ACR No. 3D15

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WARTIME REPORT

ORIGINALLY ISSUED
April 1943 as
Advance [redacted] Report 3D15

OBSERVATIONS OF COMPRESSIBILITY PHENOMENA

IN FLIGHT

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WASHINGTON

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SUMMARY

Several aspects of compressibility phenomena were examined under flight conditions on the XF2A-2 airplane, which was subjected to a series of dives and pull-outs at speeds up to 550 miles per hour corresponding to Mach numbers up to 0.74. The results indicate that the principal effects of compressibility shown by wind-tunnel tests at supercritical velocities are also manifested under flight conditions. One apparent discrepancy appeared, in that the flight results did not indicate an increase in wing pitching-moment coefficient with increasing Mach number; this discrepancy, however, is attributed to the surface roughness of the airplane wing and is not believed to be an essential contradiction of wind-tunnel results.

Although no serious vibration or other manifestations of the compression shock were observed, the position of the shock suddenly shifted slight amounts in a few cases and the possibility of appreciable shifting or oscillation under some conditions is not precluded.

Experiences during the course of the tests indicated that considerable danger exists at high speeds if airplanes are longitudinally unstable and also that careful attention must be given to compressibility effects on aerodynamically balanced control surfaces to avoid sudden overbalancing at high speed.

INTRODUCTION

It is a well-known fact that maximum airspeeds attainable by certain classes of airplane have been increased to values that would, in the light of wind-tunnel data, be expected to give rise to the formation of the so-called "compression shock" on the wings. Inasmuch as the exact nature of this compression shock is not clearly known, serious questions of immediate importance have been raised as to the behavior of an airplane subjected to it, particularly in regard to the possibility that the shock may have a violently unsteady character (as indicated in some wind-tunnel tests) which would be presumed to cause severe wing vibration and tearing of the skin or failure of rivets due to the rapid shifting of a marked discontinuity between very high and lower pressures. Many other questions of a less serious nature, but of perhaps greater importance, have also been raised with respect to the forces on and the behavior of aerodynamic forms at high speed. Because the wind tunnels have been and will necessarily continue to be the primary source of data and because the phenomena of flow in the regime of compressibility is still so imperfectly understood, it is natural that these questions include that of the reliability of wind-tunnel data when applied to the free-air and to larger-scale conditions of flight.

Because of the existence of these important questions, tests devised to permit study of compressibility effects in flight have been felt to be necessary. Such tests were therefore initiated jointly by representatives of the Bureau of Aeronautics, Navy Department, and of the National Advisory Committee for Aeronautics in discussions held at Langley Field, Va., on September 20, 1939.

The nature of the tests was decided upon after consideration of the relative importance of various possible objectives and of the practical difficulties likely to be encountered during the tests. It was finally concluded that these first tests should be devised to give, even at the expense of accuracy, the most comprehensive picture consistent with safety and practical expediency of the behavior of an actual airplane at high speed rather than to concentrate on very difficult, problematical, and somewhat academic objectives as, for example the accurate measurement of profile drag for quantitative comparison with wind-tunnel results. In other words, the objective of the tests was to find in a general way whether or not the principal features of compressibility phenomena as shown by wind-tunnel tests would be borne out under flight conditions, particularly in regard to the unsteady character of the compression shock, and to observe any peculiarities in the behavior of the airplane at speeds near the critical speed.

After some delay in obtaining a suitable airplane, the tests were finally conducted in accordance with these objectives during May and June 1940. This paper was originally issued as a memorandum report for the Bureau of Aeronautics, August 29, 1940.

APPARATUS AND METHOD

Airplane.- The airplane used in the tests was the Brewster XF2A-2, which was made available by the Bureau of Aeronautics. A photograph of this airplane is shown in figure 1. The wing surfaces of this airplane were somewhat rough and irregular, with exposed rivet heads and waves in the skin.

All easily removed equipment not essential for the tests was taken out to provide space and to compensate for the weight of recording instruments subsequently installed. The Curtiss electric propeller was provided with a feathering circuit in order that the airplane could be dived to maximum possible airspeed without overspeeding the engine. This installation required the removal of the propeller cuffs, which interfered with the cowling at high pitch settings.

As vibration tests and flutter calculations, made under the direction of Dr. T. Theodorsen of the Committee's staff, had indicated a possibility of wing flutter in the bending-aileron mode at an indicated speed of about 450 miles per hour, the ailerons were made slightly noseheavy about the hinge line as a precautionary measure. This overbalancing was accomplished by attaching thin (1/32 in.) narrow lead sheets to the leading edges with fabric and dope in a manner used by the Army Materiel Division at Wright Field. (See fig. 2.) The estimated flutter speed was thereby raised to nearly 600 miles per hour. Although there was no reason to anticipate flutter of the tail surfaces, the elevators and rudders were nevertheless balanced in substantially the same manner as a further precaution.

With these changes and with the airplane otherwise ready and serviced for the test flights, the weight, including pilot and parachute, was about 5450 pounds and the center-of-gravity position was located at 23.3 inches back of station 0, or at 28 percent of the mean aerodynamic chord. As will later be indicated, the airplane was longitudinally unstable in this condition. The center of gravity was therefore shifted forward 2.2 inches to 25 percent of the mean aerodynamic chord by removal of the tail hook and its supporting structure and by shifting the instrument batteries to more forward locations. In this condition the airplane appeared to be satisfactorily stable.

Instruments.- In line with the objectives of the test, the instrumentation was planned to permit the most accurate practicable determination of the Mach number, the critical speed, and the behavior of the compression shock, as well as to permit observation of several features of the behavior of the airplane with the occurrence of the shock. The instruments included:

1. Airspeed recorder connected to swiveling pitot-static tube
2. A special "pressure-plate" airspeed recorder designed for high critical speed and elimination of time lag
3. Stem thermometer
4. Thirty-cell recording manometer
5. Two-cell recording manometer
6. Statoscope (sensitive altimeter)
7. Two-component accelerometer (Z and X components)
8. Control-force recorder (elevator and ailerons)
9. Control-position recorder (elevator)
10. Two 35-millimeter motion-picture cameras
11. Vibrograph
12. Tachometer
13. Timer
14. V-G recorder

These instruments were mounted as shown in figure 3.

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Speed measurement.- Because of the uncertainty involved in balancing the pitot-static tube against lag effects for the unusual conditions of these tests and in the critical speed of the pitot tube, it was considered advisable to design a special airspeed recorder that would not be subject to lag or compressibility effects. The recorder designed and built for this purpose, and called herein the "pressure-plate recorder," consisted essentially of a flat disk with sharp edge mounted normal to the air stream on a shaft restrained by spring diaphragms to small longitudinal motions. The air force on this plate or disk was determined by recording the shaft displacement relative to the instrument frame by two oppositely mounted recording strain gages. Calibration of the instrument in the NACA 24-inch high-speed wind tunnel provided means for establishing the airspeed from the force measurement. In addition to the pressure-plate recorder, a swiveling pitot-static tube connected to a standard NACA recorder was also used, and the recorder was mounted in the wing to keep the pressure lines short and thus to minimize the lag errors. The pressure lines were balanced in the usual way.

Further airspeed records were obtained with the V-G recorder and by photographing, with a motion-picture camera, the instrument panel. The airspeed indicator and the V-G recorder were both connected to the fixed pitot-static tube.

In order to permit correction of the airspeed readings for compressibility and the evaluation of true airspeed and Mach number, it was necessary to determine the static pressure, the temperature, and the air density at each instant during the dives. For this purpose, measurements of temperature and pressure altitude were made during a succession of short, level-flight runs at several altitudes during the climb preceding each dive. From these surveys it was possible to compute relations between absolute altitude, pressure, and density up to the starting points of the several dives. At the starting point of each dive, the statoscope valve was closed and measurements were taken of the time variation of the pressure altitude lost to the initial altitude at each instant. From these measurements and from the previously made atmospheric soundings, it was possible to establish the temperature, pressure, and air density at any instant during the dives.

