pilot was able to exert a maximum of about 40 pounds on the stick. With this force, full deflection could be attained only up to about 130 miles per hour. Beyond this speed, the rapid increase in stick force near maximum deflection prevented full motion of the control stick. Only one-half of the available

deflection was reached with a 40-pound stick force at 300 miles per hour, with the result that the $pb/2V$ obtainable at this speed was reduced to 0.04 radian, or one-half that reached at low speeds.

Another method of presenting the results of the aileron-roll measurements is that given in figure 28, where the force for different rolling velocities is plotted as a function of speed. The relatively light forces required to reach small rolling velocities are readily seen from this figure. The excessive forces required to reach high rolling velocities and the impossibility of obtaining maximum aileron deflection much above 140 miles per hour are also illustrated.

The ailerons failed to meet the requirement of reference 1, which states that a value of $pb/2V$ of 0.07 radian should be reached with a stick force of 30 pounds at 0.8 of the maximum level-flight indicated speed, or about 230 miles per hour in this case. Under these conditions, a value of $pb/2V$ of only 0.051 radian was attained.

The pilots observed an aileron shake near full deflection. This shaking of the control system is attributed to separation of the flow from the projecting Frise balance on the lower surface of the upward-deflected aileron. The shaking was not particularly violent or objectionable on the Spitfire. This phenomenon has caused trouble, however, on airplanes with more flexible control systems.

Yaw due to ailerons.- In aileron rolls made at 110 percent of the minimum speed with full aileron deflection and with the rudder fixed, about 18° sideslip was developed. The requirement of reference 1, which states that less than 20° sideslip shall be developed in this maneuver, was therefore met.

Rolling moment due to sideslip.- The rolling moment due to sideslip of the Spitfire airplane was determined by recording the aileron angles required in steady sideslips. The results of these measurements are presented in figures 29 to 34, where the rudder, elevator and aileron angles, angle of bank, and rudder force are plotted as functions of the sideslip angle. The dihedral effect was stable in all conditions, with the exception that in left sideslips in the cruising condition the dihedral effect was practically neutral. The requirement of reference 1 was therefore met in all conditions except in left sideslips with power on.

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A further indication of the rolling moment due to sideslip is given by the rolling velocities caused by abrupt deflections of the rudder (figs. 35 and 36). The airplane always rolled in the correct direction. The rolling moment due to yawing velocity is combined with that due to sideslip in these tests.

The stick force in sideslips was not recorded, but it was observed that in conditions where the dihedral effect was stable the stick tended to return toward neutral when released.

Rudder control characteristics.- The rudder control characteristics were investigated in steady flight, in sideslips, and in abrupt rudder kicks. In the rudder kicks records were taken of the rudder force, rolling velocity, sideslip angle, and normal acceleration resulting from abrupt deflections of the rudder. The results of these tests are presented on figures 35 and 36.

A sideslip angle of about 25° resulted from abrupt maximum deflection of the rudder in the flap-up condition at low speeds. Since this sideslip angle exceeds the sideslip caused by full aileron deflection with the rudder fixed, the rudder control is believed to be sufficiently powerful to overcome the adverse aileron yawing moment.

The initial values of rudder force in rudder kicks, plotted in figures 35 and 36, show that the rudder was desirably light. The floating tendency of the rudder caused the pedal force to drop to about one-third of its initial value after the sideslip had built up. No reversal of rudder force ever occurred, however. The requirement of reference 1, which states that left rudder force should always be required for left rudder deflections and right rudder force for right rudder deflections, was therefore satisfied.

Considerable deflection of the rudder to the right was required in the power-on conditions in steady flight near minimum speed as shown on figures 9, 10, and 11. The limits of rudder travel were never exceeded, however. The rudder deflection is needed partly to offset the yawing moment caused by angularity of the flow due to the slipstream, and partly to balance the left yawing moment of the propeller itself that results from the high angle of attack of the propeller axis. The side force on the rudder necessary to maintain equilibrium of yawing moments

about the center of gravity, together with the side force on the tilted propeller, are believed to be responsible for the appreciable amount of left sideslip which was observed to occur in low-speed power-on flight with the wings laterally level.

The rudder forces required for trim throughout the speed range with the rudder trim tab neutral are plotted for the gliding and cruising conditions on figure 9. These rudder forces were unusually light.

The rudder control, in conjunction with the brakes, was sufficiently powerful to maintain directional control in take-off and landing. A time history of a take-off (fig. 23) shows that some rudder deflection was required to overcome a tendency to turn to the left. The rudder force required for this purpose, however, was observed to be relatively light.

The rudder forces required to overcome adverse aileron yaw and to maintain directional control in take-off and landing probably never exceeded half the value of 180 pounds specified as an upper limit in reference 1. No investigation of the effectiveness of the rudder in recovering from spins was attempted.

Yawing moment due to sideslip.- The yawing moment due to sideslip is indicated by the rudder deflections required in steady sideslips (figs. 29 to 34). The directional stability was satisfactory in that the rudder always moved in the correct direction in sideslips. As previously stated, the directional stability was sufficient to restrict the yaw due to ailerons to the limits specified in reference 1.

The yawing moment due to sideslip with rudder free is shown by the variation of rudder force with sideslip angle in steady sideslips. The slope of the curve of rudder force against angle of sideslip was always stable, though it was very small for small angles of sideslip. The increase of rudder force required at large angles of sideslip insured that the airplane would always tend to return to zero sideslip if the rudder were free, regardless of the magnitude of the sideslip angles.

Cross-wind-force characteristics.- The cross-wind-force characteristics of the airplane are shown by the

angles of bank required to hold steady sideslips in the various flight conditions (figs. 29 to 34). The angles of bank were small at low speeds, but they increased rapidly with speed because the side force for a given sideslip angle varies approximately as the square of the speed. The Spitfire showed a slightly smaller side-force gradient than any other pursuit-type airplane tested previously. A larger side-force gradient would seem to be desirable because the pilot would find it easier to maintain unyawed flight if a large angle of bank were required to sideslip.

Pitching moment due to sideslip.- The pitching moment due to sideslip is shown by the variation of elevator angle with angle of sideslip in the steady sideslip measurements (figs. 29 to 33) and by the variation of normal acceleration with rudder angle in the rudder kicks (figs. 35 and 36). The Spitfire showed a tendency to pitch down both in left and right sideslips. In power-on flight, this airplane failed to meet the requirement (reference 1) that less than 1° change in elevator angle should accompany 5° deflection of rudder. As shown on figure 29, left sideslip occurred in the trim condition with wings level, and at this sideslip angle the elevator angle increased approximately linearly with the left sideslip angle. The static longitudinal-stability measurements indicated that the sideslip increased as the speed was reduced until it reached about 10° at the stall. The elevator angle required for this sideslip completely overshadowed any elevator motion required to change speed in unyawed flight. The increased up-elevator angles encountered at low speeds in the cruising condition therefore do not represent static longitudinal stability. It is doubtful that this type of variation of elevator angle with speed is helpful to the pilot in maintaining a fixed trim speed. Furthermore, the static longitudinal-stability characteristics recorded by two pilots might disagree considerably, because slight errors in holding the wings level would result in appreciable differences in sideslip angle.

The violence of the pitching motions of the airplane in rudder kicks is shown by the variation of normal acceleration with rudder angle (figs. 35 and 36). Because the large accelerations made it difficult for the pilot to hold the elevator angle constant, the normal acceleration plotted on these figures may be partly the result of elevator motion. Nevertheless, the plotted values give a qualitative idea of the pitching motions caused by rudder

deflection. In all cases except at very high speed, the airplane initially pitched up, then pitched down when the rudder was deflected in either direction. The reason for the initial upward acceleration is not known. The downward acceleration is caused by the pitching moment due to sideslip, which exists after the sideslip angle has built up. The motor cut out when negative accelerations were encountered. Because of the violence of the negative acceleration, the pilot was usually unable to keep the rudder fully deflected until the maximum sideslip angle was reached.

High-powered airplanes ordinarily show an initial tendency to pitch up in rudder kicks to the left and down in rudder kicks to the right. This motion is attributed to gyroscopic moments from the propeller. The Spitfire showed this tendency for rudder kicks of small deflections at high speed, but in all other cases the airplane initially pitched up in rudder kicks both to the left and right.

Power of rudder and aileron trimming devices.- The trim tab provided on the rudder was sufficiently powerful to reduce the rudder force to zero in any flight condition. The rudder forces required for trim with the trim tab neutral are plotted on figure 9. No trim tab was provided on the ailerons, but the aileron forces for trim were light. The aileron angles required for trim throughout the speed range in the various flight conditions are plotted in figures 9, 10, and 11.

CONCLUSIONS

The flying qualities of the Supermarine Spitfire airplane observed in these tests may be summarized in terms of the accepted standards for satisfactory flying qualities as follows:

1. The short-period longitudinal oscillation was satisfactorily heavily damped in all conditions tested.

2. In all flight conditions the stick-fixed longitudinal stability was either neutral or unstable, and therefore failed to meet the accepted requirements. The requirement for a stable stick-force gradient was met in all conditions of flight except for the condition with flaps down, power on.

3. The stick-force gradient in maneuvers was 5.0 pounds per g. The requirement for a force gradient of less than 6 pounds per g was therefore satisfied.

4. The stick motion required to stall in maneuvers was 3/4 inch. This value is much less than the 4-inch stick travel recommended for satisfactory flying qualities.

5. The elevator control was adequate for landing and take-off.

6. The longitudinal trim changes due to changes in engine power, flap position, or landing-gear position were exceptionally small.

7. The power of the elevator trim tabs was adequate.

8. The damping of the control-free lateral oscillation was satisfactory. No undesirable short-period lateral oscillations were noted.

9. The aileron control was adequate at low speeds but unsatisfactory at high speeds because of the excessive stick forces required to obtain high rolling velocities.

10. Aileron yaw was within the limits specified as acceptable.

11. The dihedral effect was stable except in left sideslips with power on, where it was practically neutral.

12. The rudder was sufficiently powerful to offset aileron yaw and to maintain directional control during landing and take-off. The rudder forces required were well below the upper limit of 180 pounds specified.

13. Directional stability was satisfactory.

14. A large pitching moment due to sideslip existed.

15. The stalling characteristics in normal flight or in maneuvers were excellent though the maximum lift coefficients were low. No undesirable ground-looping tendencies were noted.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va.

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2. Vensel, J. R., and Phillips, W. H.: Stalling Characteristics of the Supermarine Spitfire VA Airplane. NACA A.C.R., Sept. 1942.

3. Gilruth, R. R., and Turner, W. N.: Lateral Control Required for Satisfactory Flying Qualities Based on Flight Tests of Numerous Airplanes. Rep. No. 715, NACA, 1941.

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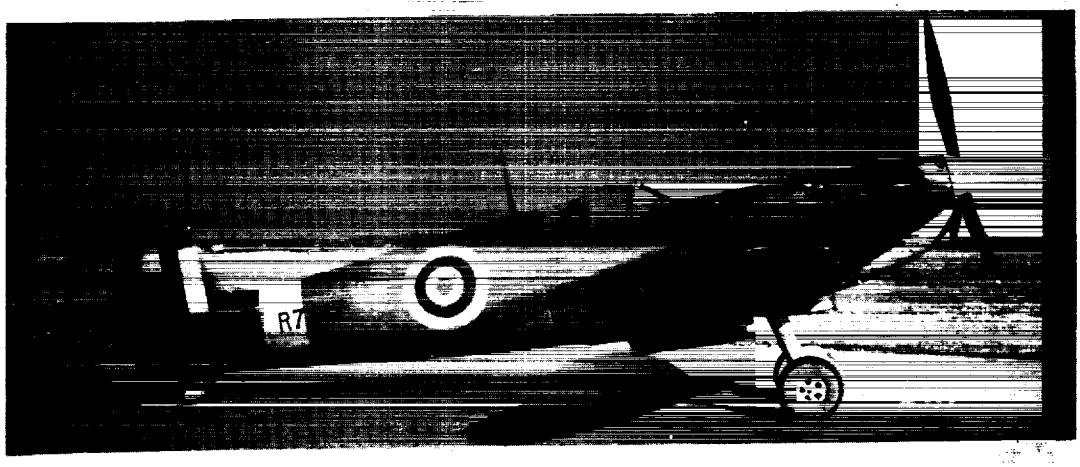

Figure 1.- Side view of the Supermarine Spitfire airplane.

Figure 2.- Front view of the Supermarine Spitfire airplane.

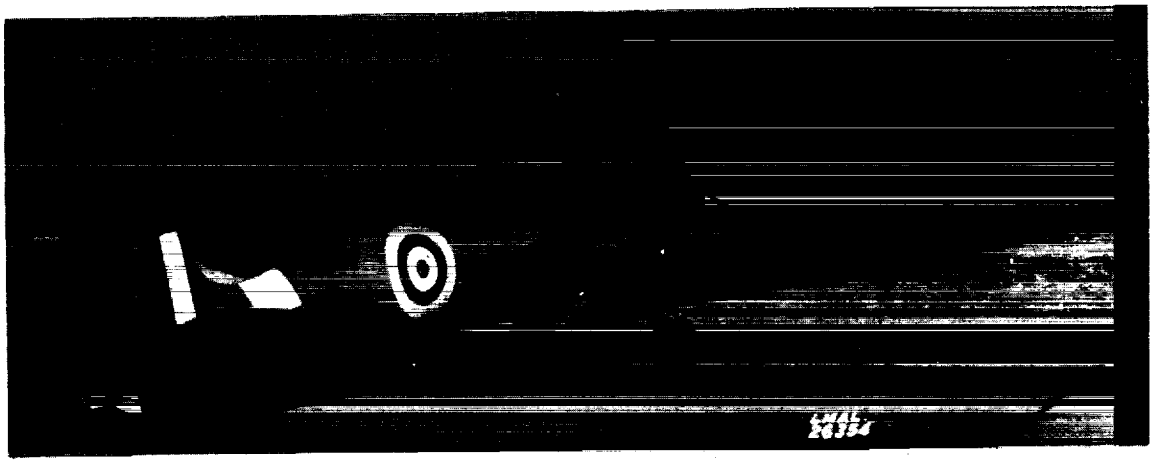

Figure 3.- Three-quarter rear view of the Supermarine airplane.

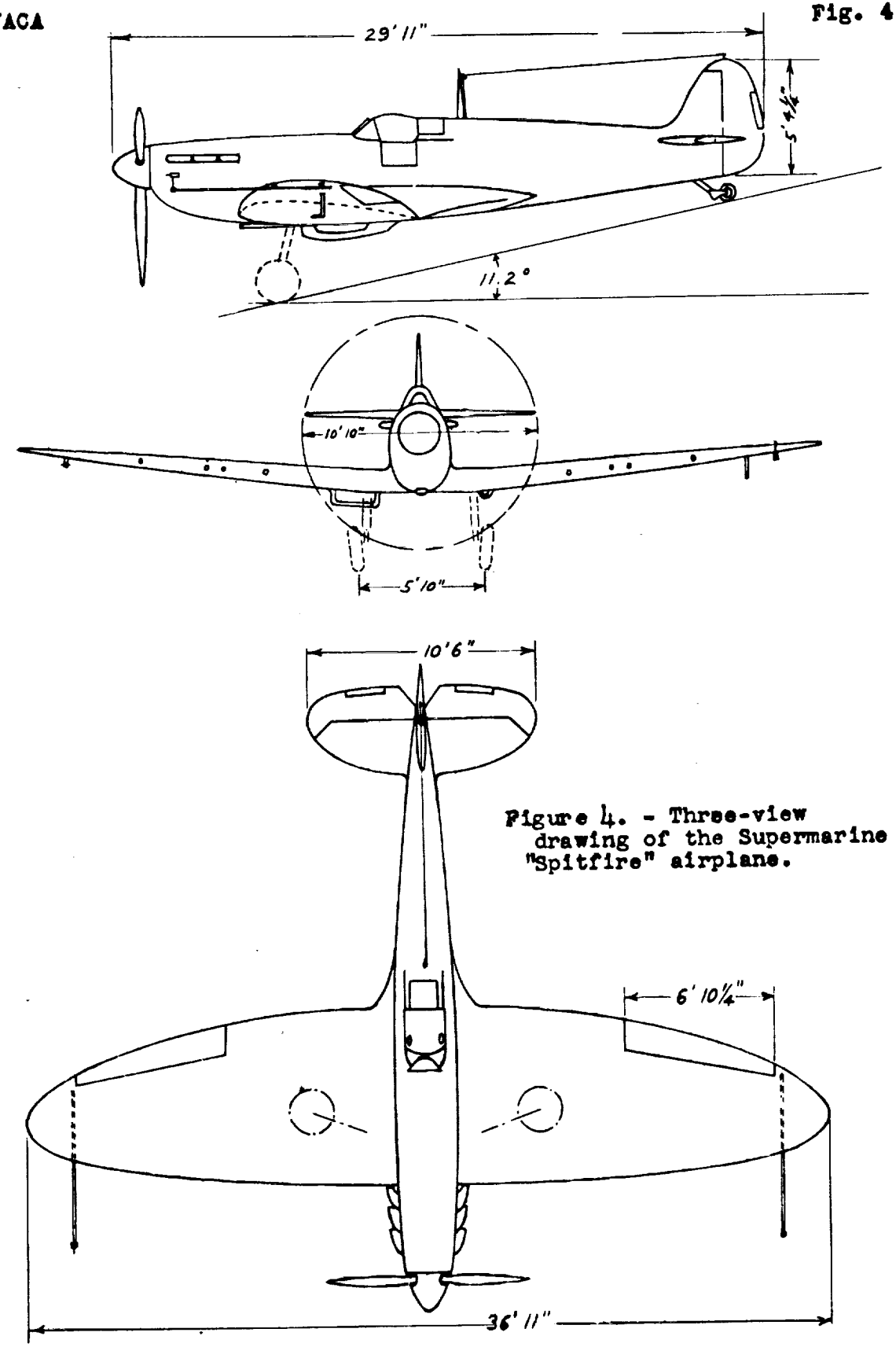

Figure 4. - Three-view drawing of the Supermarine "Spitfire" airplane.