As an incidental check on true airspeed, and in order to make good the atmospheric sounding without relying solely on the statoscope readings, attempts were made to establish the position of the airplane in space at each instant through the use of double phototheodolite measurements from a long base line. These measurements were made through the courtesy of the Coast Artillery Board at Fort Monroe, Va., and of the Mitchell Camera Corporation of Pasadena, Calif. Owing to the difficult conditions of the tests, these attempts were only partly successful and no results could be obtained in the longer dives.

Compression shock observations.- As a means of detecting any longitudinal shifting of the region of pressure discontinuity (compression shock) and of determining the critical speed for at least one wing section, pressure measurements were made along upper and lower surfaces of the section of the right wing located at $103\frac{1}{4}$ inches from the plane of symmetry (fig. 3(a)). This section conforms closely to a 23014 section; the measured profile is shown compared with a 23015 section in figure 4.

Each pressure orifice was connected by rubber tubing of $\frac{1}{4}$-inch inside diameter, approximately 15 feet long, to one side of a manometer cell (fig. 5). The opposite side of each cell was connected to a reservoir vented to an orifice in the lifting tube in which, as shown by preliminary measurements, approximately true static pressure obtained. The actual pressure in the lifting tube was, however, measured with respect to the pressure at the static orifices of the swiveling pitot-static tube, and the wing pressures relative to true static pressure could consequently be more closely approximated.

In conformity with wind-tunnel practice and as a further means of detecting the compression shock, a total-head tube was mounted 2 inches above the wing, well behind the station of maximum thickness adjacent to the pressure section. The total pressure at this location was measured relative to the total pressure at the swiveling pitot-static tube. As measurements made through a fairly large range of angles of attack in level flight showed substantial balance of the total pressures, it was evident that the wing total-head tube lay outside the boundary layer and it was therefore presumed that any sudden or large differences in pressure appearing in the dives could be attributed to heat energy losses in the compression shock. For the last dive, an additional total-head tube was installed on the wing 4 inches above the surface.

In addition to the pressure distribution and the total-head measurements, photographs of the upper surface of the wing were made with a motion-picture camera synchronized with the other instruments. This procedure was followed on the chance that changes in skin-wrinkle pattern resulting from shock phenomena might be observed and also in order to detect any changes in profile or surface irregularities that would affect the nature of the flow. For this purpose a narrow longitudinal stripe was painted on the wing at the test section and two triangular reference markers were installed on the upper surface of the wing at the main spars.

A vibrograph was improvised and installed near the right wing tip for these tests. This instrument was essentially a low-frequency, spring-restrained pendulum, of which the movement relative to the wing was recorded by a stylus on a smoked disk driven by clockwork.

<u>Other measurements</u>.- A two-component accelerometer was used to measure the normal and longitudinal components of acceleration. The longitudinal component was used to permit an approximate determination of the total drag in the later stages of the dives. The instrument board on which the accelerometer was mounted made an angle of 5° with the thrust axis and about 18° with the ground when the airplane was in the normal landing attitude.

Measurements of elevator control force and displacement were also made in order to observe the longitudinal behavior of the airplane, particularly in regard to the control forces required to maintain balance in the dives and to recover at different values of Mach number. The aileron control forces were also measured, and large aileron movements were observable by the wing camera.

The recording tachometer mentioned was intended primarily as an instrument for measuring the propeller speed in connection with terminal velocity calculations. As previously indicated, however, the propeller was substantially feathered during the dives and the propeller speeds were, except in the last dive, below the minimum limit (500 rpm) of the tachometer range. In this dive the engine revolution speed temporarily reached values as high as 700 rpm.

FLIGHT PROGRAM

The flight program was conducted in three phases as follows:

1. Preliminary flights to check instrument operation, to calibrate airspeed installations, to locate suitable static pressure reference points, to check the position of the total-head tube relative to the boundary layer, and to obtain pressure-distribution data at low speeds.

2. Practice dives for familiarization, to check the behavior of the airplane at increasingly severe speeds and accelerations, and to calibrate the indicating accelerometer against the V-G recorder.

3. Contract, or test, dives at three speeds to obtain complete data through a range of Mach numbers and lift coefficients.

The preliminary flights were made by NACA pilots, and the practice and test dives were performed by Mr. E. W. Burke of the Brewster Aeronautical Corporation. During the course of the practice dives and subsequently, however, a number of dives were made by Messrs. M. N. Gough and W. H. McAvoy of the NACA to check the longitudinal stability and other characteristics of the airplane.

As a matter of record in connection with contract requirements, as well as to bring out the potentially serious effect of longitudinal instability at high speeds, the V-G records for the six practice dives and the six contract dives completed are shown herein as figures 6 and 7.

In the first five practice dives (fig. 6) the center of gravity of the airplane was at the 28-percent location. In this condition the airplane apparently responded satisfactorily, although the unsteadiness noted in the V-G records of the first four dives was a cause of increasing concern in view of the fact that the air was smooth in all cases. During the pull-out from practice dive 5 something occurred that is not entirely clear as the pilot "blacked out" and was not able later to recall the entire sequence of events. In any event, conscious control of the elevator was lost at some point and the airplane assumed a high angle of attack at rather high speed. The abnormality of the V-G record for this case is evident (fig. 6).

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After this experience specific tests were devised to check the longitudinal stability, and the results showed that the airplane was unstable. Repeat tests made following the change to the 25-percent center-of-gravity location indicated that in this condition an appreciable margin of stability was present. No subsequent difficulty was experienced.

The required contract dives consisted of the following:

1. Vertical dive and pull-out with a normal acceleration of 8.0g. Indicated airspeed not less than 300 miles per hour and not more than 325 miles per hour.

2. Vertical dive and pull-out with a normal acceleration of 8.0g. Indicated airspeed not less than 400 miles per hour.

3. Vertical dive to terminal velocity and pull-out with a normal acceleration of 8.0g. Dive to start at altitude not less than 30,000 feet and to continue to an altitude not greater than 12,000 feet before starting pull-out; alternatively, dive to continue 32 seconds before starting pull-out.

These dives were completed, although it was necessary to repeat the 300-mile-per-hour dive twice and the terminal dive once because of unsatisfactory operation of the recording instruments. One attempt to perform the terminal dive was not completed by the pilot because of unsatisfactory aileron operation that will be discussed in detail in the section on RESULTS.

SYMBOLS

g gravitational unit of acceleration (32.2 ft/sec)

ρ mass density of air, slugs per cubic foot

a velocity of sound, feet per second

V airspeed, feet per second

M Mach number (V/a)

P pressure coefficient

c_n section normal-force coefficient

C_D drag coefficient of airplane

W weight of airplane, pounds

q dynamic pressure

n number of g units of acceleration

S wing area, square feet

F_S stick force, pounds

Subscripts:

cr critical

max maximum

RESULTS

Time histories.- Figures 8 to 12 show the variation with time of a number of recorded quantities taken during the several dives including (fig. 10) the attempted terminal dive that was not completed because of malfunctioning of the ailerons and (fig. 11) the completed terminal dive made May 13, 1940, for which the pressure-distribution records were unsatisfactory. These figures are largely self-explanatory and require no comment at this point. Figure 13 shows several groups of simultaneous photographs taken with the wing and cockpit cameras during the period of aileron malfunctioning and will be discussed subsequently in relation to the time history of figure 10.