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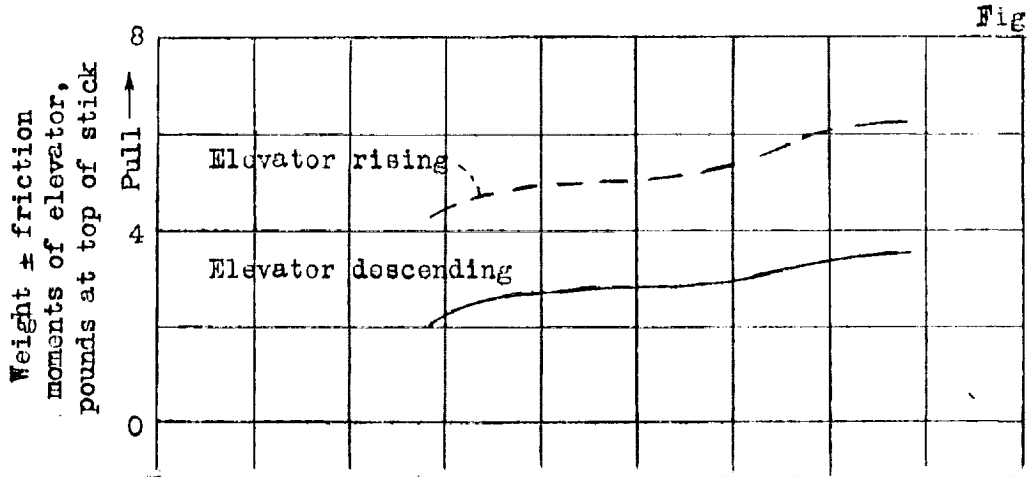
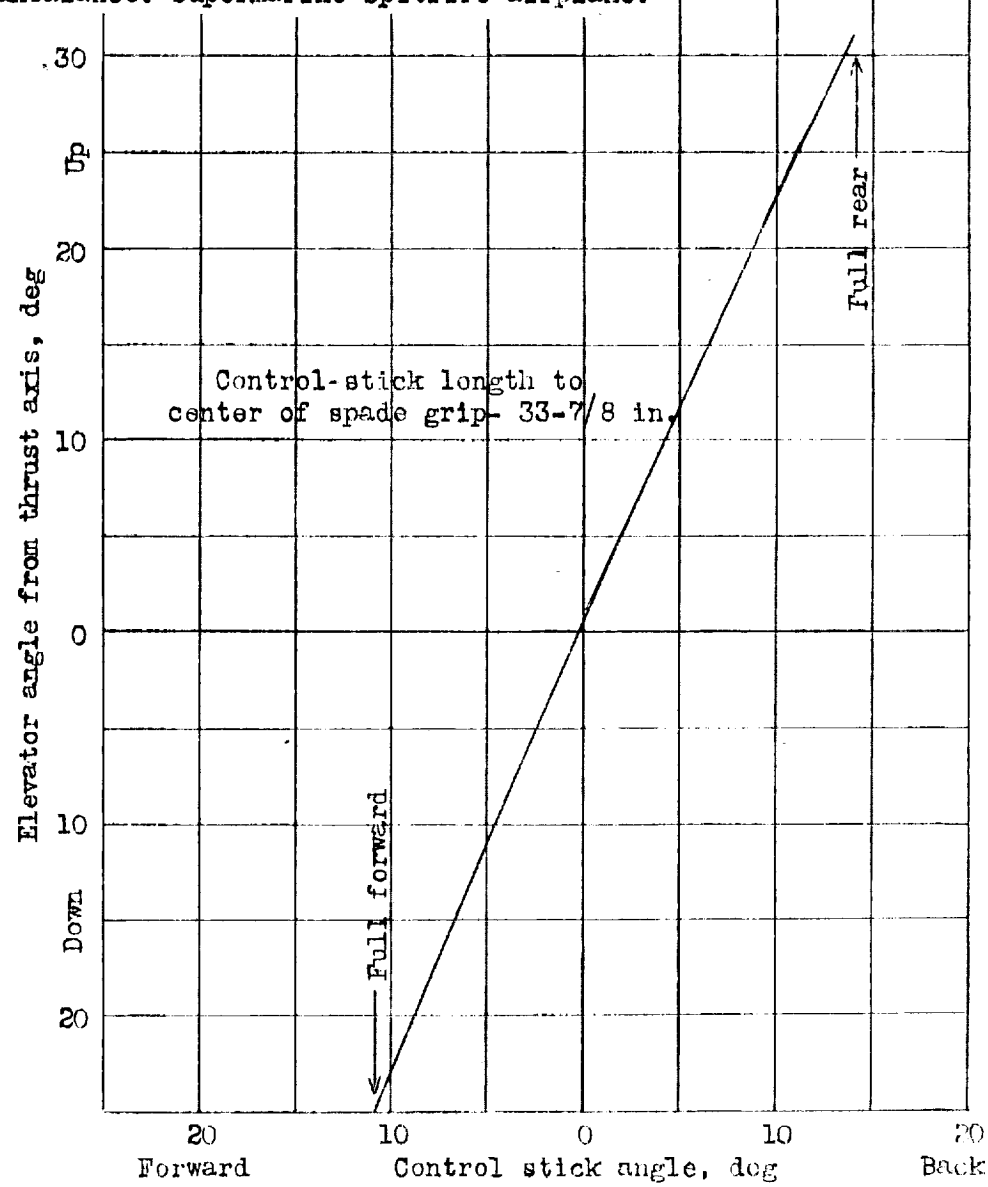


Figure 5.- Variation with control-stick angle of elevator angle and stick force caused by elevator friction and unbalance. Supermarine Spitfire airplane.



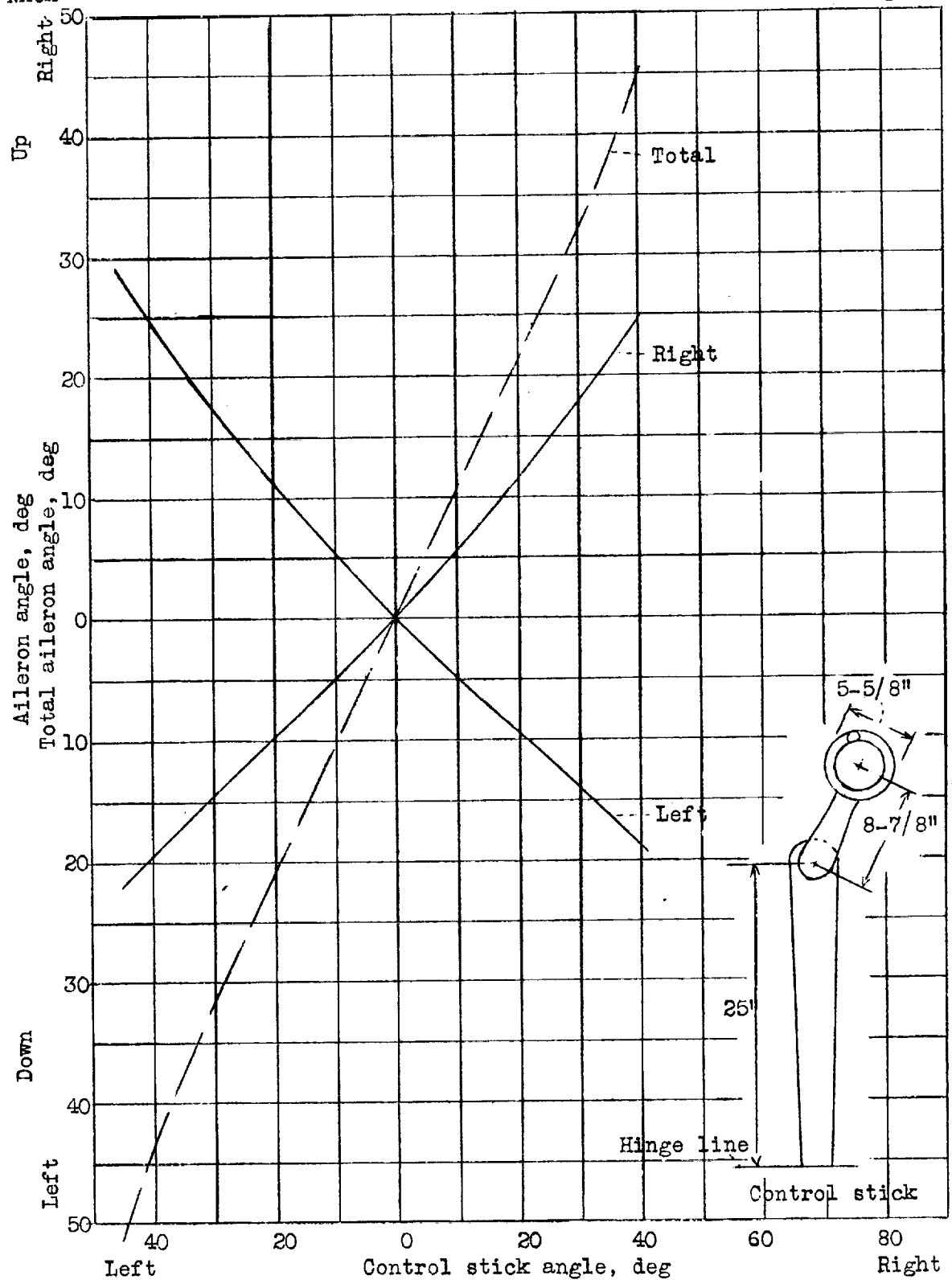


Figure 6.- Variation of left, right, and total-aileron angles with control stick angle. Supermarine Spitfire airplane.

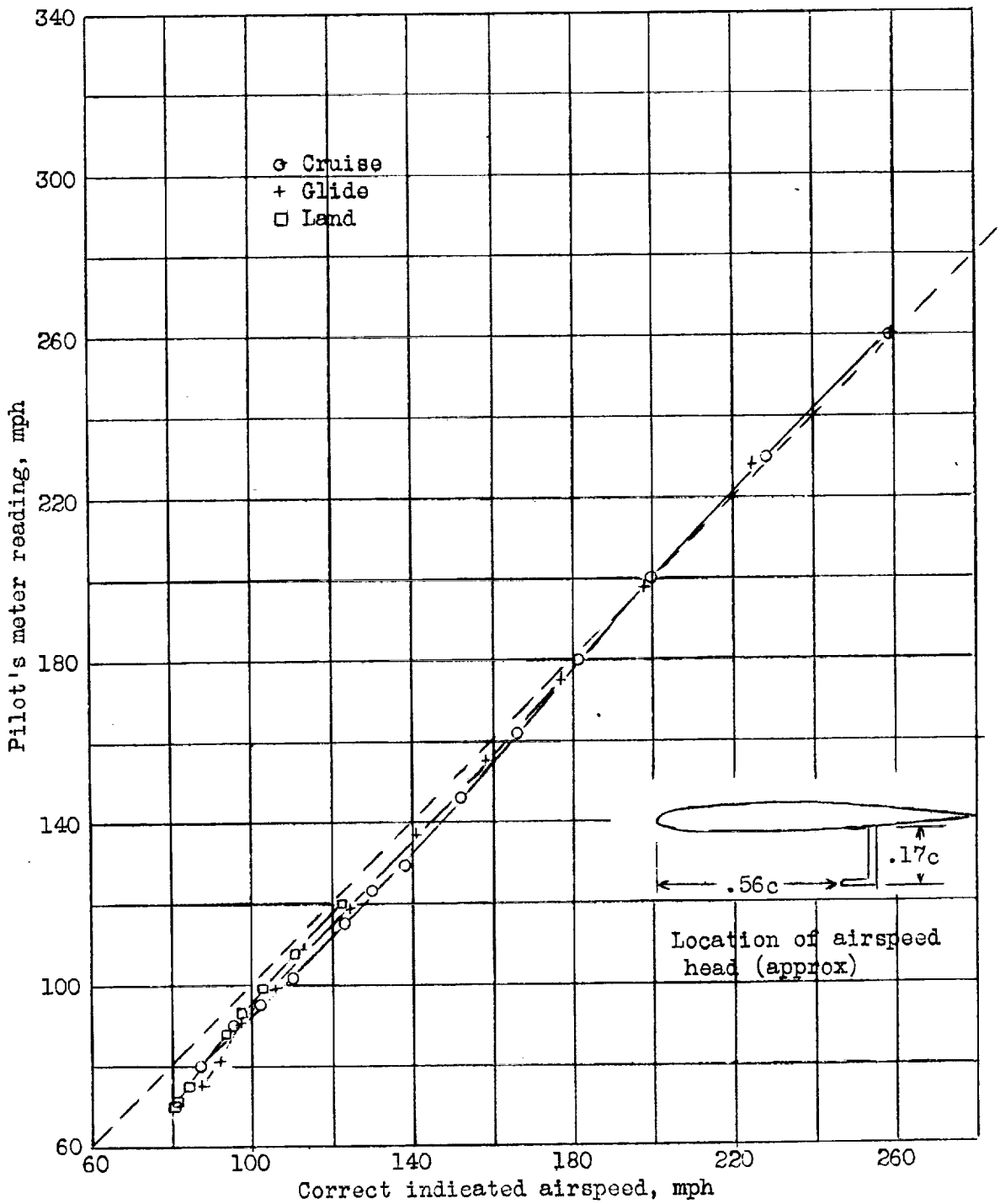


Figure 7.- Calibration of the pilot's airspeed indicator in the Supermarine Spitfire airplane.

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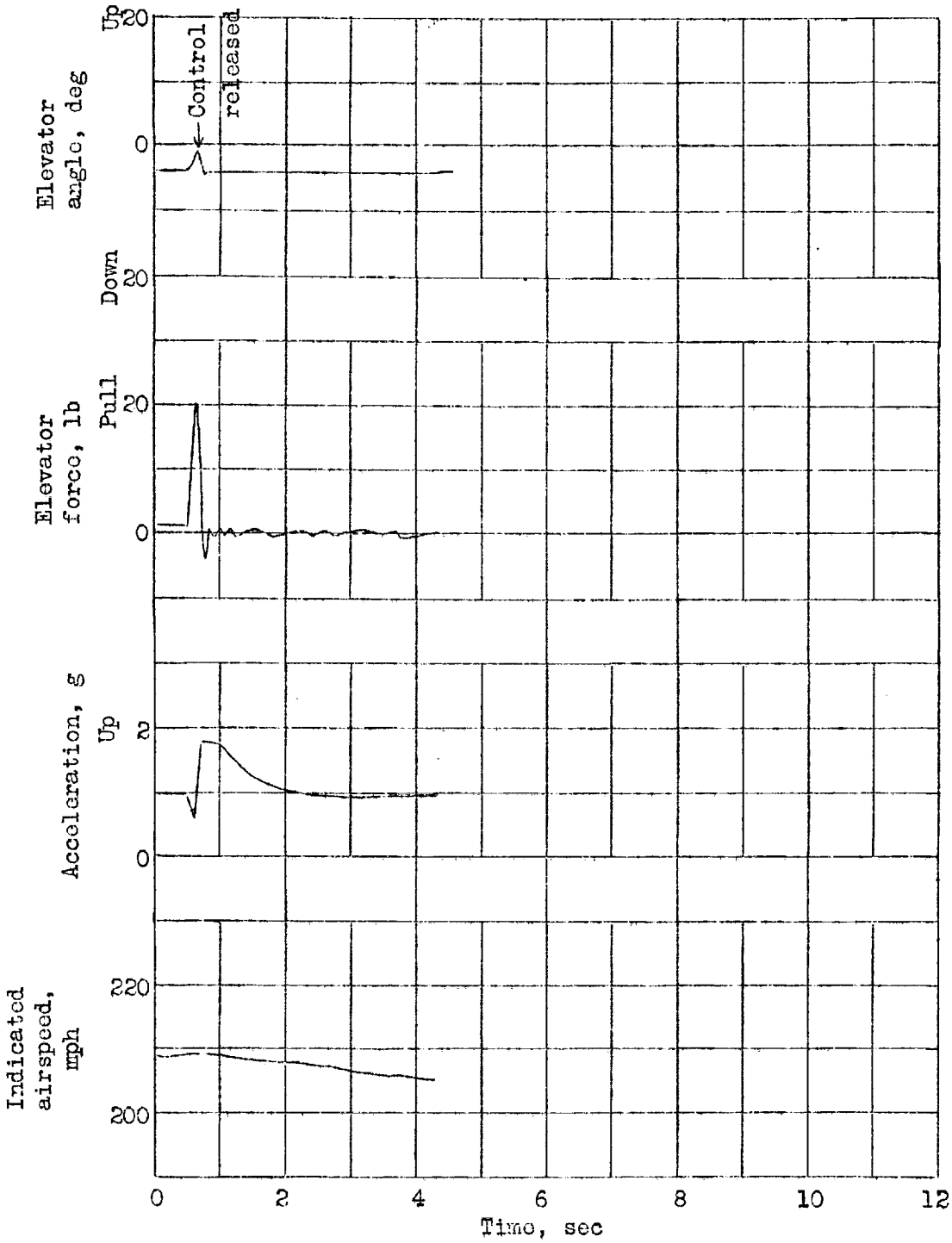

Figure 8.- Time history of a short-period oscillation started by suddenly pulling back the control stick and releasing it in level flight; flaps and gear up. Supermarine Spitfire airplane.