Pressure data.- Some results of the pressure measurements are shown in figures 14 to 16. In order to indicate most clearly the variations with Mach number, it was necessary to make comparisons at the same values of normal-force coefficient. Because it was impossible to choose comparable points directly from the records and also because of the desirability of eliminating accidental errors

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from the comparisons, the records were evaluated in the following manner:

For each dive and pull-out, the pressure distribution was plotted for uniformly spaced time intervals. These plots were integrated to determine the values of section normal-force coefficient c_n and the pitching-moment coefficient about the quarter-chord point $c_{m_{c/4}}$. Values of the pressure coefficient P at the several points of measurement along the test section were then plotted against c_n. Because of the variation of P with Mach number at fixed values of c_n, the data were divided into five groups, each group including values lying within a limited range of Mach numbers. Typical plots of this character for lower orifice 5 and upper orifice 10 are shown in figure 15, the circles representing points lying between M values of 0.25 and 0.34, the inverted triangle values of 0.35 to 0.44, etc., as indicated on the figures.

From these plots mean curves of P against c_n for the various Mach numbers were established and, finally, mean pressure-distribution curves were derived at different values of M for each of several values of c_n. These derived plots are presented in figure 16. The values of pitching-moment coefficient noted thereon were obtained by integration of the mean curves with respect to the aerodynamic center for low-speed flow. This aerodynamic center was established in the usual way from values of $c_{m_{c/4}}$ and c_n obtained from the original pressure plots. The data for the lower speed conditions are shown in figure 17.

It will have been noted on figure 15(b) that, within the range of Mach numbers between 0.65 and 0.74, the pressures at orifice 10 on the upper surface (which was located near the station of maximum thickness) show extreme variations. This phenomenon also appeared at orifices 8 and 9. Because it was not practicable to subdivide the data into smaller groups than those given - that is, because no mean curves could have been established - the mean pressure plots for the highest range of M were made to show the pressure extremes within that range.

The pressure plots shown in figure 14 were obtained during the preliminary flights under level-flight conditions. The curves shown on these plots are theoretical curves based on the method of reference 1. The vertical lines at each pressure station represent a maximum possible (not probable) range of instrument error and will be discussed subsequently in the section on PRECISION.

Drag coefficient.- The drag results are shown in figure 18. These results were obtained by utilizing the proper longitudinal component of acceleration (corrected for the angle of instrument board) and the relation

$$C_D = \frac{W(1 - n)}{qS}$$

in which W is the weight, for this purpose taken as 5450 pounds; S the wing area of 209 square feet; and q the true dynamic pressure, that is, the measured value corrected for compressibility effect.

Elevator control force.- The increment of elevator control force has been plotted against acceleration increment for the several dives in figure 19. The significance of this plot is discussed subsequently in this report.

PRECISION

The degree of precision with which some of the pertinent quantities were measured can be given as a fixed number, whereas for others, generally those depending upon pressure measurements, no fixed value can be given. Those quantities belonging to the first group, for which estimates of accuracy may be immediately assigned, are

Normal acceleration, $\pm 0.1g$

Longitudinal acceleration, $\pm 0.02g$

Increments in elevator deflection, $\pm 0.25^\circ$

Absolute values of elevator deflection, $\pm 0.5^\circ$

Elevator and aileron forces, ± 4 pounds

The accuracy of the air density (and consequently of the true airspeed), air temperature, velocity of sound, and altitude depend upon the accuracy of the temperature-altitude survey made by the pilot during the climb. Even if somewhat larger errors than are thought to exist are assigned to the pilot's individual observations, the final errors in the density-altitude relations have been minimized by an integration process which gave, for each of the dives, curves of pressure and density, as measured from the initial altitude to the ground, against the absolute altitude. If a 2° error in temperature is allowed, the effect on the density ρ and on the velocity of sound a are small because these quantities are determined by the absolute temperature. It is estimated that ρ is known at any point in the dive to an accuracy of approximately 1 percent; whereas the error in the velocity of sound at any point is negligible since it depends on the square root of the absolute temperature. Thus, the accuracy with which the Mach number M is given depends almost entirely upon the accuracy with which the true airspeed is known.

When the various errors that enter into the determination of the true airspeed given by the swiveling-head installation are considered, it is estimated that, at speeds in excess of 350 miles per hour, the error is less than $1\frac{1}{2}$ percent; whereas at the lower speeds (100 to 150 miles per hour) the error may be as much as $\pm 2\frac{1}{2}$ percent. Although it was thought when the tests were devised that the accuracy of the airspeed measurements given by the pressure plate might be better than that of the swiveling head, recent tests by Walchner (reference 2) have indicated that the usual Prandtl type head has a critical velocity well above that reached in the dives. This fact, together with the fact that short balanced pressure lines were used, probably gives a greater accuracy to the swiveling-head airspeed measurements for the reason that reliable measurements could not be obtained with the pressure-plate head at speeds below 200 miles per hour and also because it was necessary to introduce slight corrections for the effect of the resultant acceleration parallel to the direction of the boom. When these errors in airspeed are considered, values of the Mach number at any point in the dive can be said to be accurate to within ± 0.01.

If it can be assumed that static pressure exists at the static openings of the swiveling head, the error in the altitude at any instant is no more than ± 300 feet.

The absolute accuracy of the individual pressure measurements varied according to the position of the orifice along the airfoil. In general, the better absolute accuracy was obtained with the more sensitive manometer cells, which were located toward the rear of the chord, although on a percentage basis the accuracy in this region is poor because of the relatively low pressures. In addition, because the sensitivity of the manometer cells was necessarily based upon the expected pressures to be measured at the highest speed, the maximum possible errors at the low speeds as indicated by figure 14 are relatively high. Figure 16 shows, however, that the probable error, as indicated by the lengths of the vertical lines, decreases with an increase in Mach number. The magnitude of the absolute probable error is obtained from these figures by multiplying the length of the lines by the value of q that applies for the Mach number in question.

Small accidental errors in the pressure distribution caused by fairing and other factors result in only relatively small changes in the integrated normal-force coefficient but may cause larger changes in the pitching-moment coefficient. In fact, most of the point scatter (±0.006) shown in the section c_n and c_m relations (fig. 17) can be attributed to these inaccuracies.

Consideration of the effect of the pressure errors on the determination of the critical speed indicates that the critical Mach number is accurate to about ±0.01.

The accuracy of the total drag measurements (fig. 18) depends upon the accuracy of the longitudinal acceleration and airspeed measurements. Since the accelerometer reading, $1 - n$, used in the relation for C_D is a small quantity at the lower speeds early in the dives, the percent error is high at the beginning of the dives but rapidly decreases as the terminal velocity is approached. At the highest Mach numbers the error in q becomes of greater importance and the minimum error, which occurs at the high Mach numbers, is about ±5 percent. The accuracy of the drag measurements is sufficient, however, to show the trend of C_D with increasing values of M.

DISCUSSION

The data presented make it evident that the test conditions extended well beyond the critical speed of the test section of the wing; it may be inferred from this evidence that many other exposed parts, such as wing sections near the root, propeller-blade roots, cowling, and crown of the cockpit enclosure were even more advanced beyond their own critical speeds. The first question of interest, therefore, is whether the presumable existence of a compression shock or shocks was manifested by any unusual behavior of the airplane.

Indications of compression shock.- Evidences of the behavior of the airplane and of the compression shock were obtained from the vibrograms, the wing photographs, the pressure records, and the records from the total-head tubes. The vibrograms are not given because they are difficult to describe and, as they showed an absence of serious vibration, it was considered inadvisable to introduce them. The records showed that the vibration existing was irregular and that the amplitudes did not exceed about one-sixteenth inch. There was no evidence that the existing vibration could be attributed in any way to a compression shock; in fact, the vibration records showed the same character well below and above the critical speed.

The wing photographs likewise showed that any vibration existing must have been small, although the method used would not allow evaluation of amplitudes as small as those indicated by the vibrograms. The indications of the vibrograms and photographs were substantiated by the pilot, who detected no vibration.

A brief study of the actual pressure records indicates that zones of steep pressure gradients did not oscillate or shift back and forth; that is, the few shifts observed were unidirectional. For example, after the acceleration peak in the pull-out from the fastest dive, the pressure ratio at orifice 9 on the upper surface changed in the following manner:

Time (sec)	M	c_n	P
36.5	0.695	0.25	1.4 (approx.)
37.0	.690	.23	.75

As the other pressures showed no great change, however, the zone of steep pressure gradient in the region of orifices 8 to 10 shifted forward rather suddenly a short distance with slight reduction of M and c_n. With further reduction of these quantities during the later stages of the pull-out, there was no return to higher pressure at orifice 9. The sudden forward shifts observed in the pressure records did not manifest themselves in the acceleration records or other observations and were therefore of little consequence under the conditions of these particular tests.

The total-pressure records indicate that at subcritical speeds (300 mph dive, fig. 8) there was no appreciable loss of pressure even up to quite high values of lift. In the supercritical region, however (figs. 9 and 12), the pressure losses appear to vary primarily with the acceleration or lift. The records displayed no sudden changes other than those associated with changes in angle of attack arising from movement of the elevator.

From the foregoing discussion it seems that, although the wings and other parts of the airplane were well into the supercritical region of flow, no serious consequences of the compression shock were manifest. The possibility of serious consequences under other circumstances is not, however, precluded by this result, particularly in view of the forward shifts of the steep gradient noted. These sudden shifts, although slight and unidirectional in these tests, are nevertheless an indication of a critical combination of Mach number and lift coefficient for the particular section and structure tested, and it therefore appears that conditions may be encountered in which oscillating behavior of the zone of steep pressure gradient is a possibility.

Further consideration of this problem on the basis of the possible physical nature of the flow beyond the critical speed has led to the belief that the return to high pressure at the rear of the supersonic region of flow is accomplished in part through convergence of the stream tubes, somewhat as shown in figure 20, and that this convergence is accompanied by separation at some point forward of the rear critical point. The existence of separation would account for many phenomena observed in the wind-tunnel as well as in the flight tests. If convergence and separation actually do influence the return to subsonic velocities, the occurrence of an instability can easily be visualized because on the one hand, the supersonic velocities would tend to increase toward the rear by virtue of the expansion forced by the airfoil boundary and, on the other hand, separation would tend to cause convergence and reduction of velocity at some forward point. The position of the separation point would therefore be rather indeterminate and would depend on a number of factors including Mach number, angle of attack, profile shape, and surface roughness or irregularities. In these tests the effect of the considerable surface roughness may very likely have been to force separation within a limited region, thus contributing to a stable condition.

General features of airfoil characteristics and related questions.- Reference to the pressure distributions given in figures 14 and 16 indicates that the pressure ratios increase with Mach number as would be expected from wind-tunnel data (reference 3). Although no quantitative study of these ratios has been made, it is apparent that beyond the critical speed the pressure ratios are very great even at low values of c_n. This result is in qualitative agreement with the results of the wind-tunnel tests.

The shapes of the diagrams, however, are not entirely what might have been expected from the wind-tunnel results, in that the curves are rather irregular and the pronounced pressure rise at the rear limit of the supersonic region is never very far back on the section. The irregularity of the curves is due principally to the existence of irregularities in the surface of the wing. A study of these irregularities has indicated definite coincidence between sudden changes in surface or profile of the wing and sudden changes in slope of the pressure diagrams. The irregularity of the diagrams, therefore, has no fundamental significance, although the causative irregularity of the wing surface probably has considerable influence on the main features as well as on the detailed features of the pressure diagrams.

The relatively far forward average position of the zone of pronounced pressure rise is of considerable importance, insofar as it has a distinct bearing on the moment coefficient and on the areas over which the high pressures act. Wind-tunnel results are positive in their indications of a pronounced increase in pitching-moment coefficient with increasing Mach number above the critical speed. Slightly cambered airfoils as well as highly cambered airfoils show this increase. The NACA 2209-34 airfoil, which, of the series of airfoils reported in reference 4, has a mean camber line most nearly like that of the 23014 section of the tests reported herein, shows increments of moment coefficient as high as 0.04. The pitching-moment coefficients from the tests of this report (summarized in table I) do not show such an increase; in fact, they show no trend at all with Mach number, the values scattering slightly about a mean value of 0.025.

The qualitative disagreement between the flight and wind-tunnel tests regarding the pitching-moment coefficient and the position of the zone of steep pressure gradient is believed to be a result of the surface roughness and irregularities of the full-scale wing, which caused flow separation to occur at a forward point in conformity with ideas previously expressed as to the nature of the flow. High moment coefficients might, therefore, still be expected on smooth wings in accordance with wind-tunnel indications.

Although the profile drag of the wing section itself was not measured, the approximate drag of the entire airplane was determined at the higher Mach numbers as previously described. The results shown in figure 18 are clear evidence of the existence of the rapidly increasing drag coefficients that have been indicated by wind-tunnel tests.

The slope of the wing lift curve was not measured because of the great difficulties that would have been involved in measuring angle of attack. Evidence of the qualitative behavior of the slope was obtained, however, from the measurements of elevator control force in the pull-outs. It can be shown that, for a fairly stable airplane having a constant slope of lift curve, the ratio of control-force increment to normal acceleration increment is constant, regardless of speed, provided the stick force is not applied impulsively. In other words, the acceleration imposed simply corresponds to the angle of trim for the elevator setting used to pull out. Since the lift on the wing and the hinge moment of the elevator both vary as V^2, and since the change in trim is a linear function of elevator setting, the ratio of acceleration increment to control-force increment remains constant no matter what the speed or acceleration.

The principle stated in the preceding paragraph can be used in conjunction with the stick-force and acceleration measurements to check the actual behavior of the lift-curve slope, inasmuch as the airplane conformed to the re-

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quirement of stability and the stick forces were rather slowly applied. The essential data are shown in figure 19, in which increments of acceleration are plotted against increments of control force for the several pull-outs. Each pair of values for one pull-out establishes a point on a straight line through the origin, which line represents the relation appropriate to the lift slope corresponding to the single value of the Mach number. The shaded wedge indicates the spread or scatter for four pull-outs, each having a Mach number of about 0.42 and it therefore furnishes a means of appraising the reliability of the results as an indication of the behavior of the lift-curve slope with Mach number.

From the foregoing remarks it will be appreciated that, if the slope of the lift curve changes much with the Mach number, the slope of the lines of Δg against ΔF_s will increase with increase of lift-curve slope and vice versa because, in order to effect a given change in lift with a high lift slope, the elevator deflection will not need to be large to cause the required change in trim and the hinge moments and control forces will remain relatively small. Examination of figure 19 indicates that the scatter of the data for a number of pull-outs at nearly the same speed is small compared to the spread of the results over the whole range of speeds. The speed or Mach number must, therefore, have something to do with the results. On the basis of this premise, steep slopes of Δg against ΔF_s corresponding to steep lift-curve slopes would be expected and vice versa. The data actually show that the slope of the curve of Δg against ΔF_s first increases as the Mach number increases from 0.42 to 0.62, then rapidly decreases as the Mach number increases to 0.72, and finally increases rapidly again as the Mach number continues to increase to 0.74. This result is exactly similar to wind-tunnel results, which show (references 4 and 5) that the slope of the lift curve increases with Mach number in the subcritical range, decreases rapidly to low values beyond the critical speed, and finally increases sharply again.

<u>Aileron instability</u>.- Figure 10 indicates quite clearly that, 22 seconds from the start of the records, a severe flapping of the ailerons occurred and continued through nine low-period oscillations during the pull-out which, of course, was immediately made. From the pilot's subsequent statement it appeared that the ailerons, which had previously (at the lower speeds) been considered satis-

factory, suddenly became unstable or aerodynamically overbalanced, and the oscillations resulted from the pilot's effort to return the stick to neutral followed by a subsequent overbalance in the opposite direction, and so forth. Figure 13, which shows several groups of simultaneous photographs taken with the wing and cockpit cameras during the period of aileron flapping, bears out the pilot's statement quite well.