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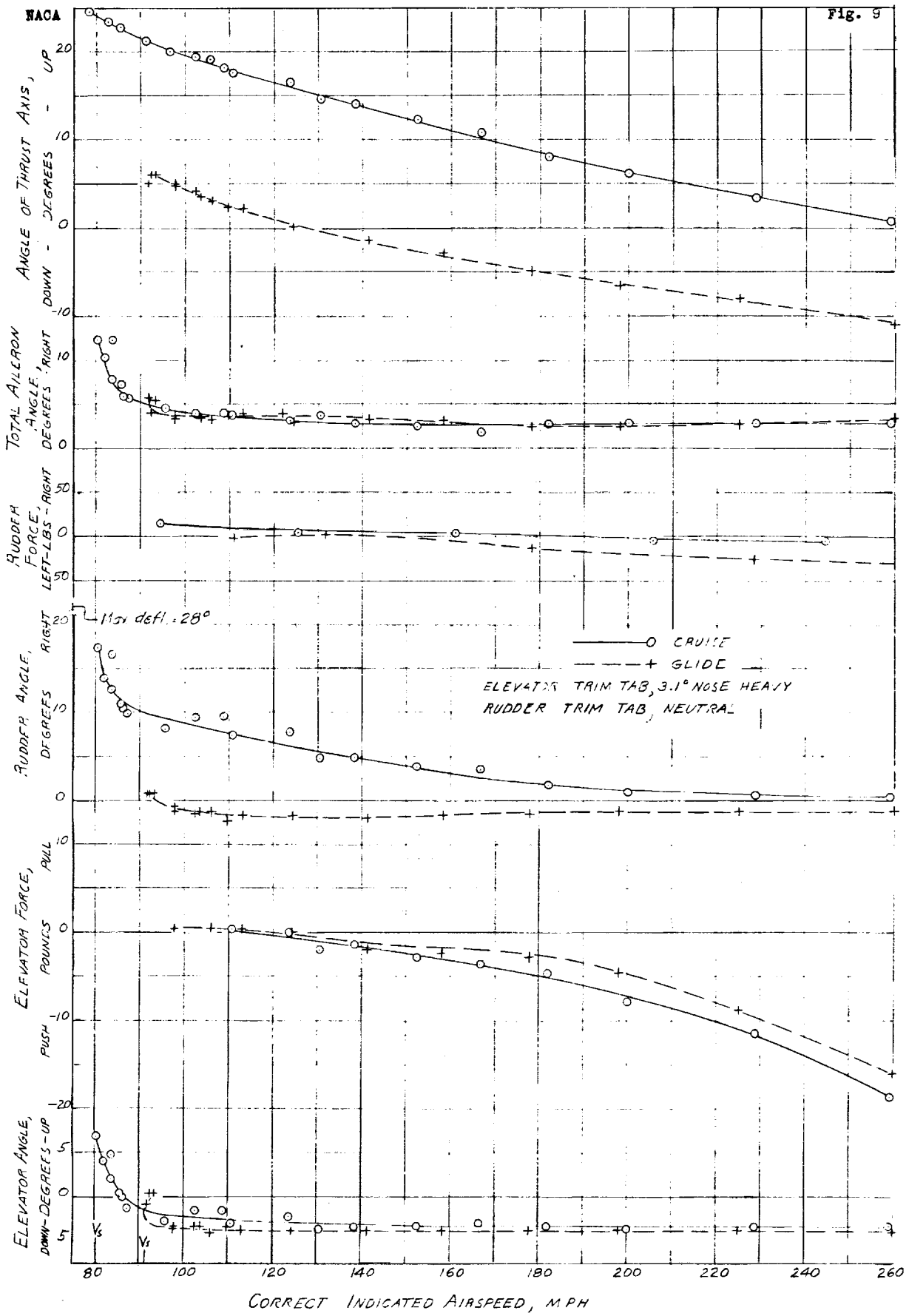

Figure 9. - Static longitudinal-stability characteristics of the Supermarine Spitfire airplane in the cruising and gliding conditions.

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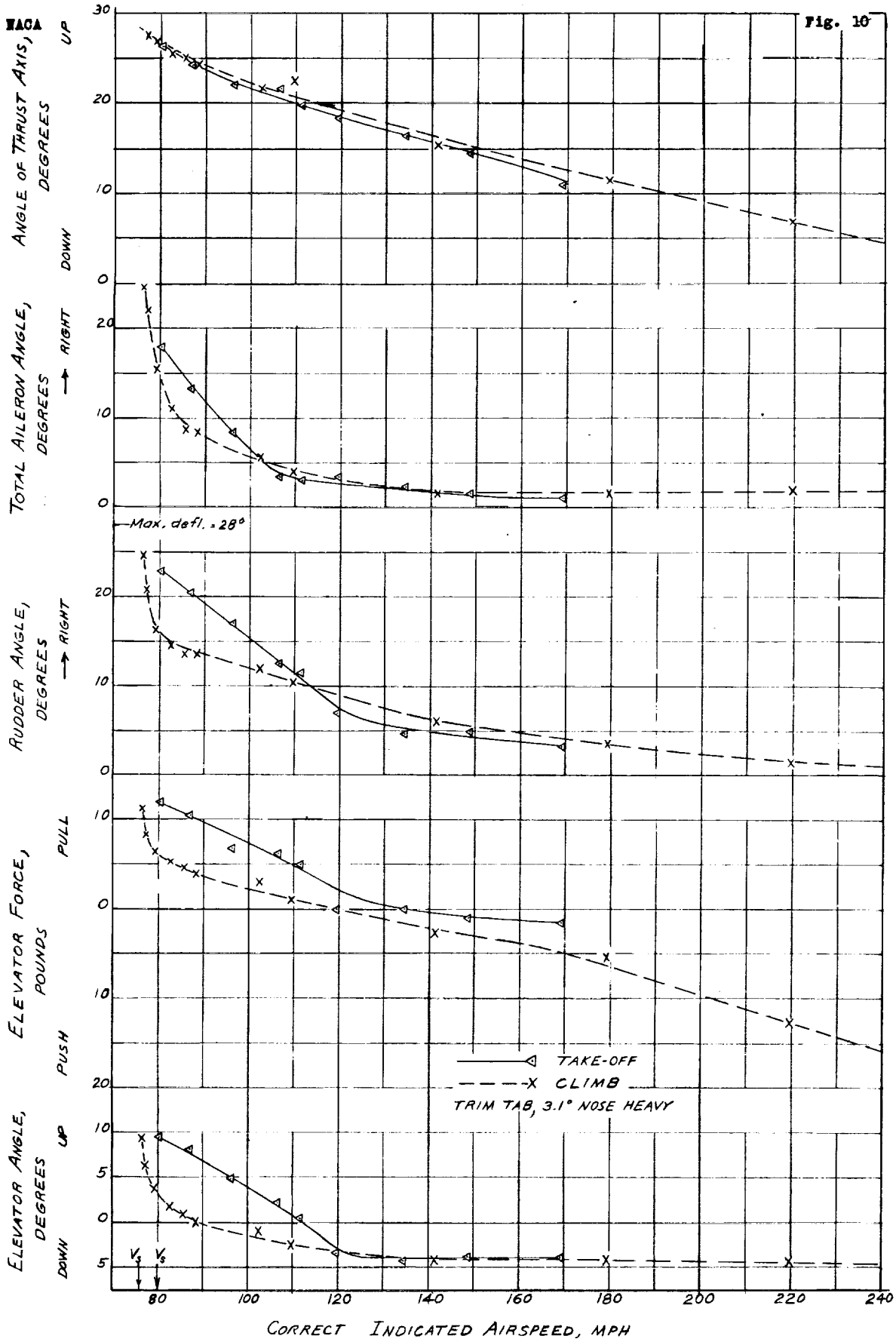

Figure 10. - Static longitudinal-stability characteristics of the Supermarine Spitfire airplane in the climbing and take-off conditions.

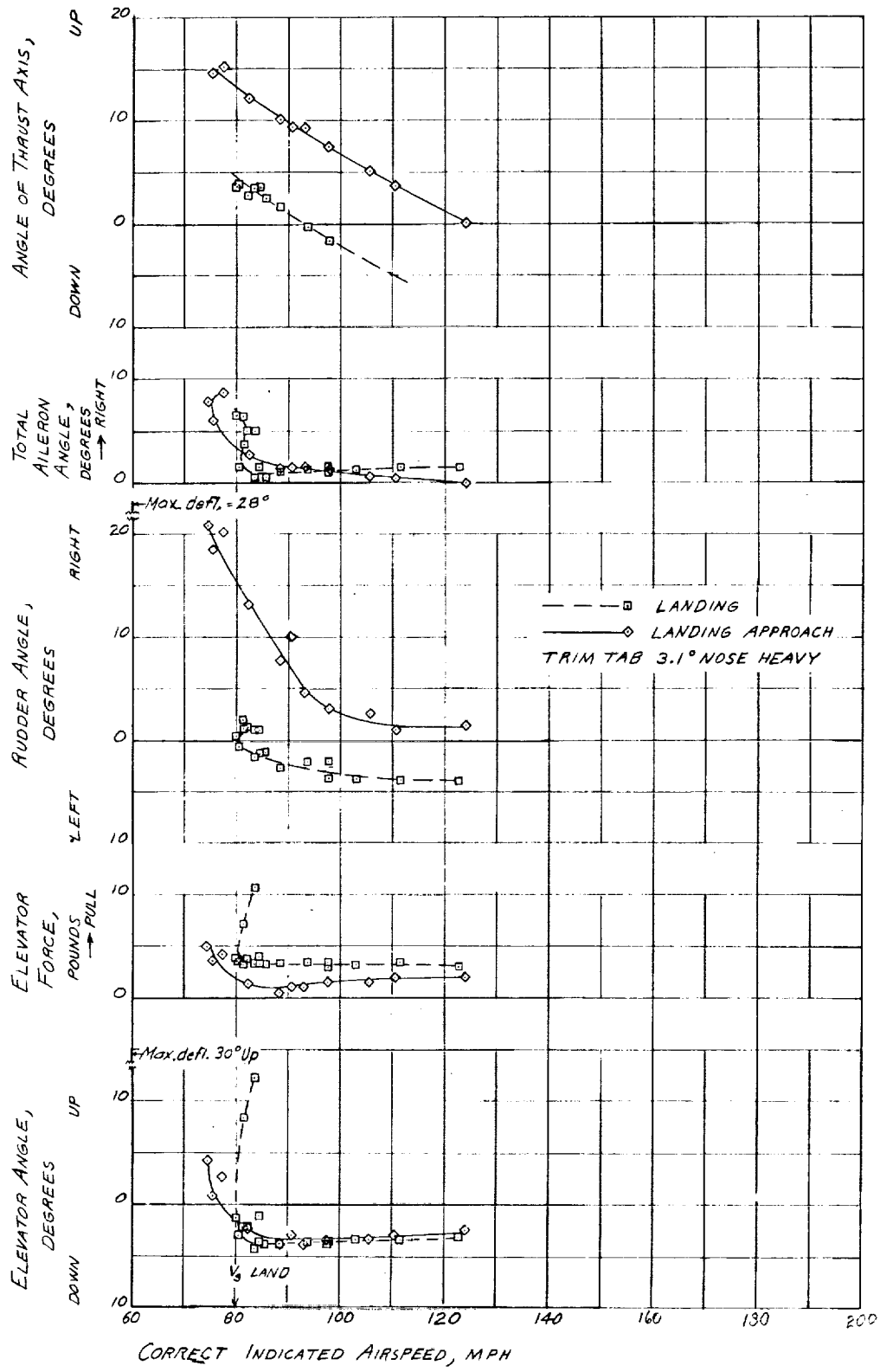

Figure 11. - Static longitudinal-stability characteristics of the Supermarine Spitfire airplane in the landing and landing-approach conditions.

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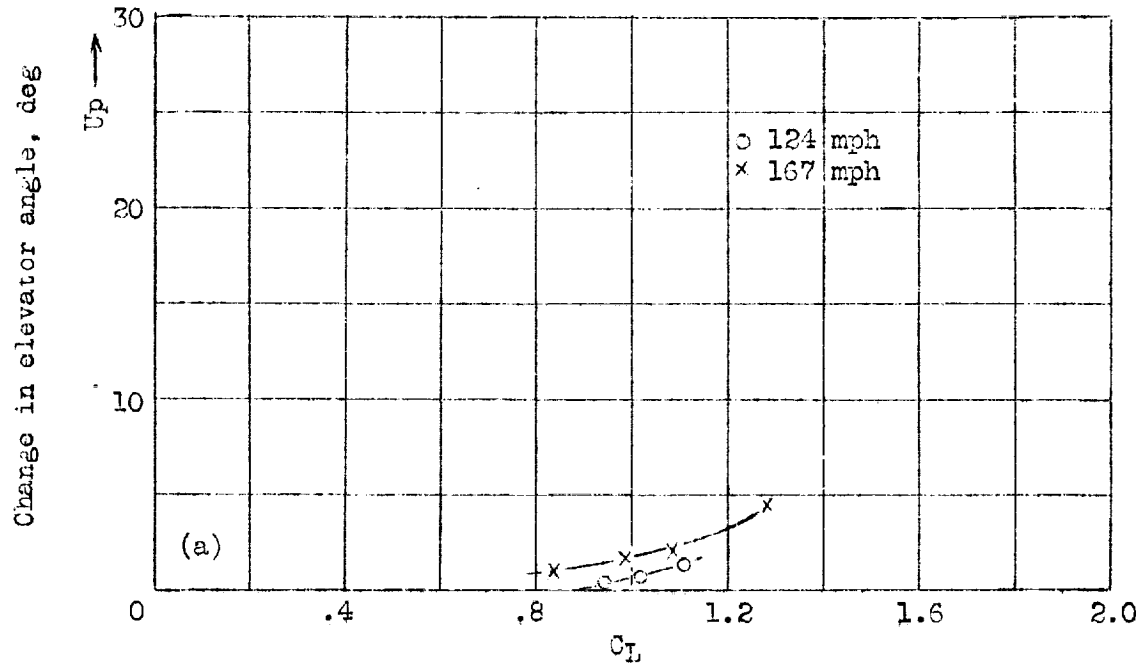
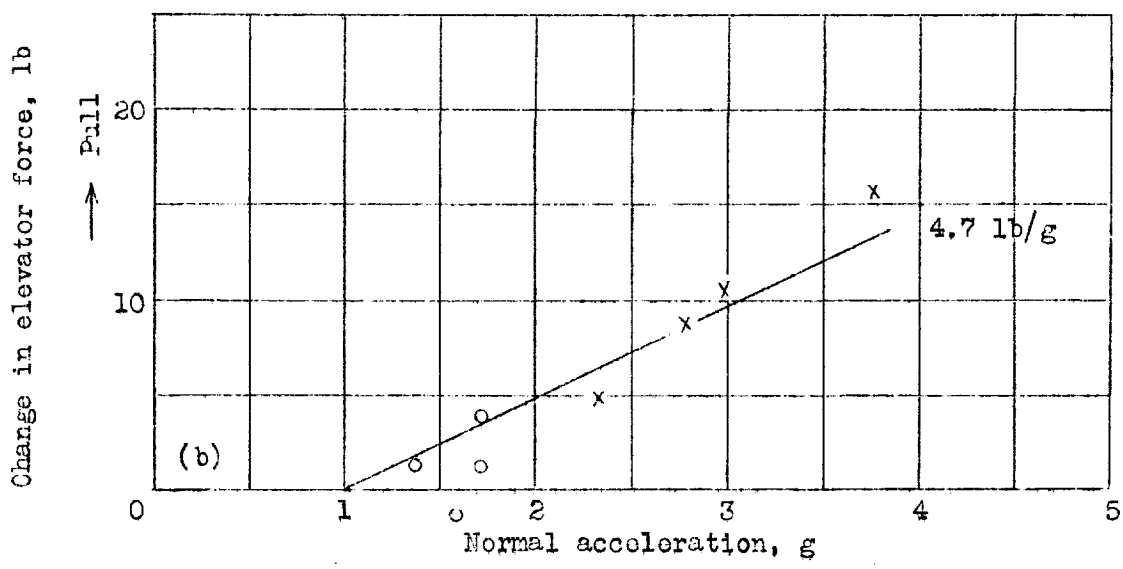


(a) Variation of change in elevator angle with lift coefficient in pull-ups.



(b) Variation of elevator force with normal acceleration in in pull-ups.

Figure 12.- Elevator characteristics in pull-ups. Supermarine Spitfire airplane.

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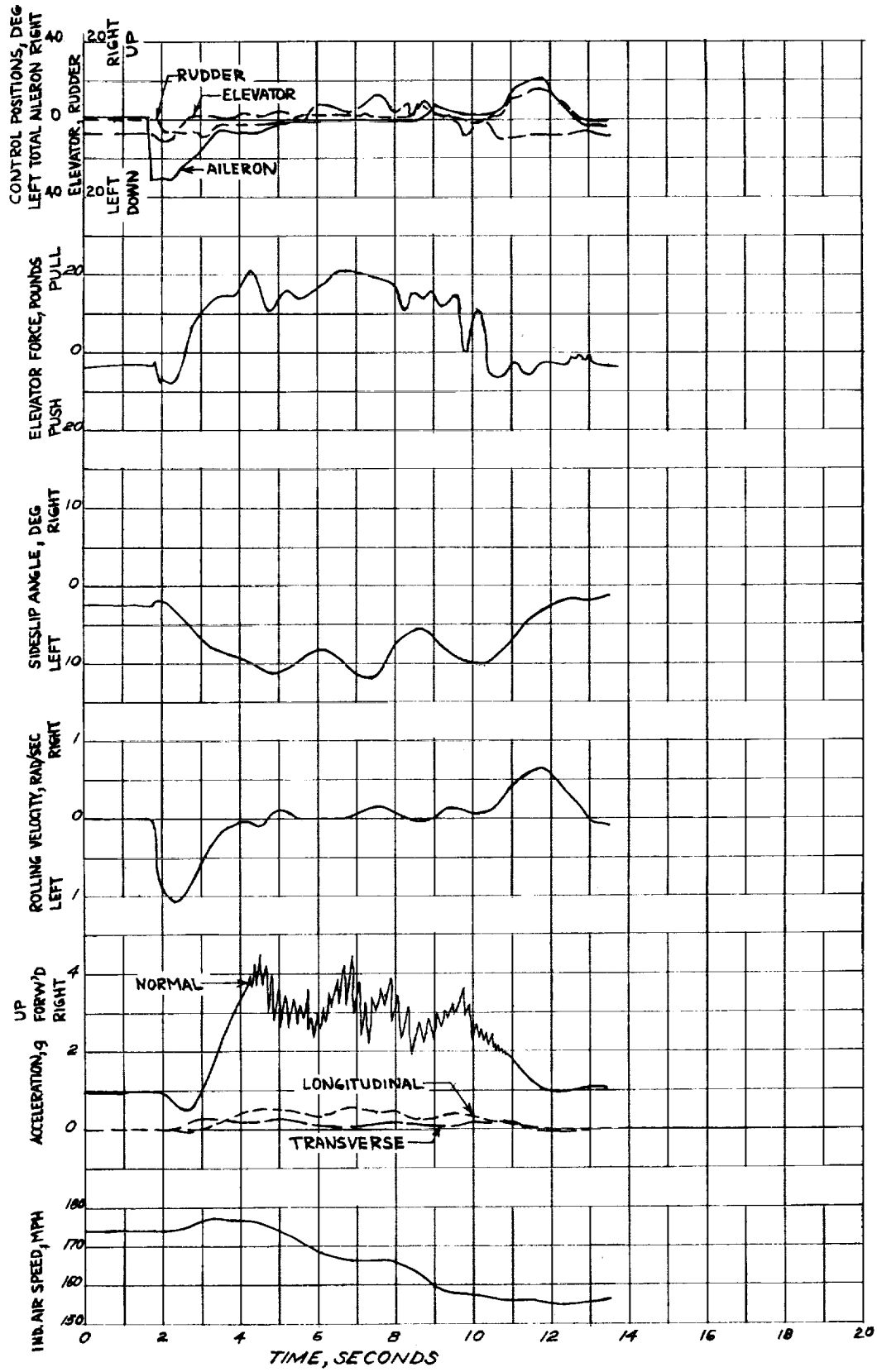

Figure 13. - Time history of a rapid 180° turn to the left started at 174 miles per hour, in which the stall was reached. Supermarine Spitfire airplane; gun parts covered.

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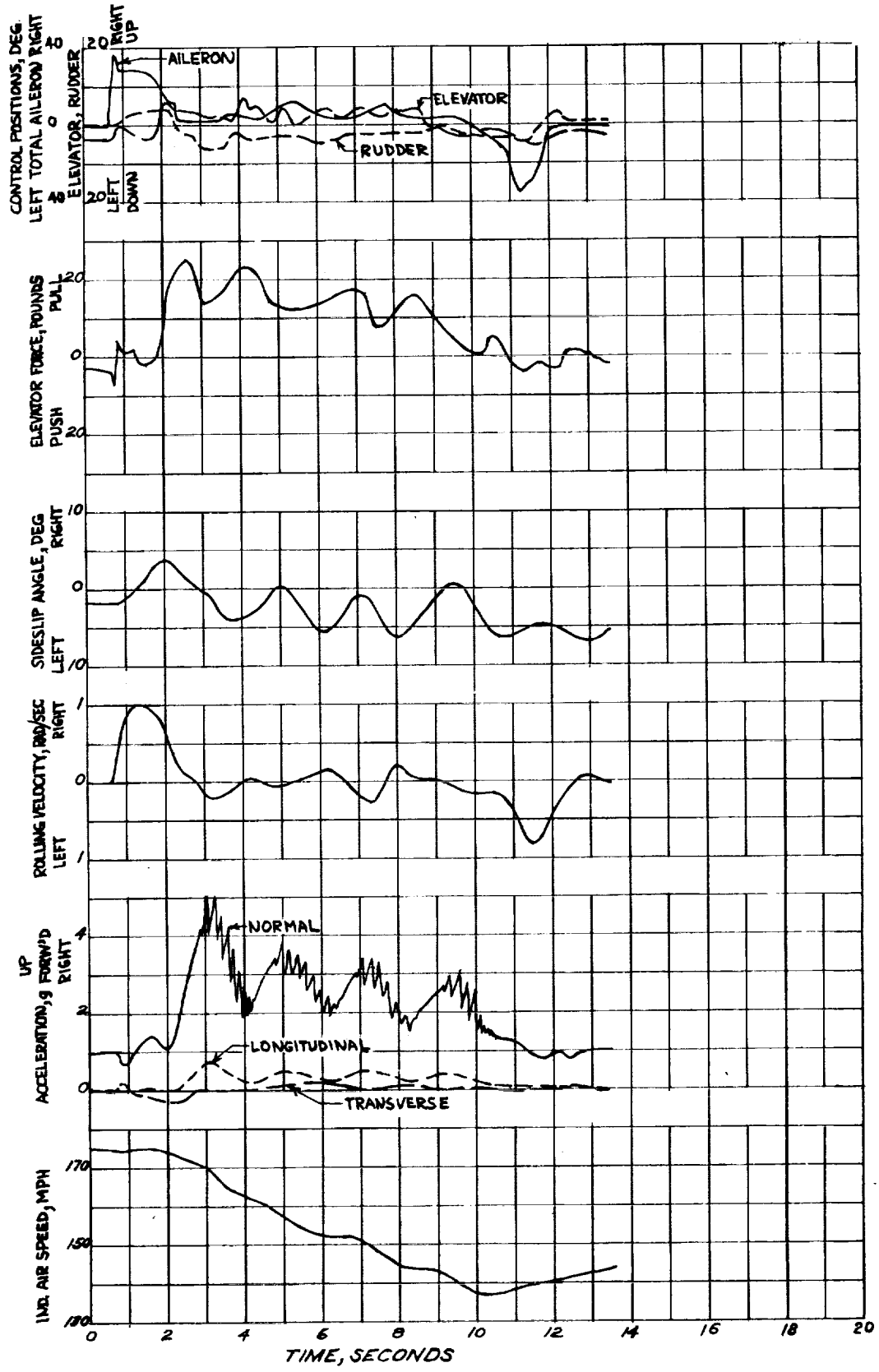

Figure 14. - Time history of a rapid 180° turn to the right started at 175 miles per hour, in which four stalls occurred. Supermarine Spitfire airplane with gun parts covered.

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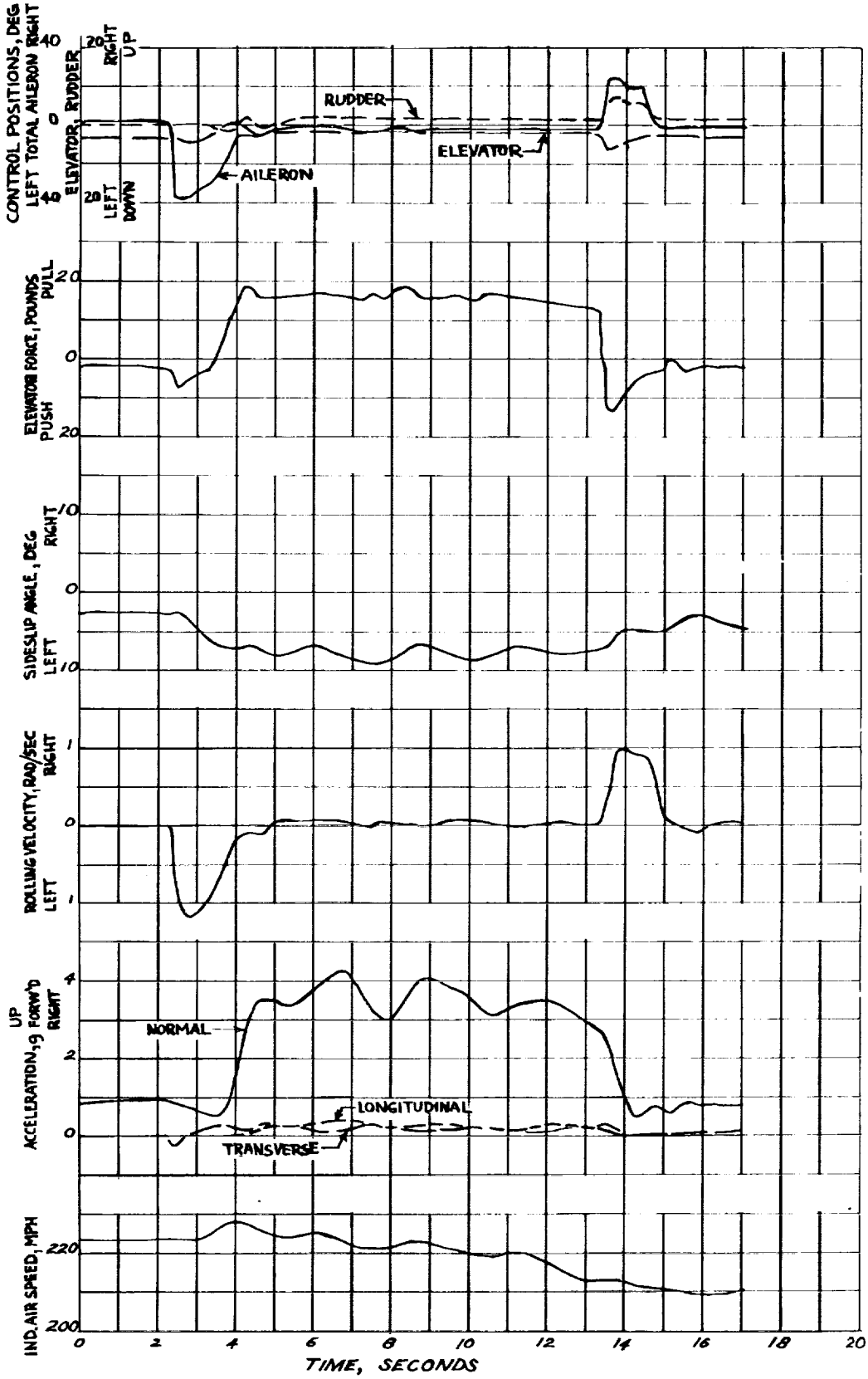

Figure 15. - Time history of a rapid 180° turn to the left started at 223 miles per hour. Supermarine Spitfire airplane with gun parts covered.

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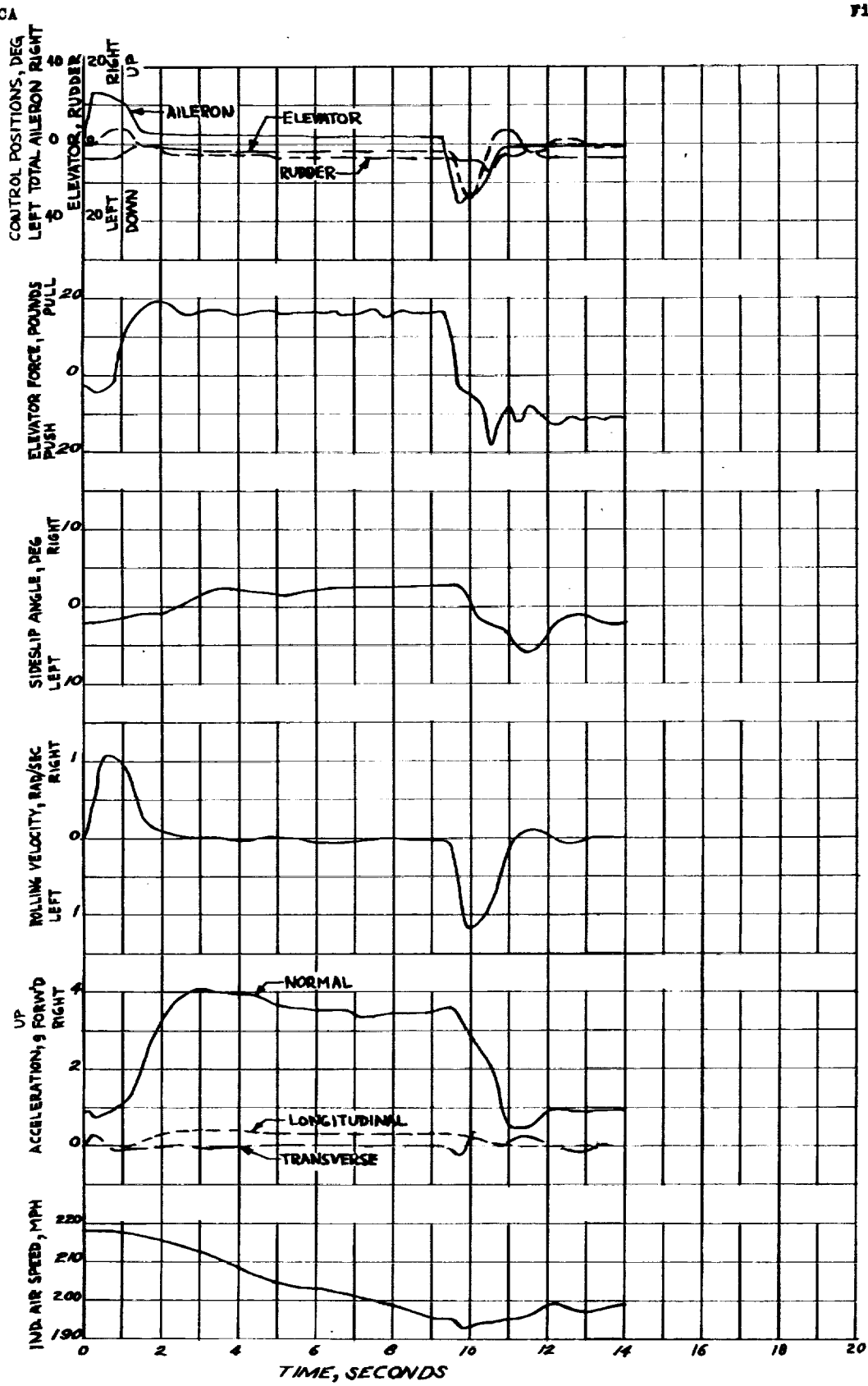

Figure 16. - Time history of a rapid 180° turn to the right started at 218 miles per hour. Supermarine Spitfire airplane with gun parts covered.

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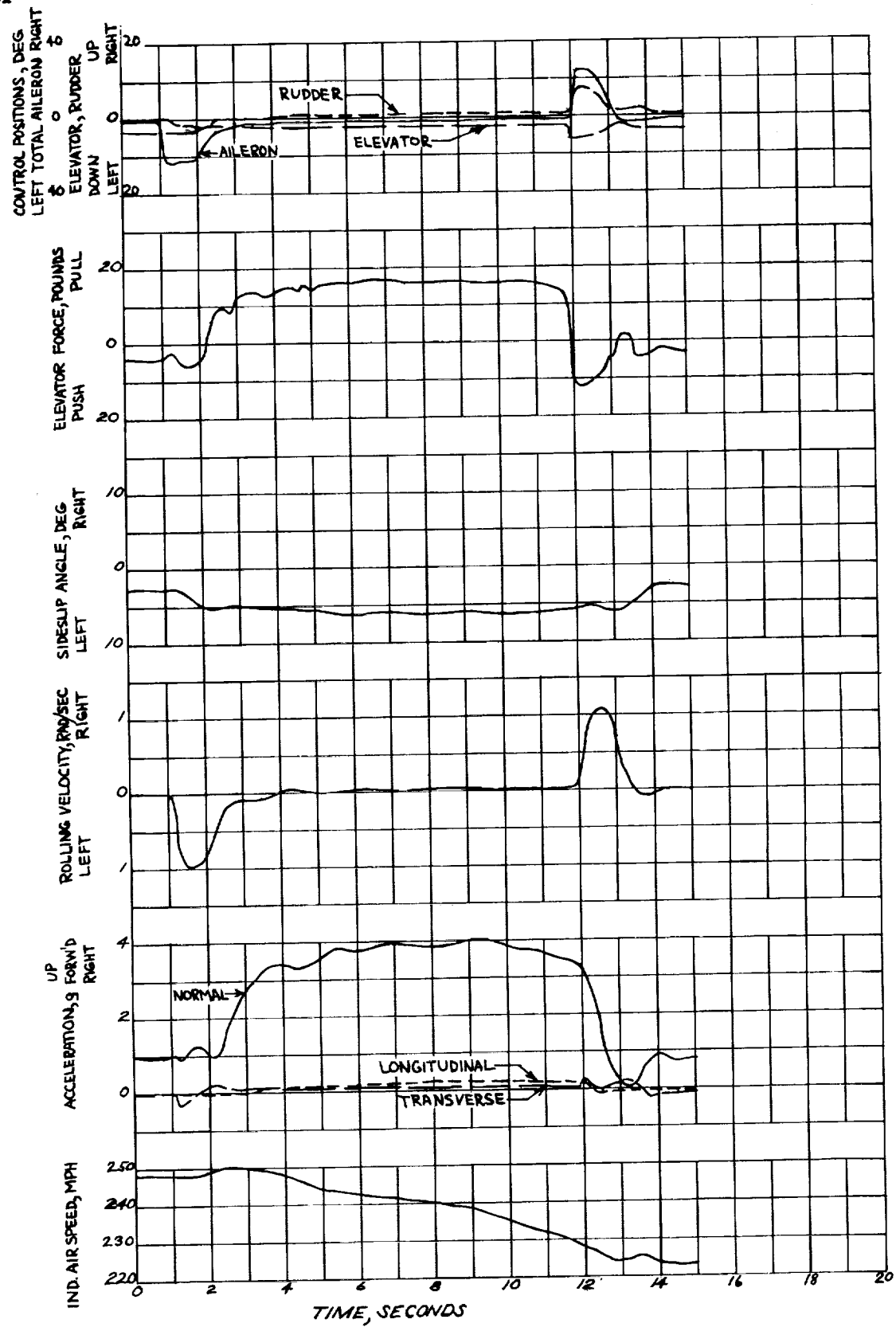

Figure 17. - Time history of a rapid 180° turn to the left started at 248 miles per hour. Supermarine Spitfire airplane with gun ports covered.