The airspeed curve could not be shown on figure 10 because of failure of the timing circuit on the airspeed recorder and some other instruments. However, an estimate of the true airspeed at the time of the aileron unbalance was easily made on the basis of data obtained in other dives, and the value is known to have been about 500 miles per hour. This value is substantially greater than any speed that had been previously reached during the tests.

A study of the results led to the belief that the trouble was probably caused by compressibility effects on the aileron noses which were aggravated by the slight deformation resulting from the installation of the balance weights. Figure 2 illustrates, in exaggerated form, the nature of the deformation. The bulge was not actually more than about one-sixteenth inch from the original surface and was rather smoothly faired. It seemed clear from pressure-distribution results obtained in wind-tunnel tests (reference 3) that the projecting or bulging lower surface of the aileron nose would have a fairly low critical speed and that, consequently, the pressure reduction on the nose would increase with speed at a far greater rate than the pressure reduction at any point to the rear of the hinge.

On this basis, the balance weights were reinstalled above the nose in such a way that the thickness of the bulge on the lower surface was reduced to the thickness of one layer of doped fabric. This change completely remedied the trouble. The experience therefore shows that qualitative knowledge of compressibility effects based on wind-tunnel data was the deciding factor in solving this particular problem; it also indicates the necessity for careful attention to the details of aerodynamic design and construction of control surfaces in the light of compressibility effects.

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CONCLUSION

It may be concluded from the results of these tests that the principal effects of compressibility indicated by wind-tunnel results are also manifested under flight conditions; that is, the large drag increase occurs in the supercritical region, the slope of the lift curve changes with Mach number in the same qualitative manner as in the tunnel tests, and extremely low pressures exist over the airfoil surface in the regions of supersonic velocity. No increase of wing moment coefficient with Mach number was observed, however, and the region of steep pressure gradient (compression shock) was never found to exist much to the rear of the station of maximum thickness; this discrepancy may possibly be explained on the basis of the effect of surface roughness on the location of the separation point and the result does not necessarily contradict the wind-tunnel results, which were obtained on smooth airfoils.

Although no serious vibration or other serious manifestations of the compression shock were observed in these tests, the position of the shock suddenly shifted slight amounts in some cases and the possibility of serious shifting or oscillation under other conditions is not precluded but rather indicated.

Experiences of the tests indicated that grave danger exists at high speeds if airplanes are longitudinally unstable and also that careful attention must be given to compressibility effects on aerodynamically balanced control surfaces to avoid sudden overbalancing. In fact, these tests were successfully completed only because of the high strength of the XF2A-2 airplane and because the magnitude of aileron unbalance that suddenly occurred at a speed of 500 miles per hour was, by pure chance, within the pilot's strength.

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National Advisory Committee for Aeronautics,
Langley Field, Va.

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2. Walchner, O.: The Effect of Compressibility on the Pressure Reading of a Prandtl Pitot Tube at Subsonic Flow Velocity. T.M. No. 917, NACA, 1939.

3. Stack, John, Lindsey, W. F., and Littell, Robert E.: The Compressibility Burble and the Effect of Compressibility on Pressures and Forces Acting on an Airfoil. Rep. No. 646, NACA, 1938.

4. Stack, John, and von Doenhoff, Albert E.: Tests of 16 Related Airfoils at High Speeds. Rep. No. 492, NACA, 1934.

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TABLE I

MOMENT COEFFICIENTS ABOUT LOW-SPEED AERODYNAMIC CENTER

c_n	$c_{m_{a.c.}}$		
	M = 0.5	M = 0.6	M = 0.7 (a)
0	-0.021	-0.030	-0.022 (-0.030)
.1	-.024	-.030	-.027 (-.034)
.2	-.024	-.028	-.019 (-.027)
.3	-.027	-.025	-.020 (-.028)
.4	-.025	-.022	-.025 (-.033)
.5	-.028	-.017	-.015 (-.026)
.6	-.025	------	--------------
.8	-.028	------	--------------
1.0	-.029	------	--------------

[a]Parenthetical values refer to pressure diagrams for maximum limit of pressures. (See fig. 16.)

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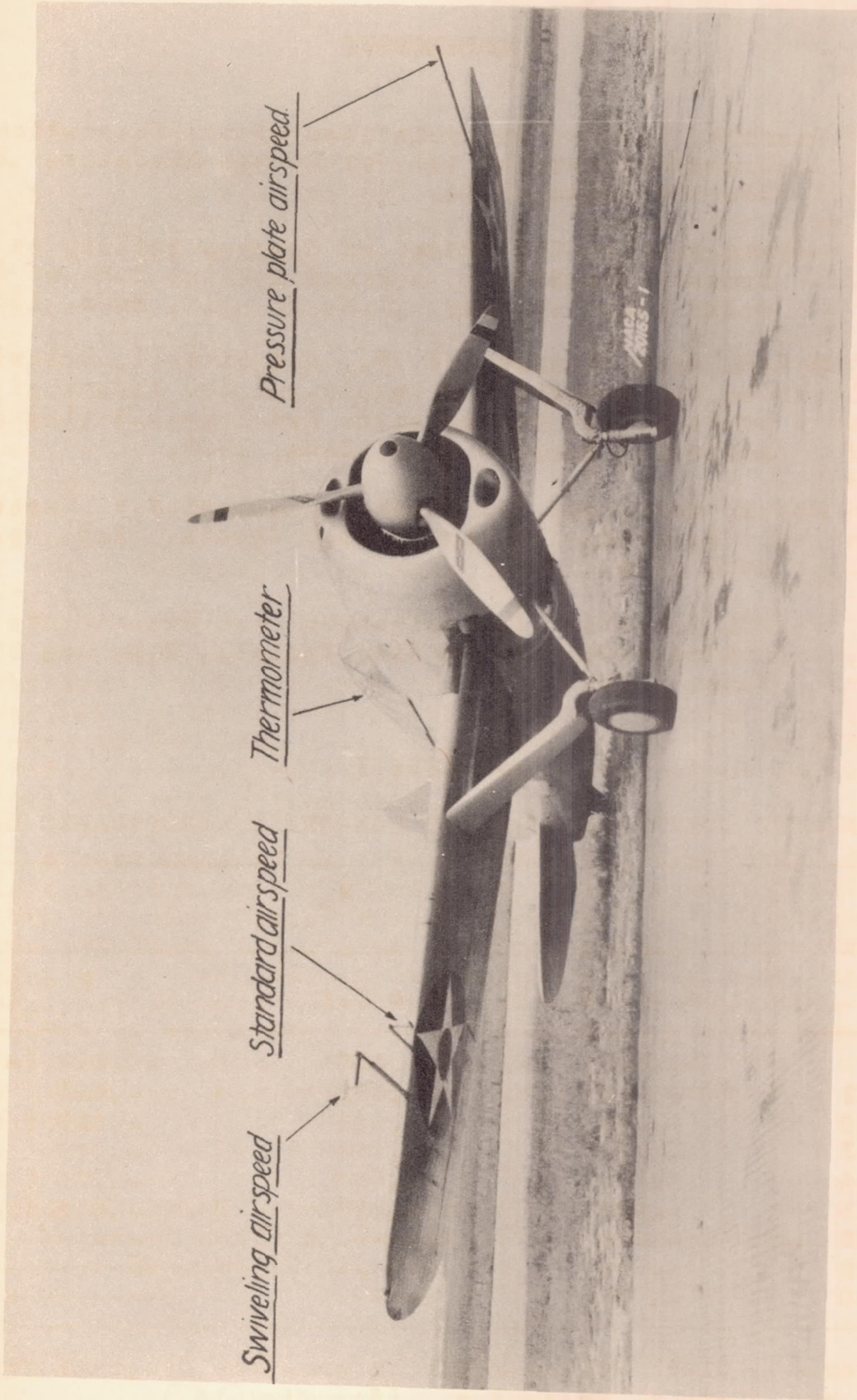


Figure 1.- Three-quarter front view of XF2A-2.