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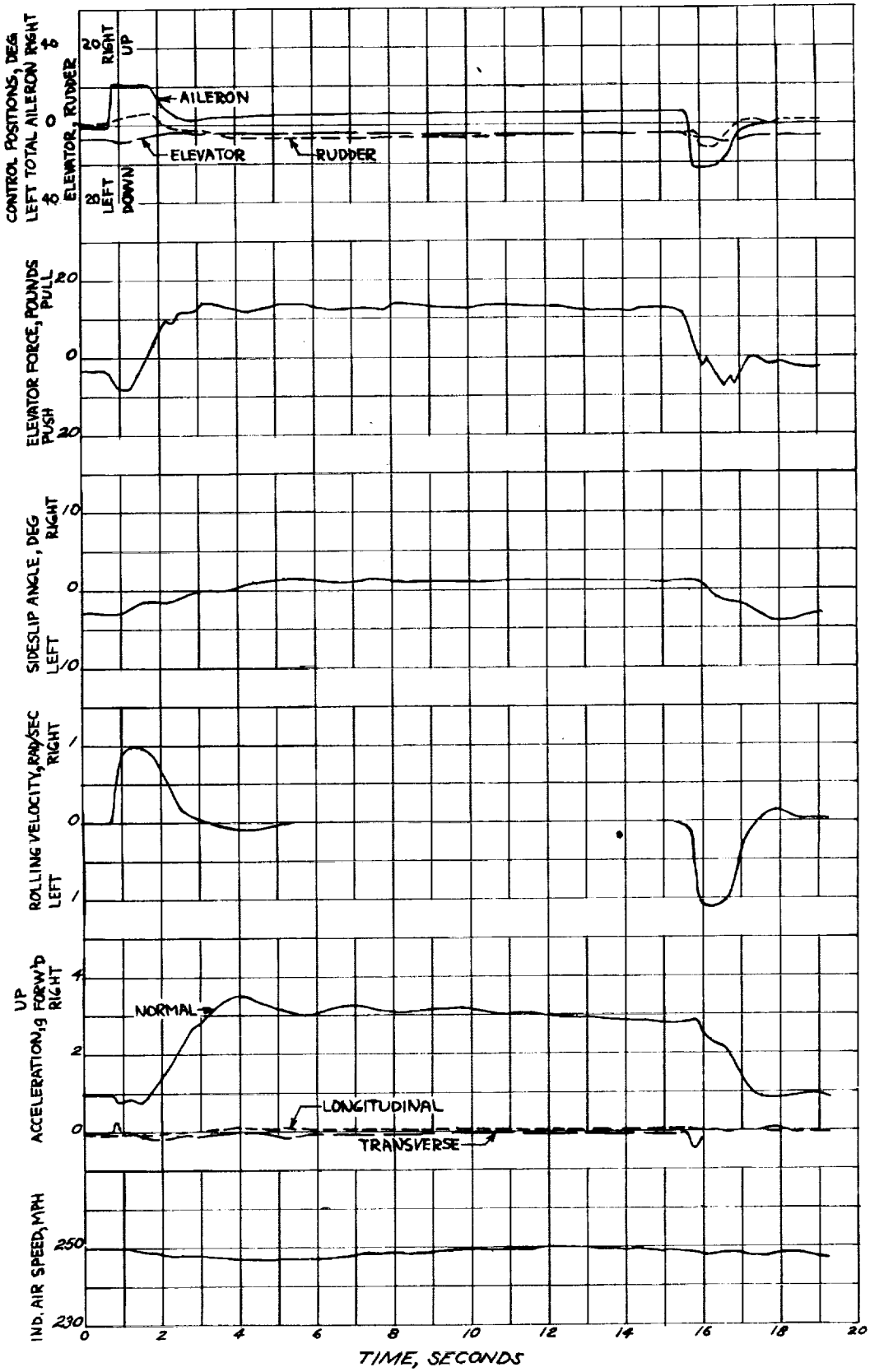

Figure 18. - Time history of a rapid 180° turn to the right started at 250 miles per hour. Supermarine Spitfire airplane with gun ports covered.

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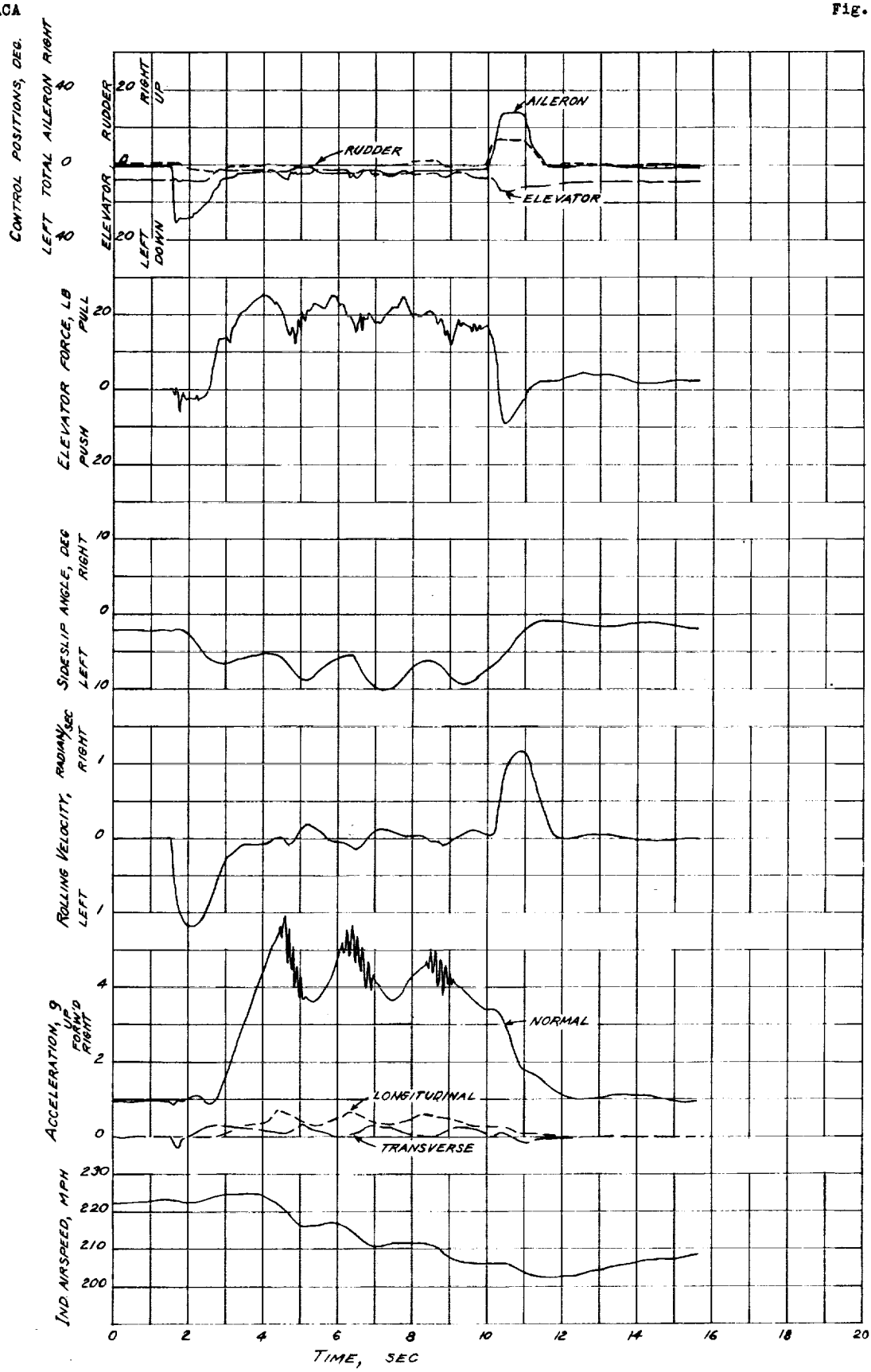

Figure 19. - Time history of a highly accelerated left turn in which the stall was reached three times. Supermarine Spitfire airplane with gun ports covered.

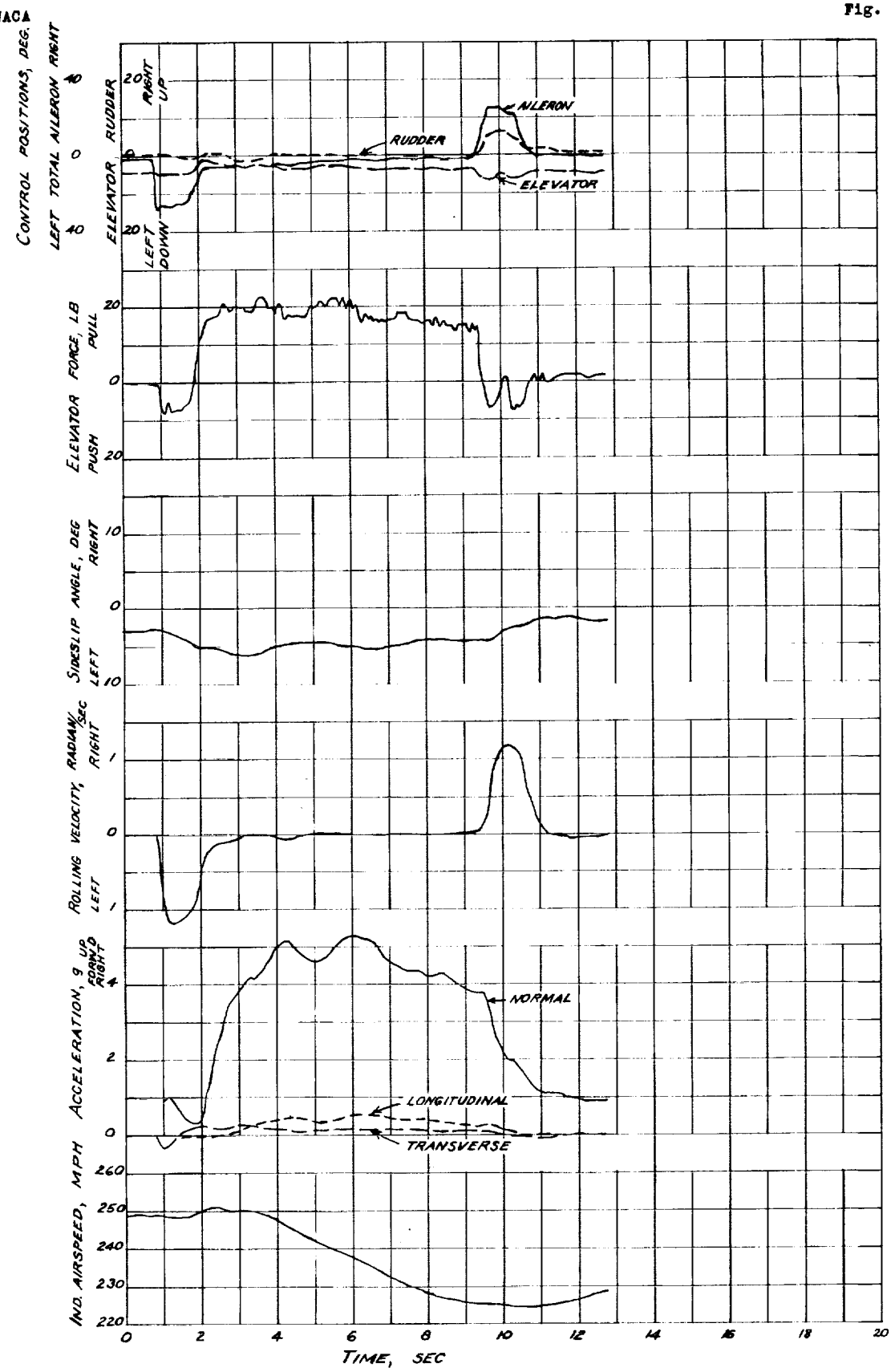

Figure 20. - Time history of a highly accelerated left turn started at 249 miles per hour. Supermarine Spitfire airplane with gun ports covered.

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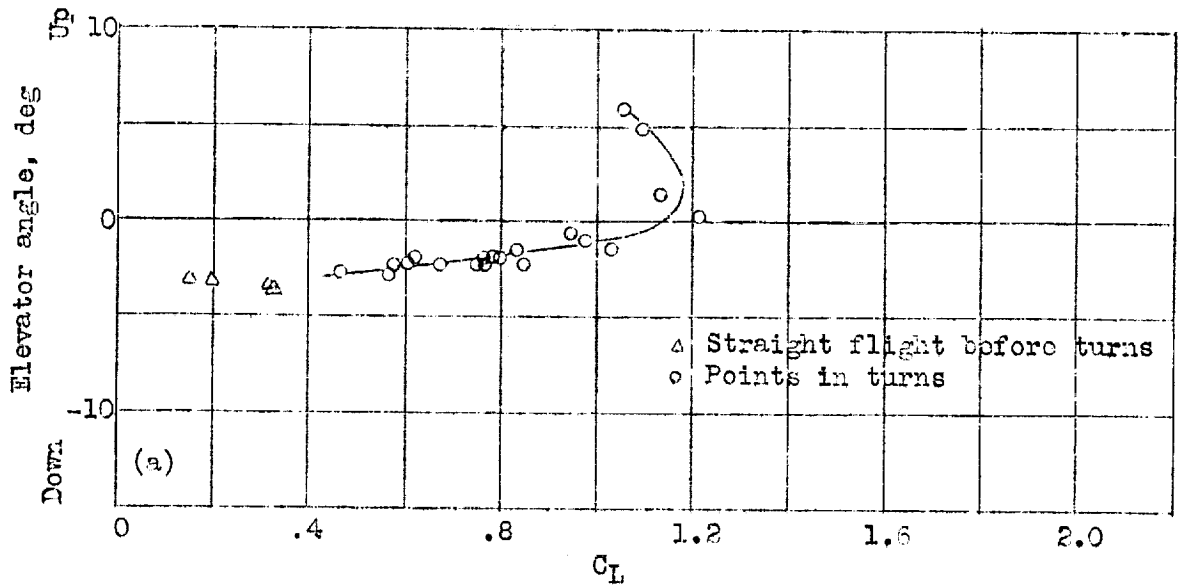
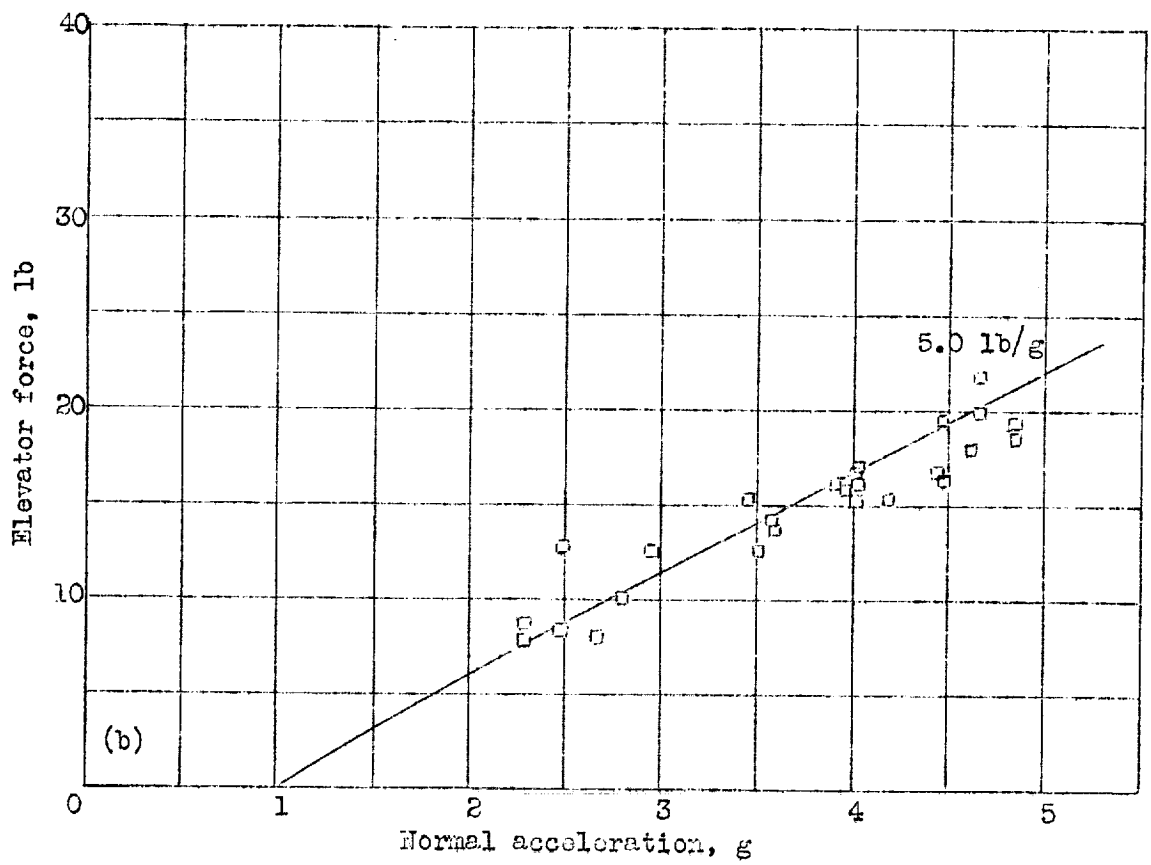


(a) Variation of elevator angle with lift coefficient in turns



(b) Variation of stick force with normal acceleration in turns.

Figure 21.- Elevator characteristics in turns. Supermarine Spitfire airplane.

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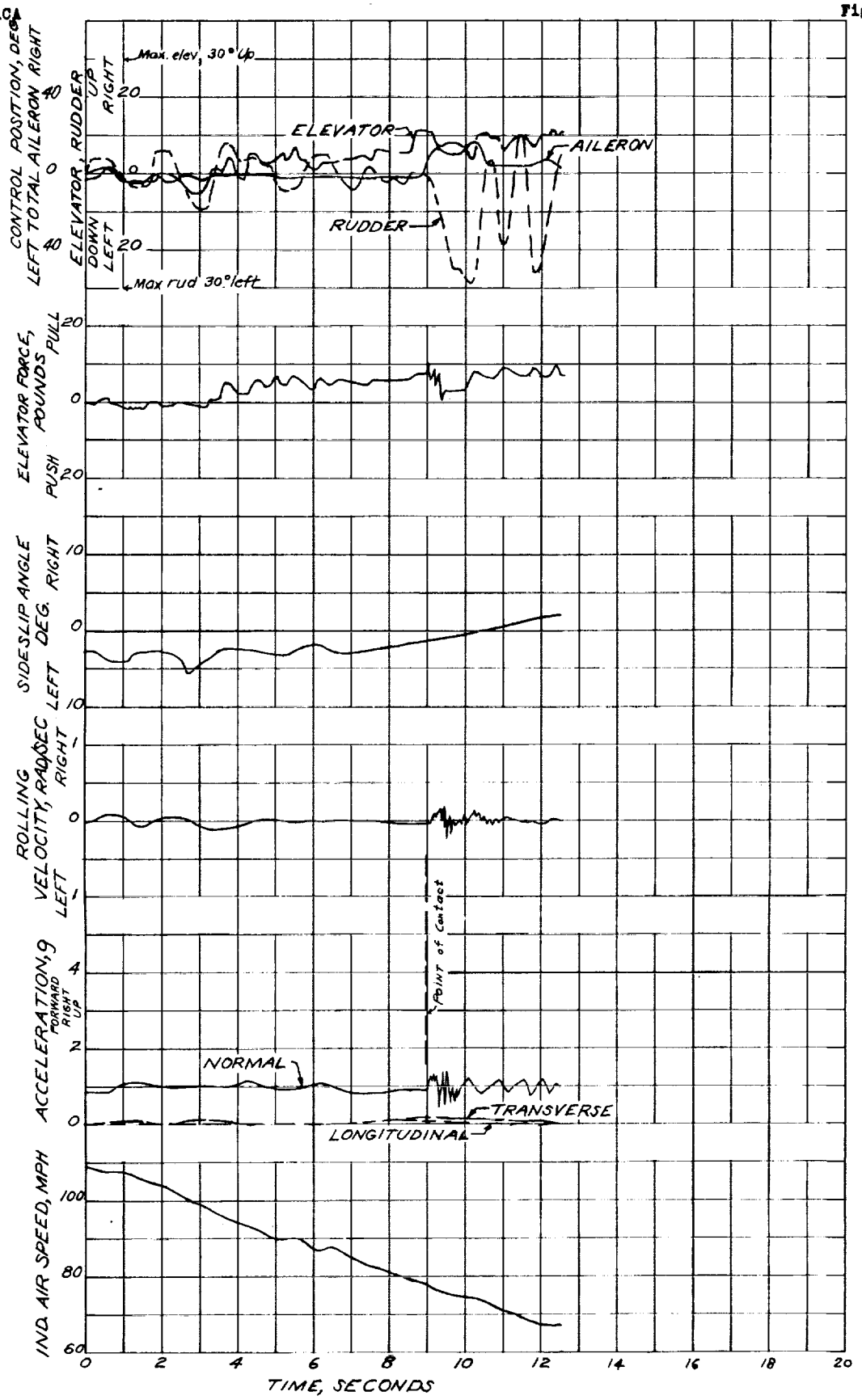

Figure 22. - Time history of a three-point landing (trim tab, 2.3° nose heavy). Note unsteadiness of airplane and controls caused by tail buffeting and instability in the approach. Supermarine Spitfire airplane.

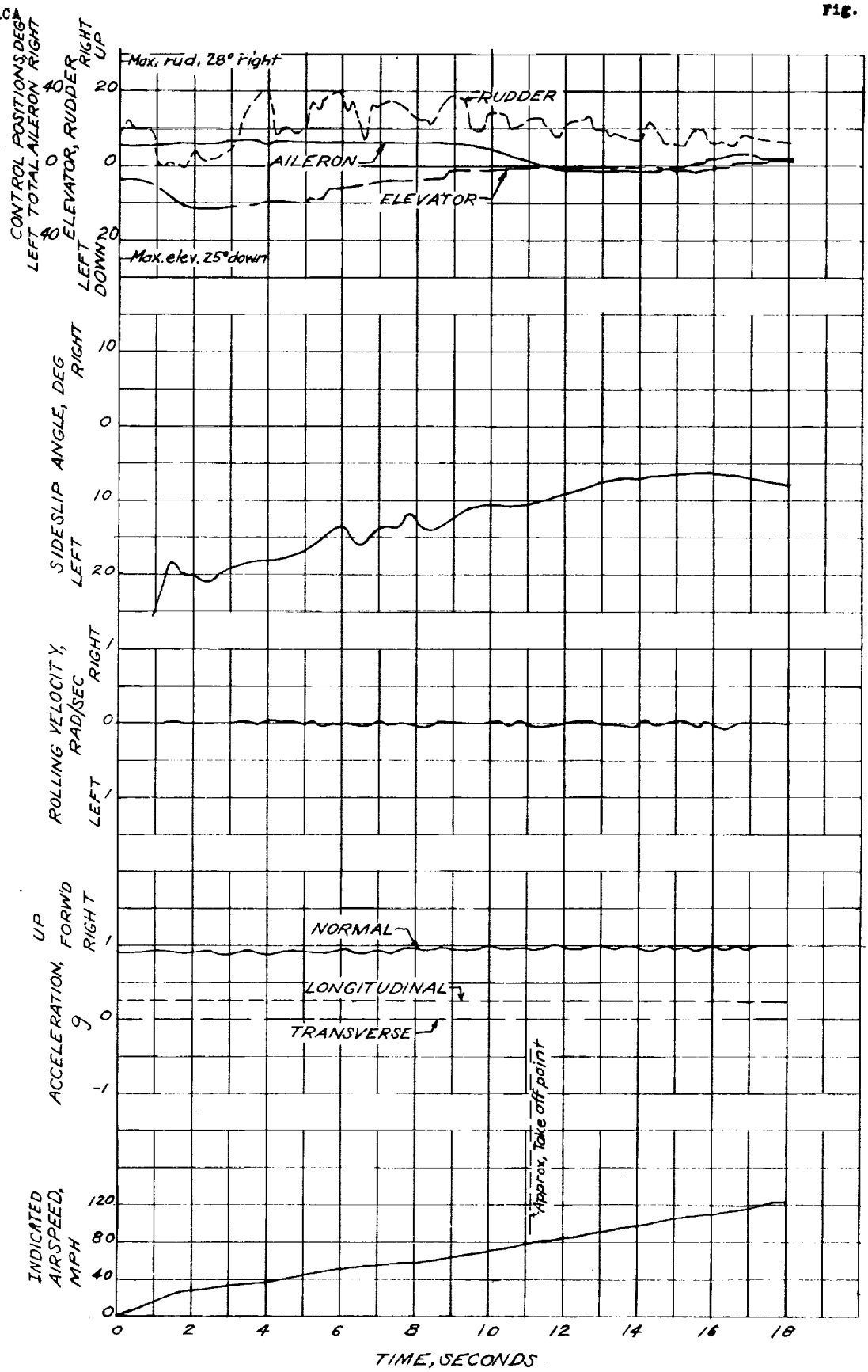

Figure 23. - Time history of a tail-high take-off made with 48 in. Hg manifold pressure. Supermarine Spitfire airplane.

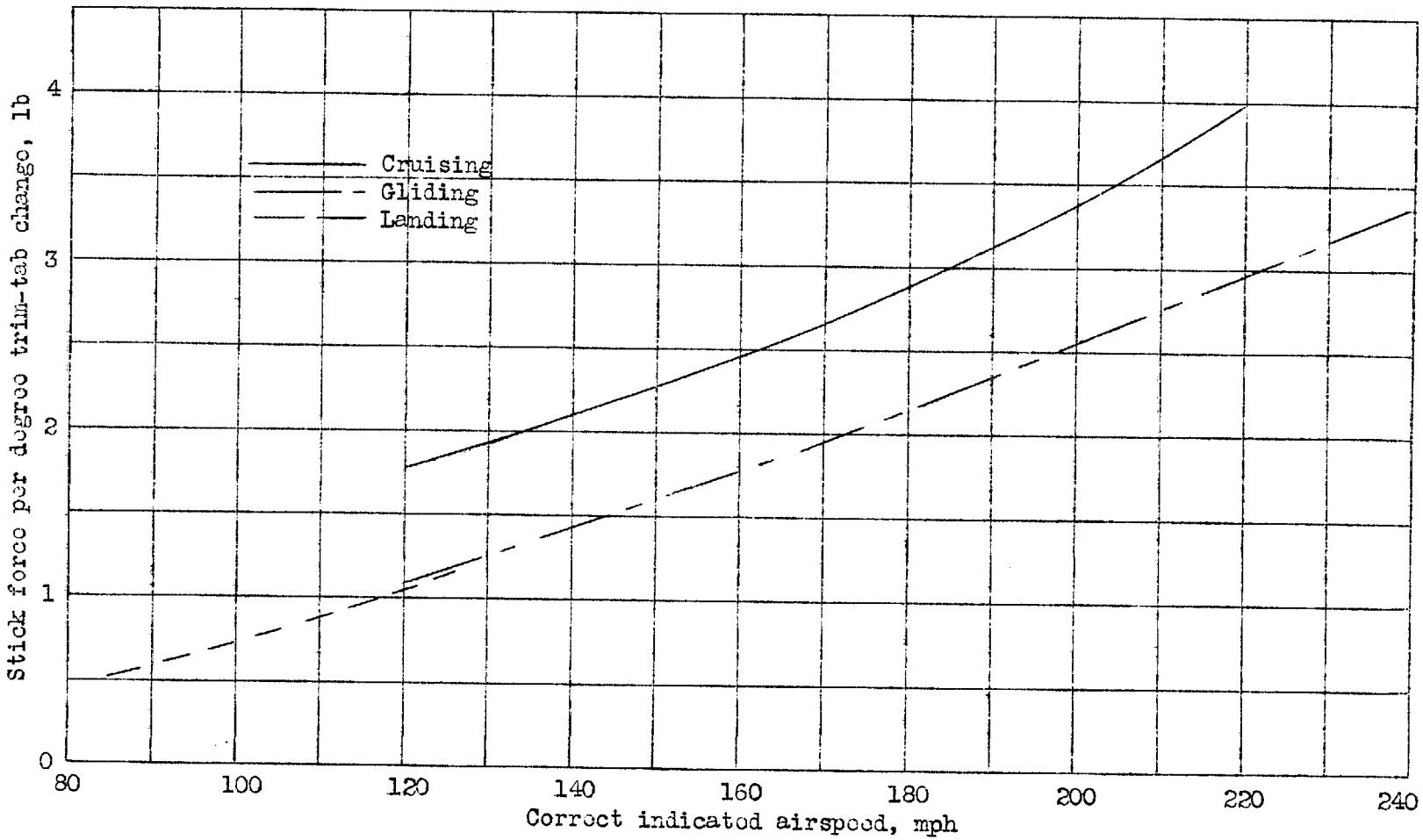

Figure 24.- Variation of power of the elevator trim tabs with indicated airspeed in the cruising, gliding, and landing conditions. Supermarine Spitfire airplane.

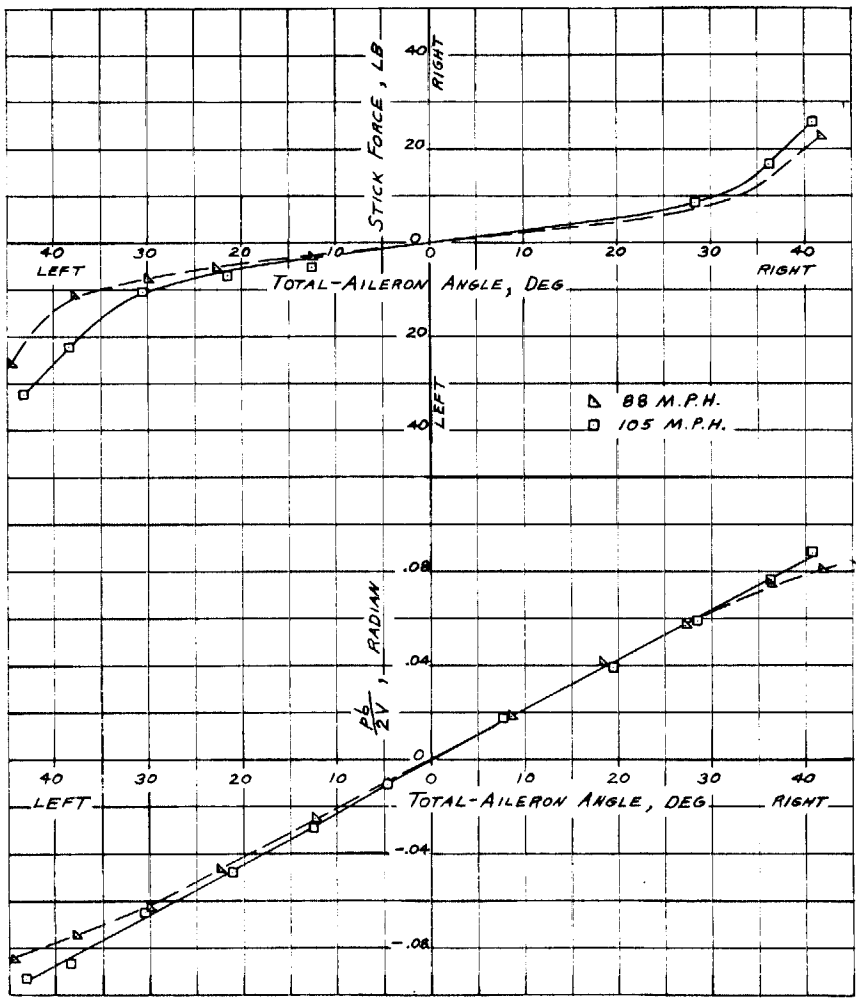

Figure 25. - Variation of aileron force and pb/2V with total aileron deflection in the landing condition (flaps down, gear down, power off). Supermarine Spitfire airplane.