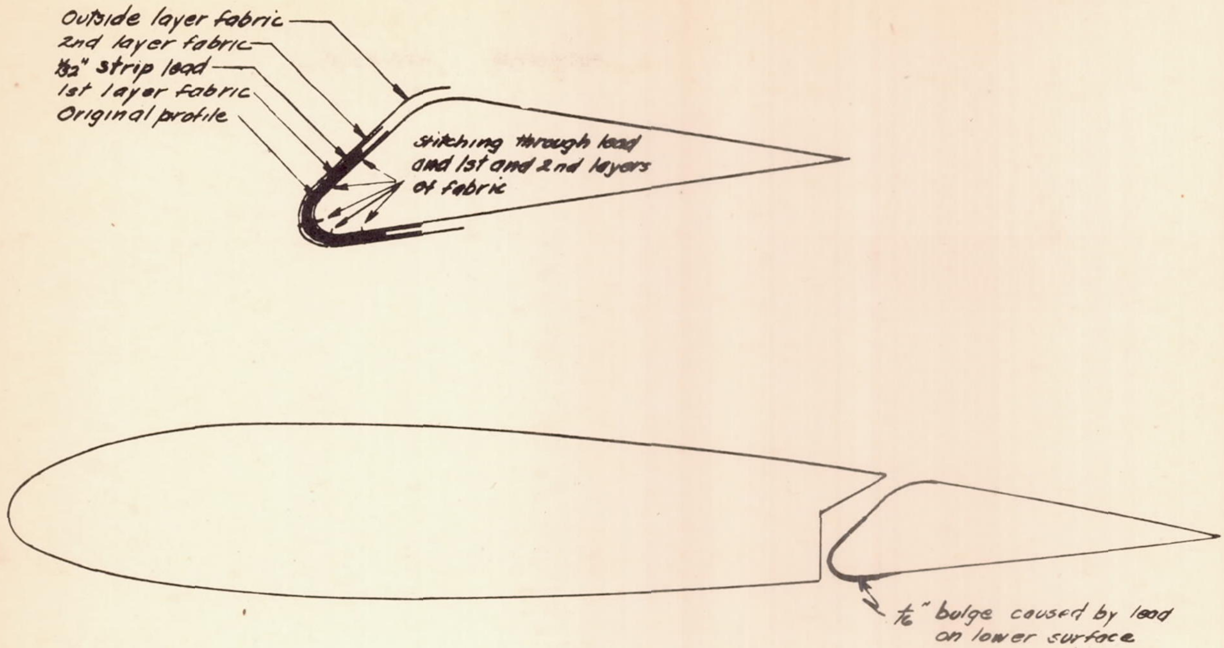
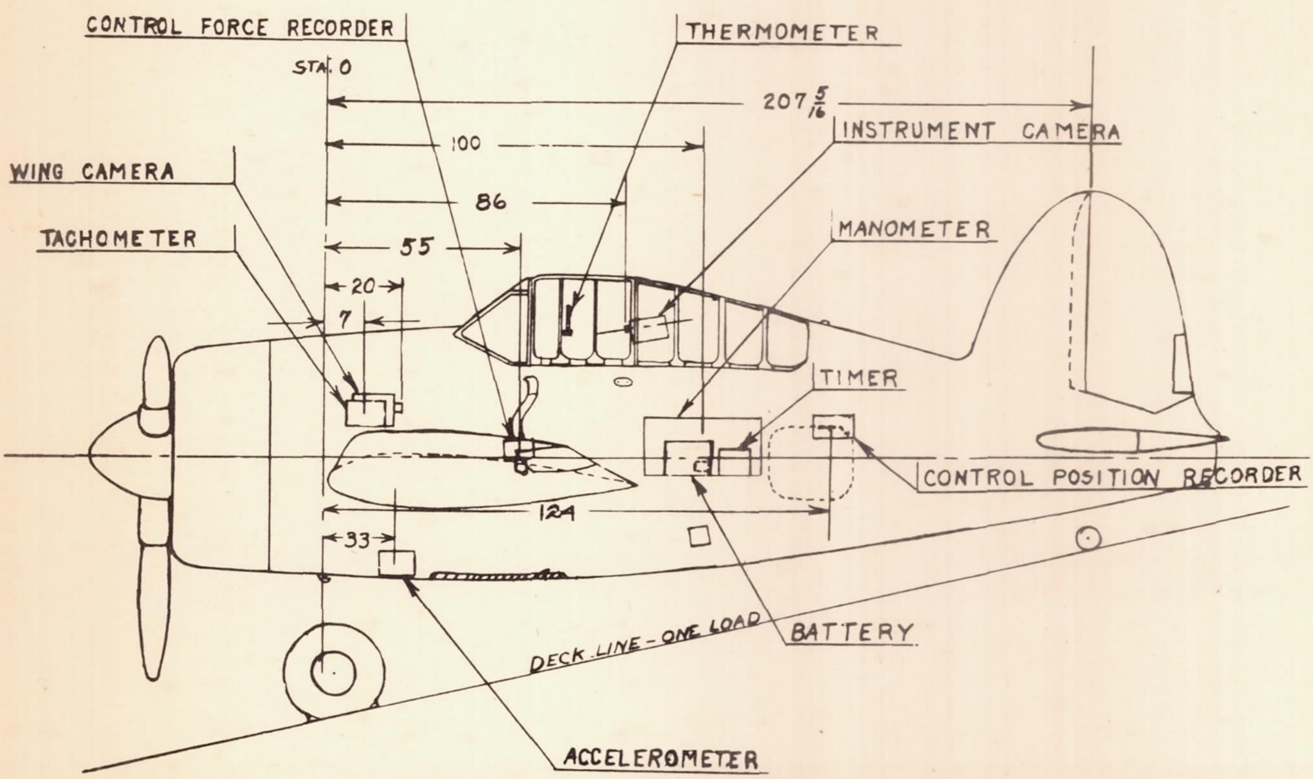


Figure 2.— Method of attaching lead to aileron leading edge.



(b) Side elevation.

Figure 3 .- Concluded. Instrument layout.