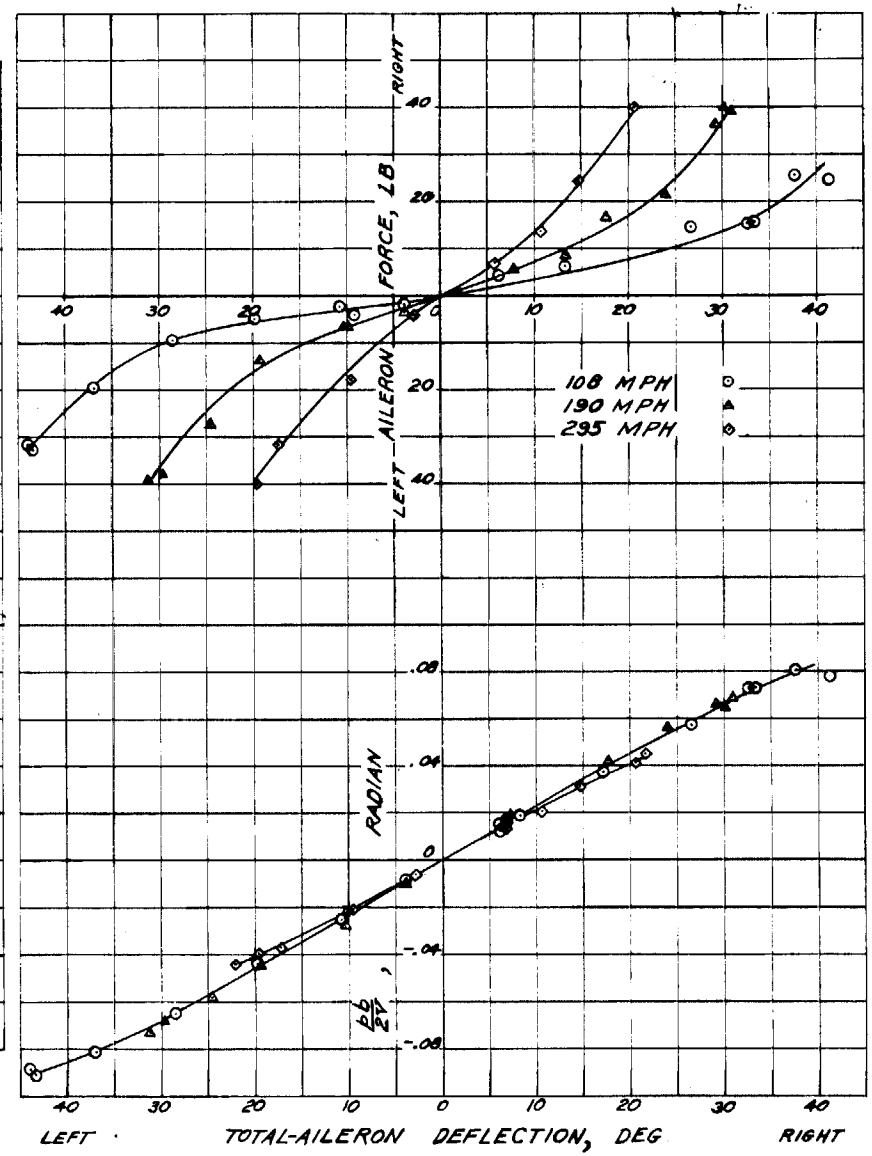

Figure 26. - Variation of aileron force and pb/2V with total aileron deflection in level flight (flaps up, gear up). Supermarine Spitfire airplane.

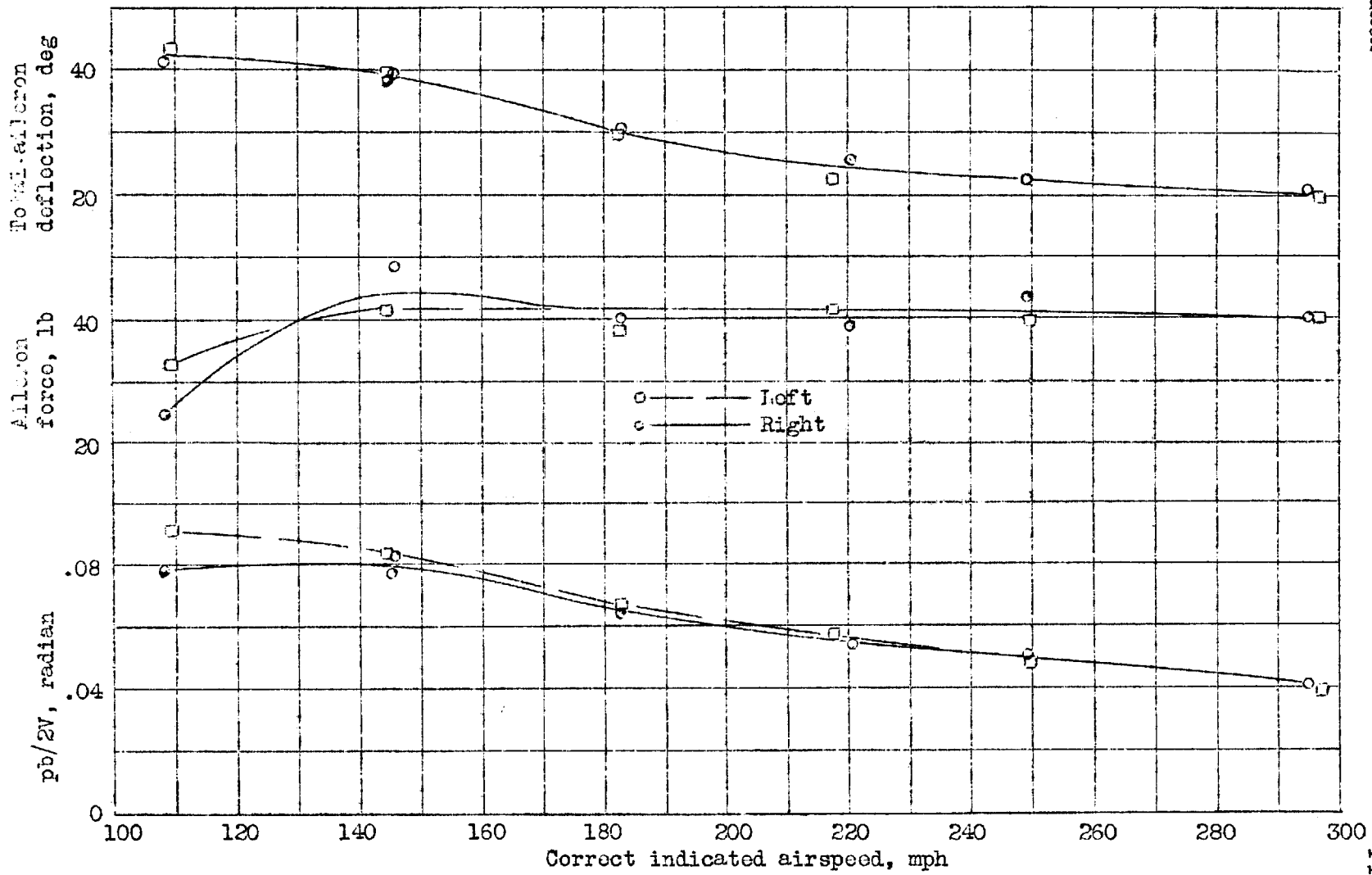

Figure 27.- Variation of maximum obtainable aileron deflection, aileron force, and pb/2V with indicated airspeed in level flight (flaps up, gear up). Supermarine Spitfire airplane.

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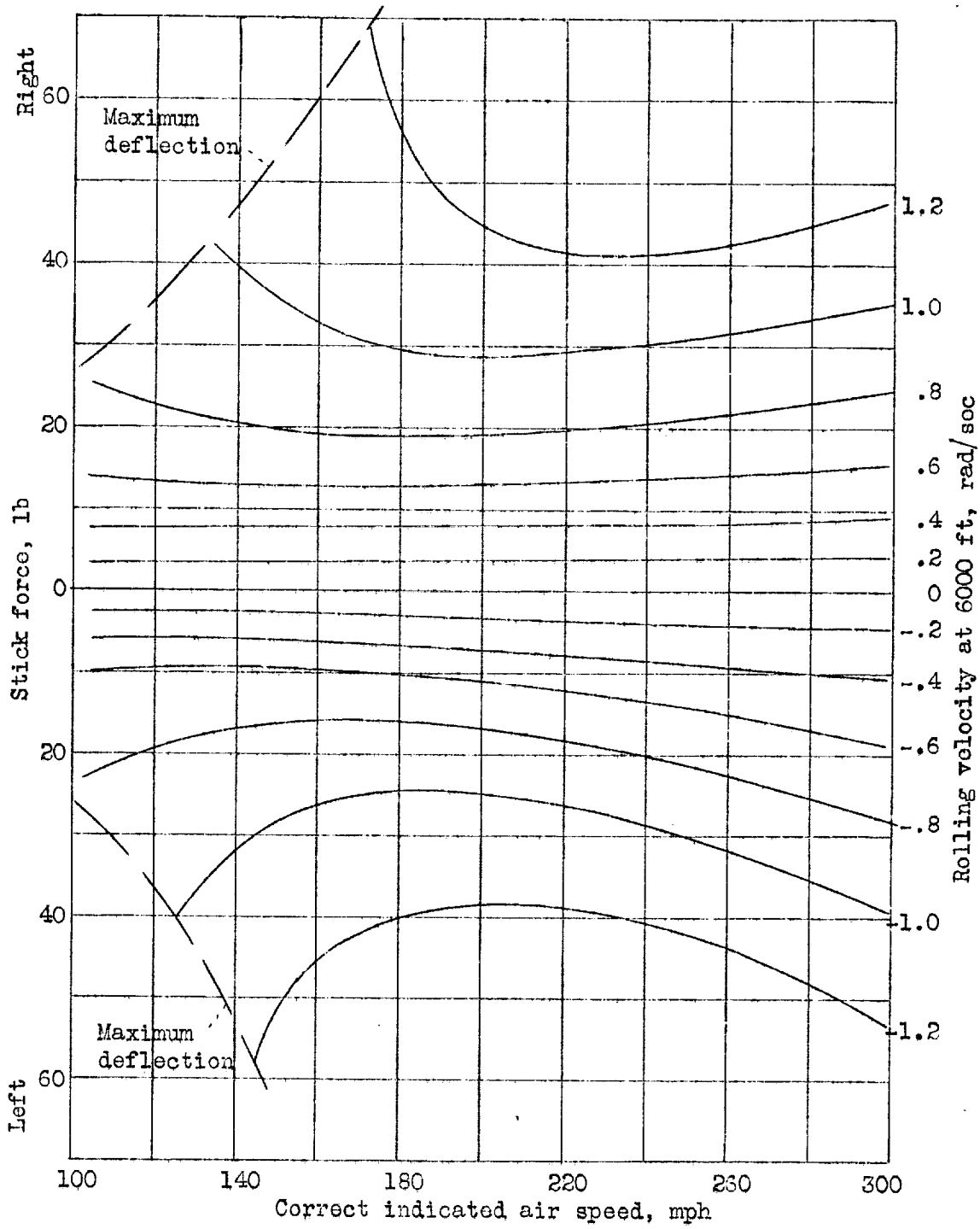

Figure 28.- Variation of aileron stick force with speed for different rolling velocities in level flight (flaps up, gear up). Supermarine Spitfire airplane.

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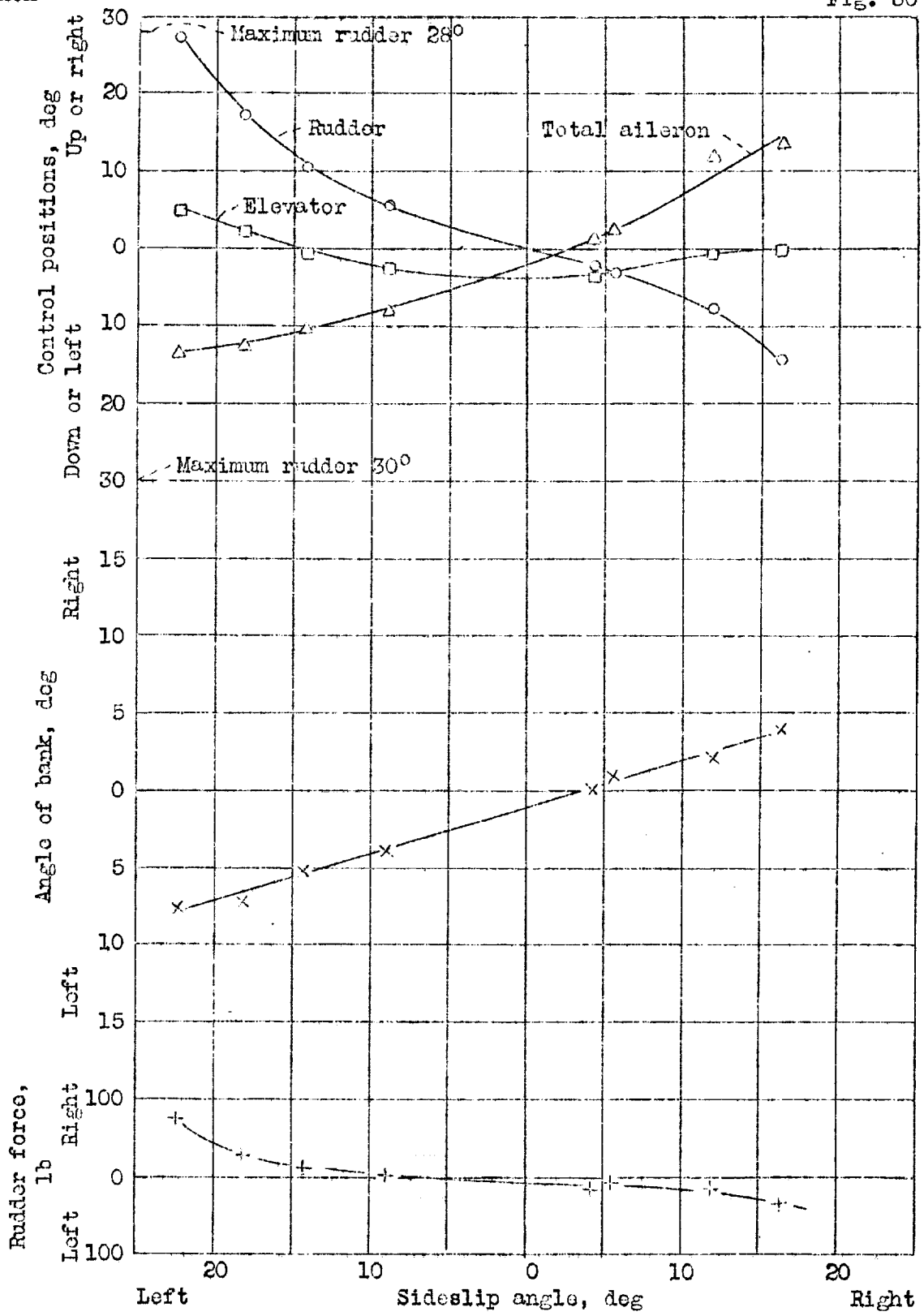

Figure 30.- Steady sideslip characteristics in the gliding condition (flaps up, gear up, power off) at 107 miles per hour, Supermarine Spitfire airplane.

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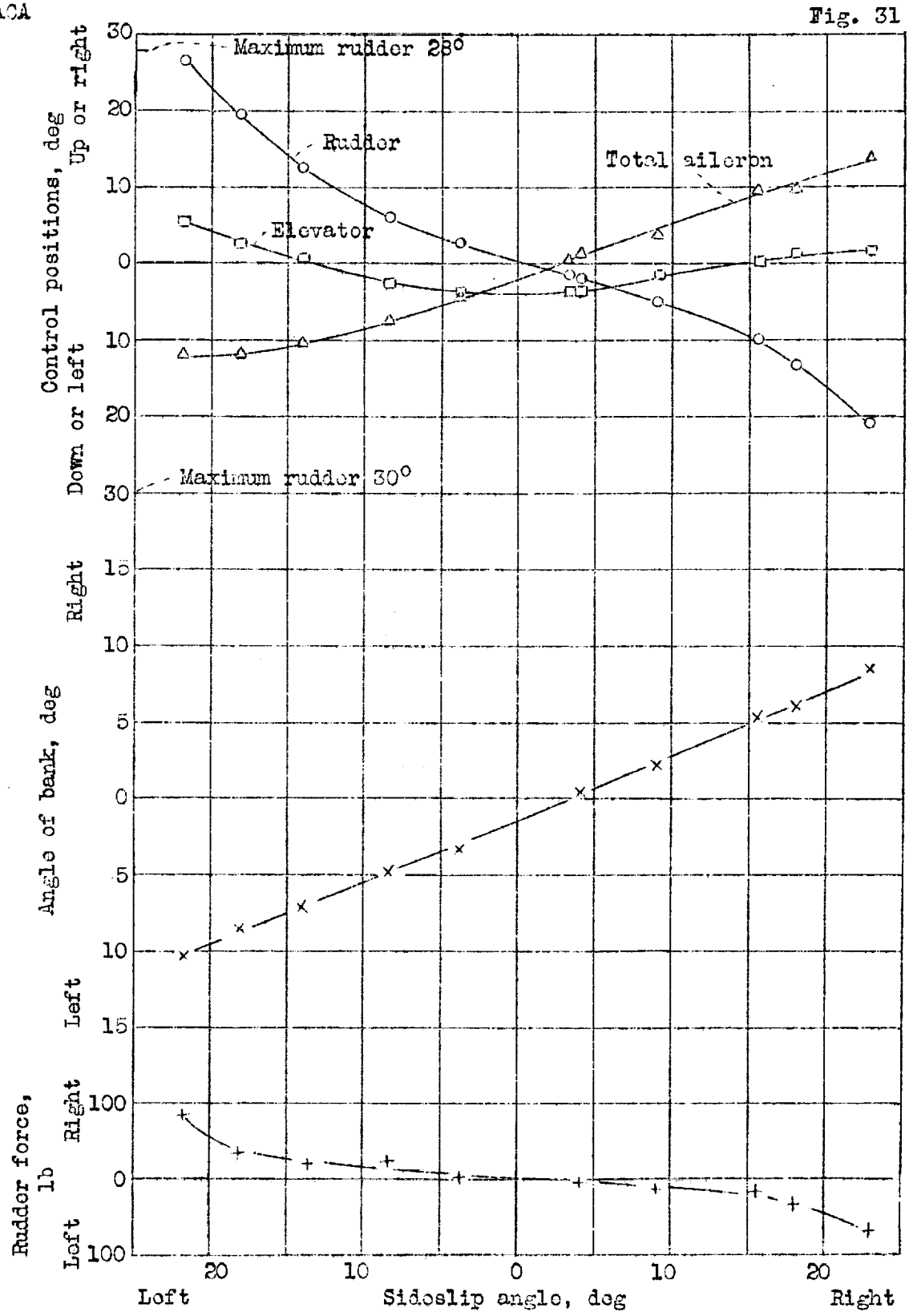

Figure 31.- Steady sideslip characteristics in the gliding condition (flaps up, gear up, power off) at 122 miles per hour. Supermarine Spitfire airplane.