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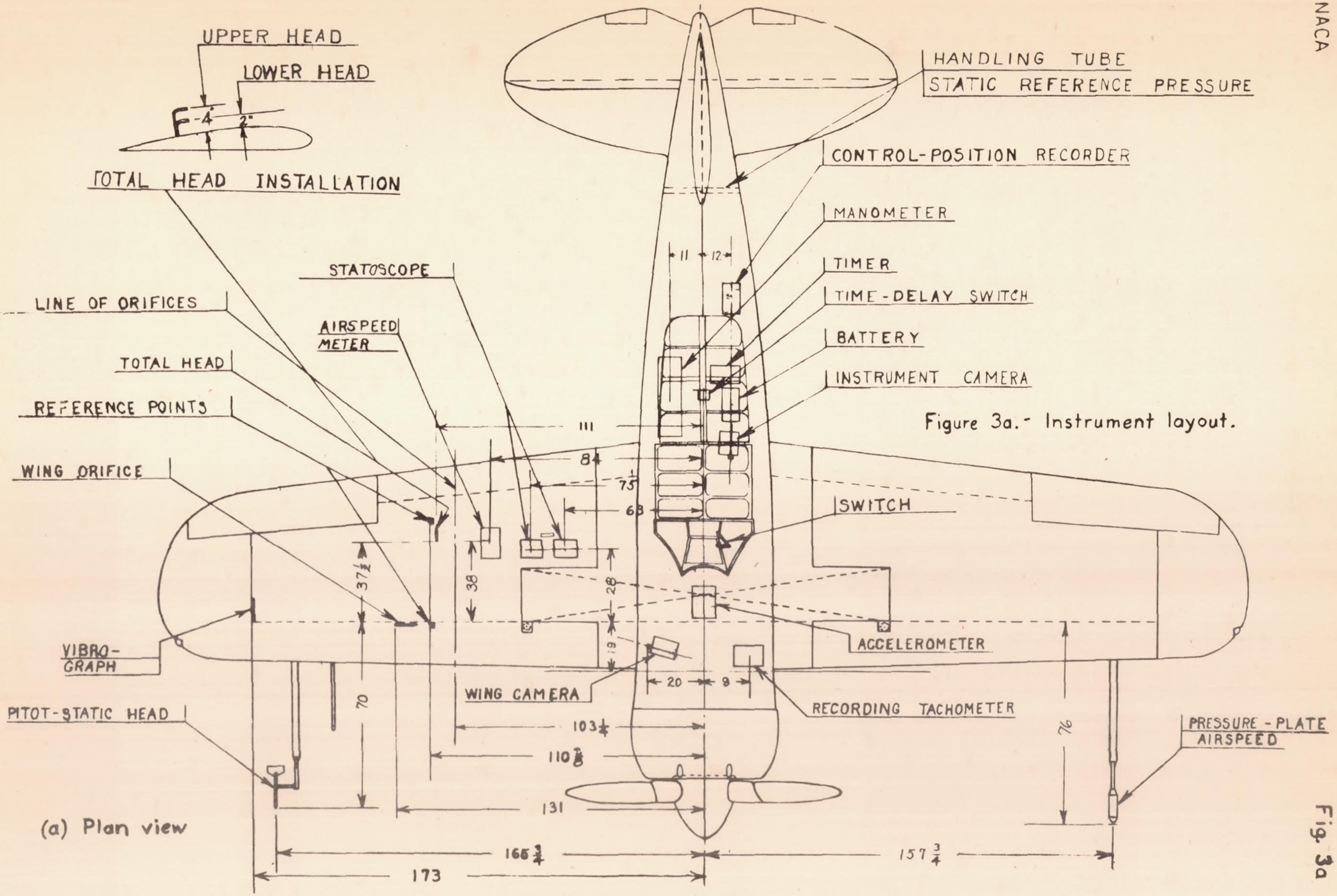


Figure 3a.- Instrument layout.

(a) Plan view

Fig. 3a

Orifice location in percent chord

Number	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
Upper	2.46	4.83	7.41	9.96	13.09	16.19	19.07	25.33	30.22	34.91	44.35	60.76	76.15	87.28	96.44
Lower	1.19	2.22	5.30	7.27	9.77	14.89	18.96	25.09	30.11	40.38	55.92	70.71	83.18	96.48	

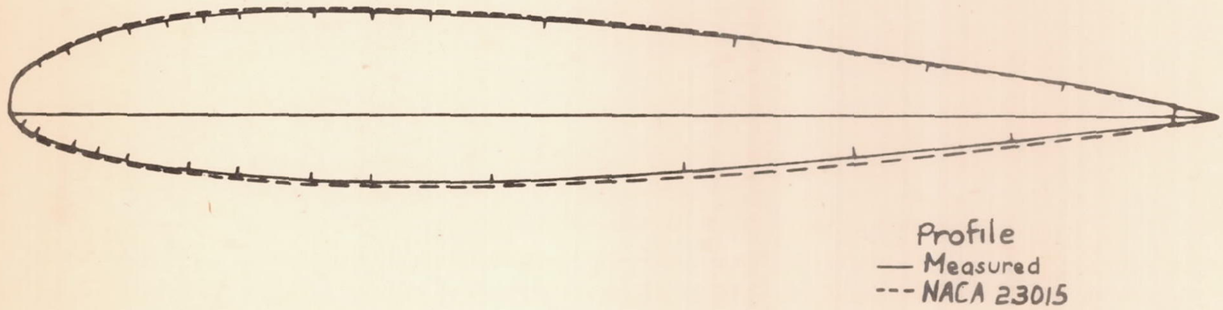


Figure 4. - Measured rib profile.

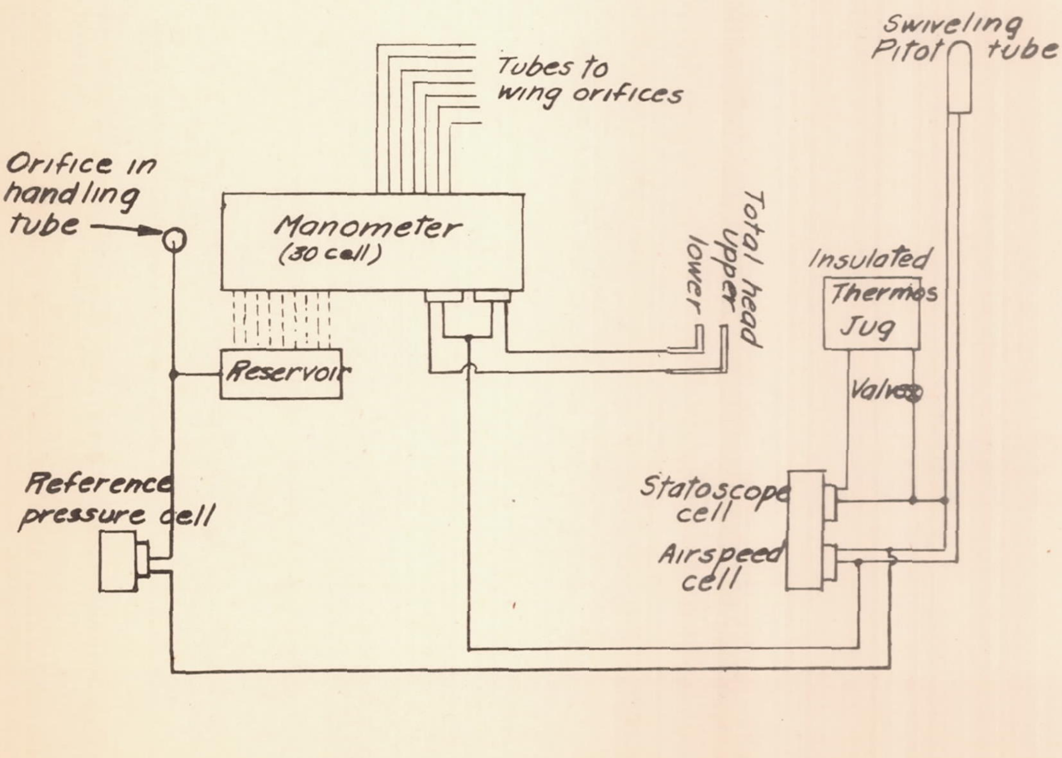
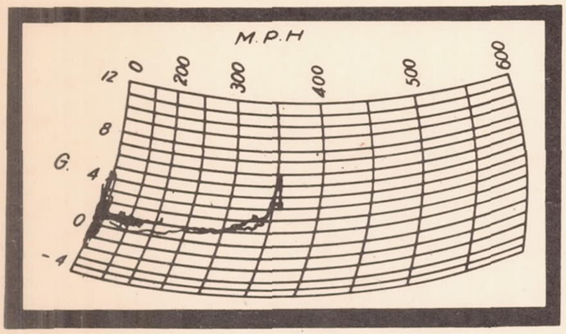
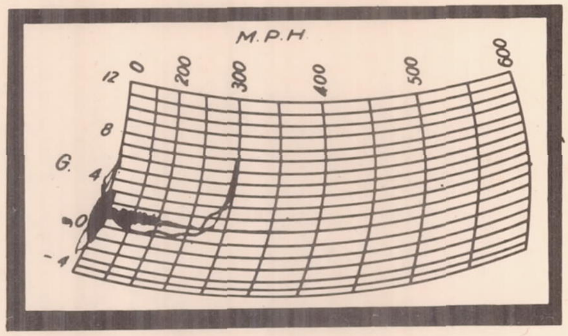


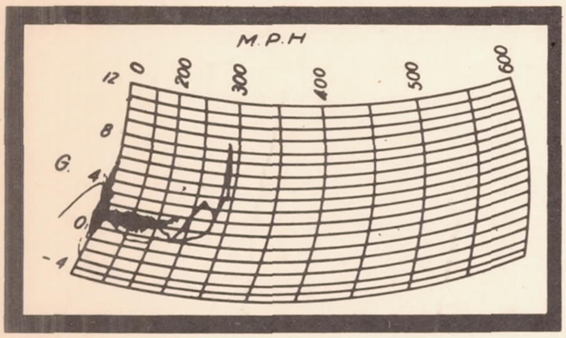
Figure 5.- Pressure - cell layout for XF2A-2 dive tests.