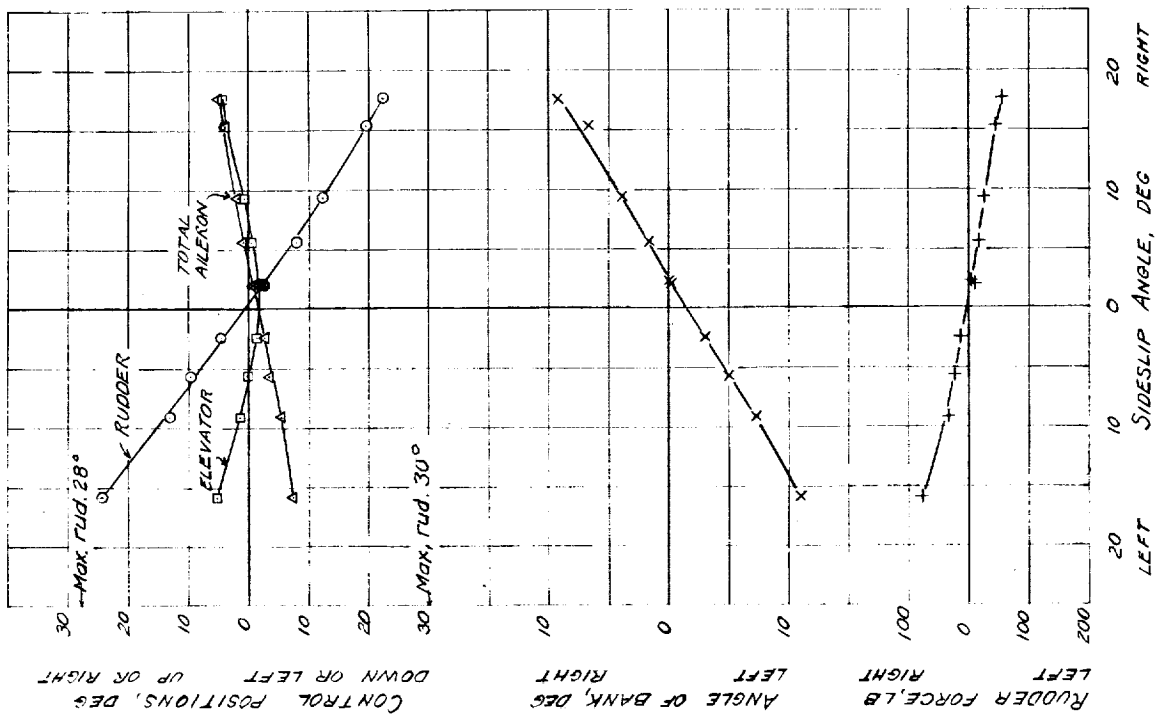

Figure 33. - Steady sideslip characteristics in the landing condition (flaps down, gear down, power off) at 124 miles per hour. Supermarine Spitfire airplane.

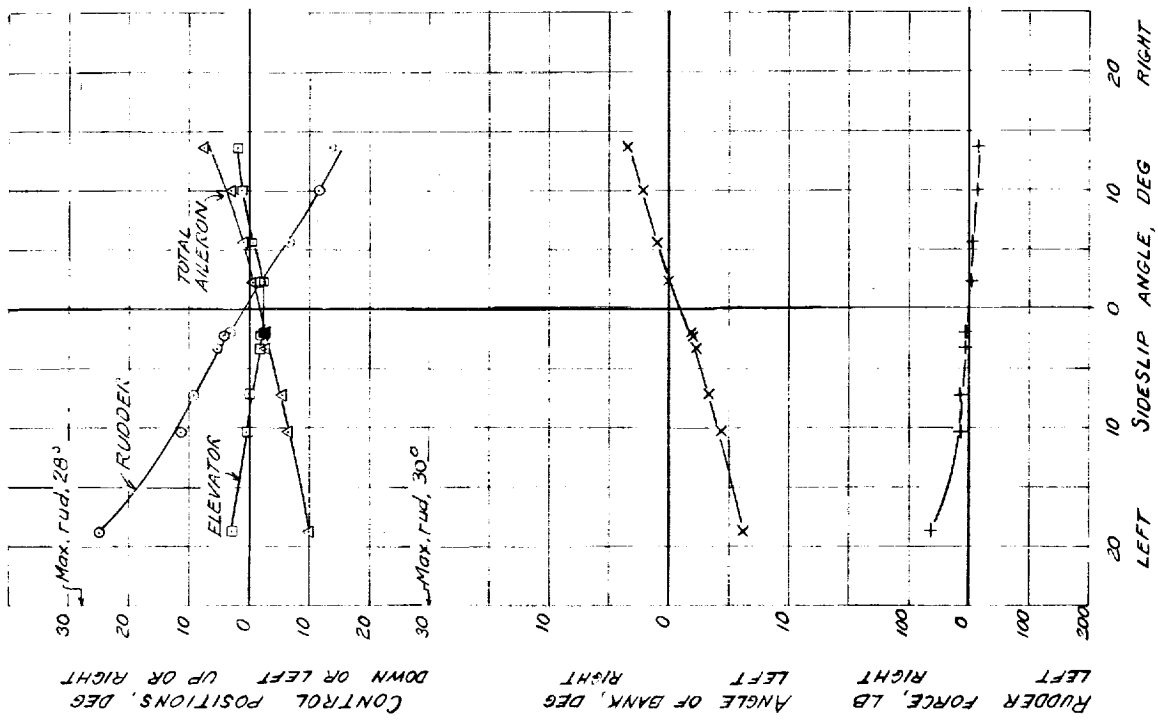

Figure 32. - Steady sideslip characteristics in the landing condition (flaps down, gear down, power off) at 95 miles per hour. Supermarine Spitfire airplane.

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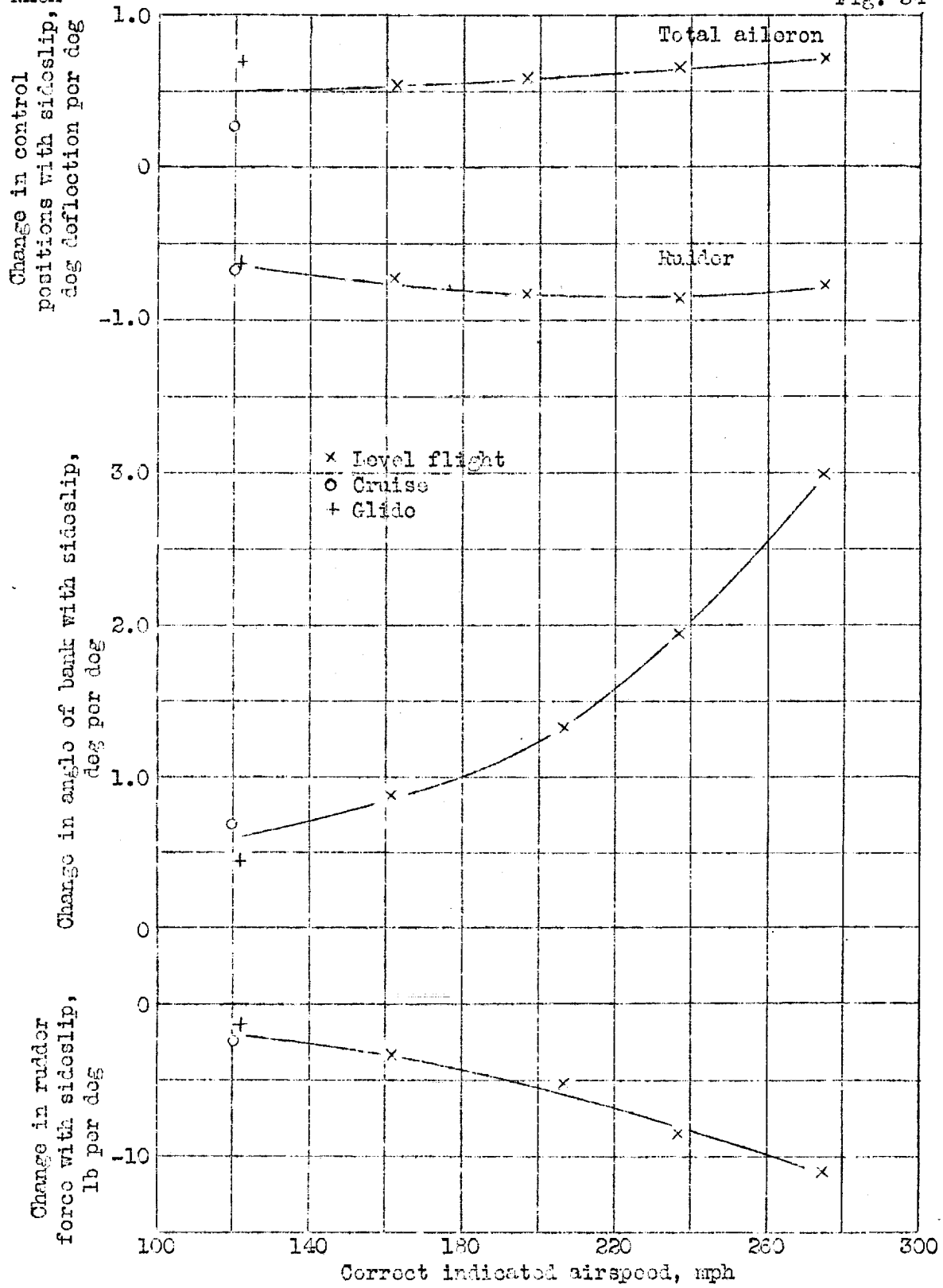

Figure 34.- Steady sideslip characteristics for small rudder deflections at various speeds in level flight with flaps and gear up, hood closed. Supermarine Spitfire airplane.

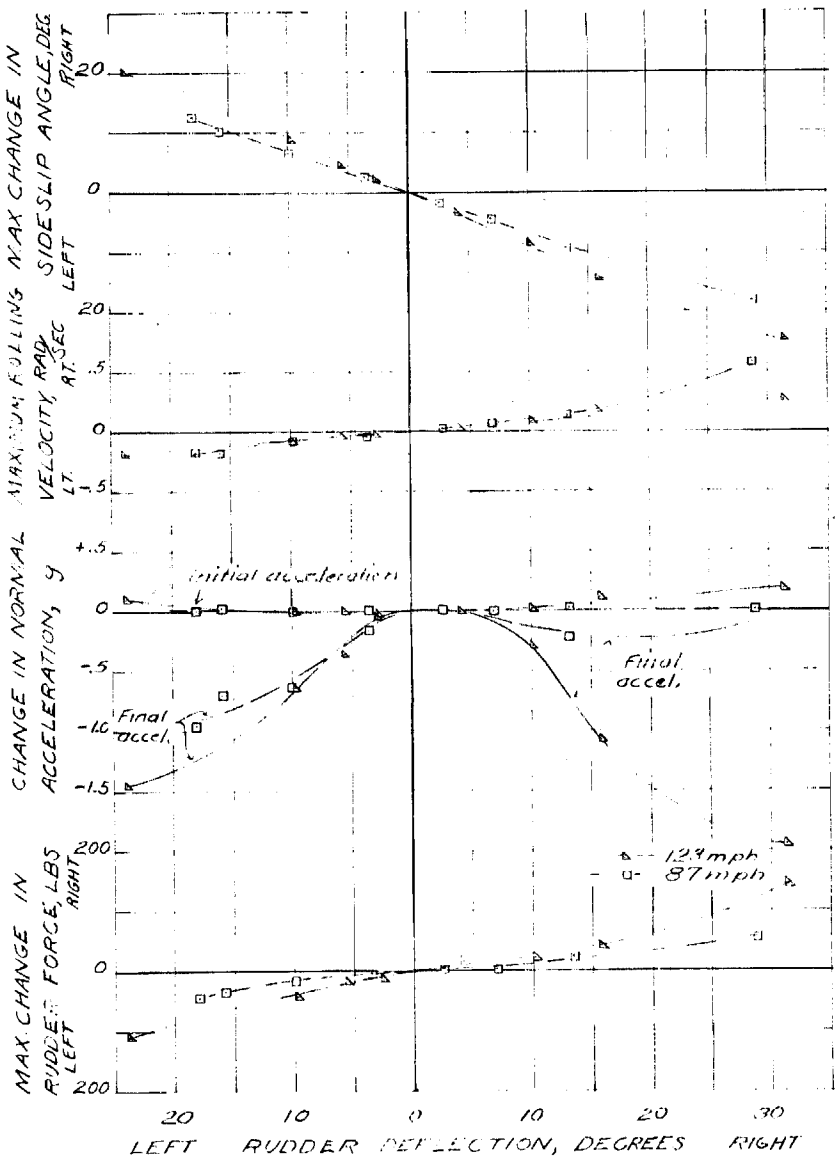

Figure 35. - Variation of maximum change in sideslip angle, rolling velocity, change in normal acceleration, and change in rudder force, with rudder deflection in abrupt rudder kicks in the landing condition. Supermarine Spitfire airplane.

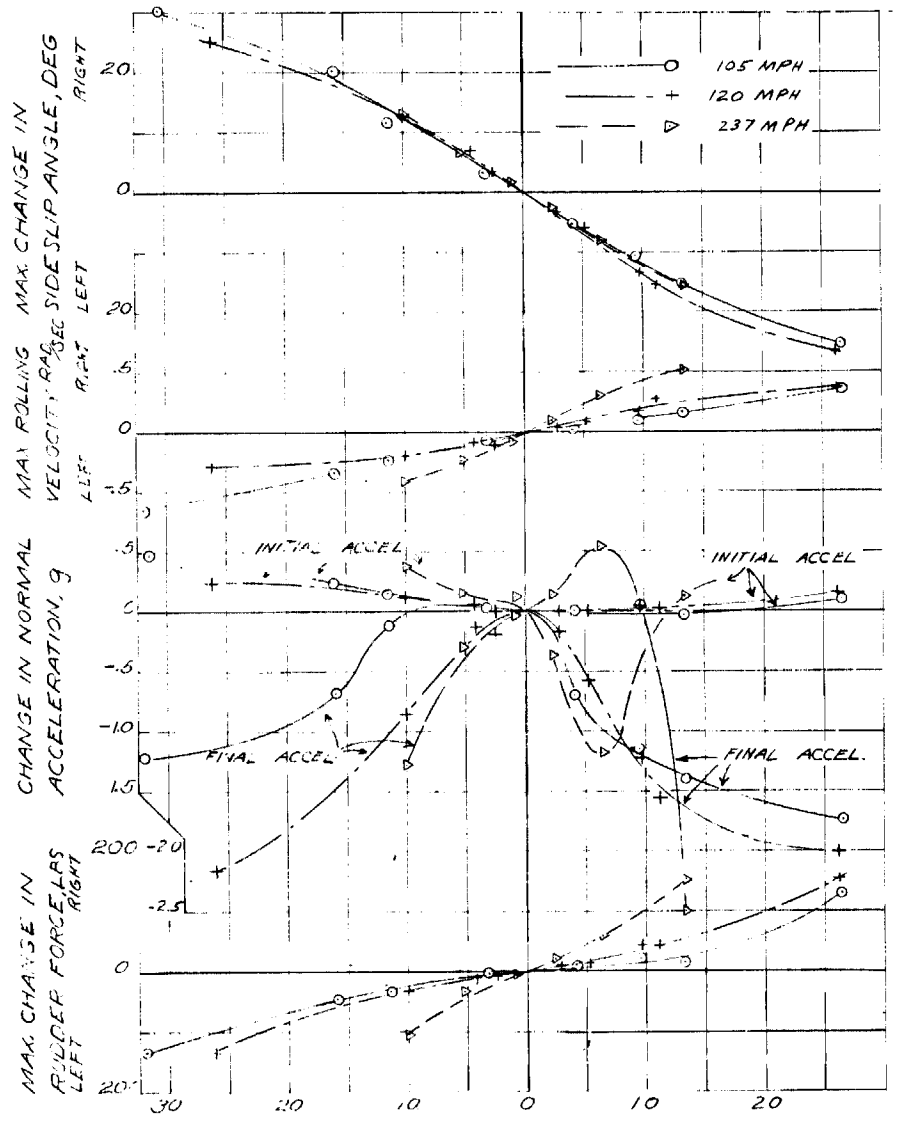

Figure 36. - Variation of maximum change in sideslip angle, rolling velocity, change in normal acceleration, and change in rudder force, with rudder deflection in abrupt rudder kicks in level flight. Supermarine Spitfire airplane.

TITLE: Measurements of the Flying Qualities of a Supermarine Spitfire va Airplane

AUTHOR(S): Phillips, W. H.; Vensel, J. R.
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ABSTRACT:

Airplane is a single-place, single-engine, low-wing, cantilever monoplane with retractable landing gear and partial wing span split flaps. General specifications are given. All flight tests were made with cg 31.4 inches behind leading edge of wing at root. Mean aerodynamic chord of 85 inches was computed to be 4.80 inches back of leading edge of wing at root. Flying qualities observed by these tests may be summarized in terms of accepted standards for satisfactory flying qualities.

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