DIVE NO. 1
DATE: 4-23-40



DIVE NO. 2
DATE: 4-23-40



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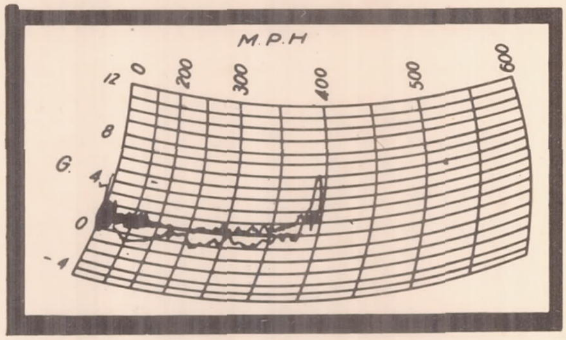

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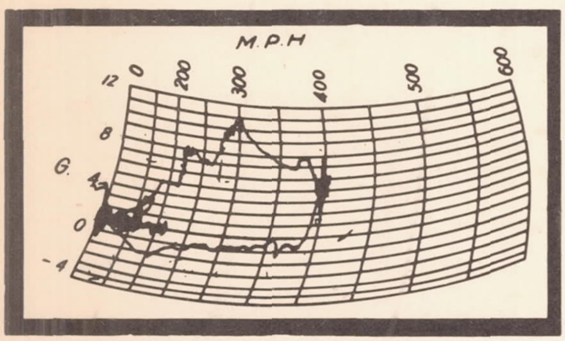

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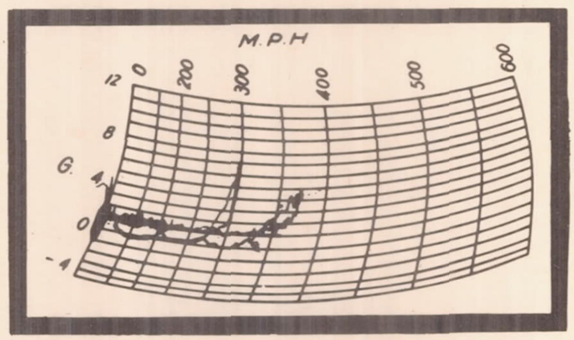
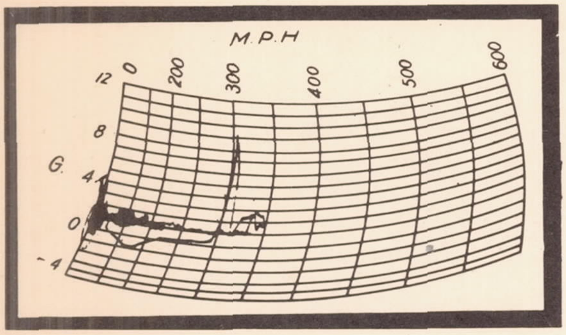
DATE: 4-25-40

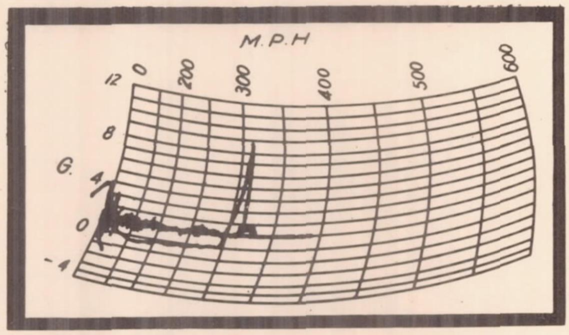


DIVE NO. 6
DATE: 4-27-40

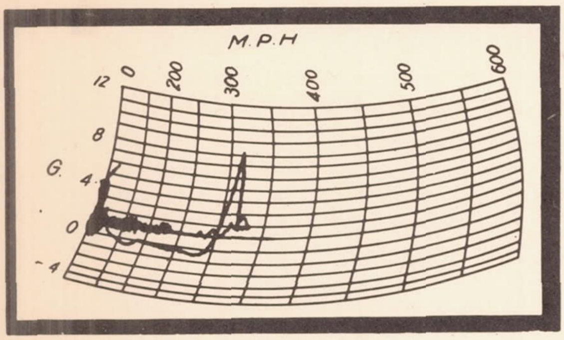
Figure 6.- V-G records of practice dives in XF2A-2.



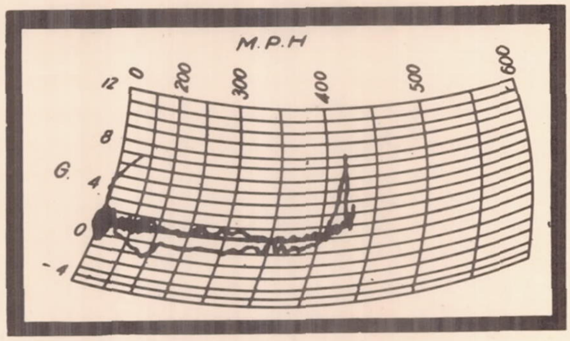
300 M. P. H.
DATE: 4-29-40



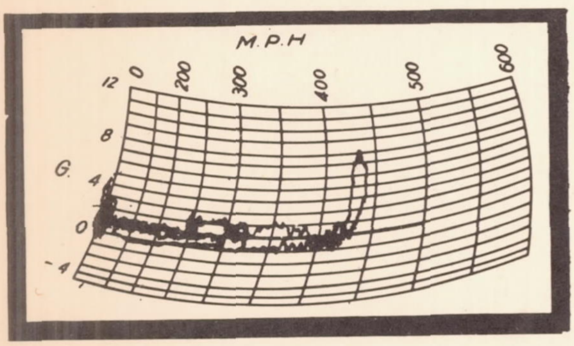
300 M.P.H. DIVE
DATE: 4-29-40



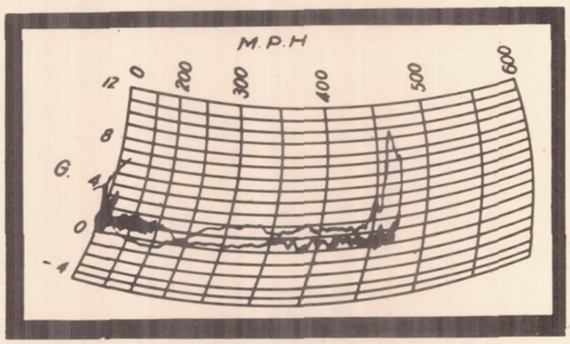
300 M.P.H. DIVE
DATE: 5-7-40



400 M. P. H.
DATE: 5-7-40



TERMINAL VELOCITY DIVE
DATE: 5-13-40



TERMINAL VELOCITY DIVE
DATE: 6-5-40

Figure 7.- V-G records of contract dives in XF2A-2.

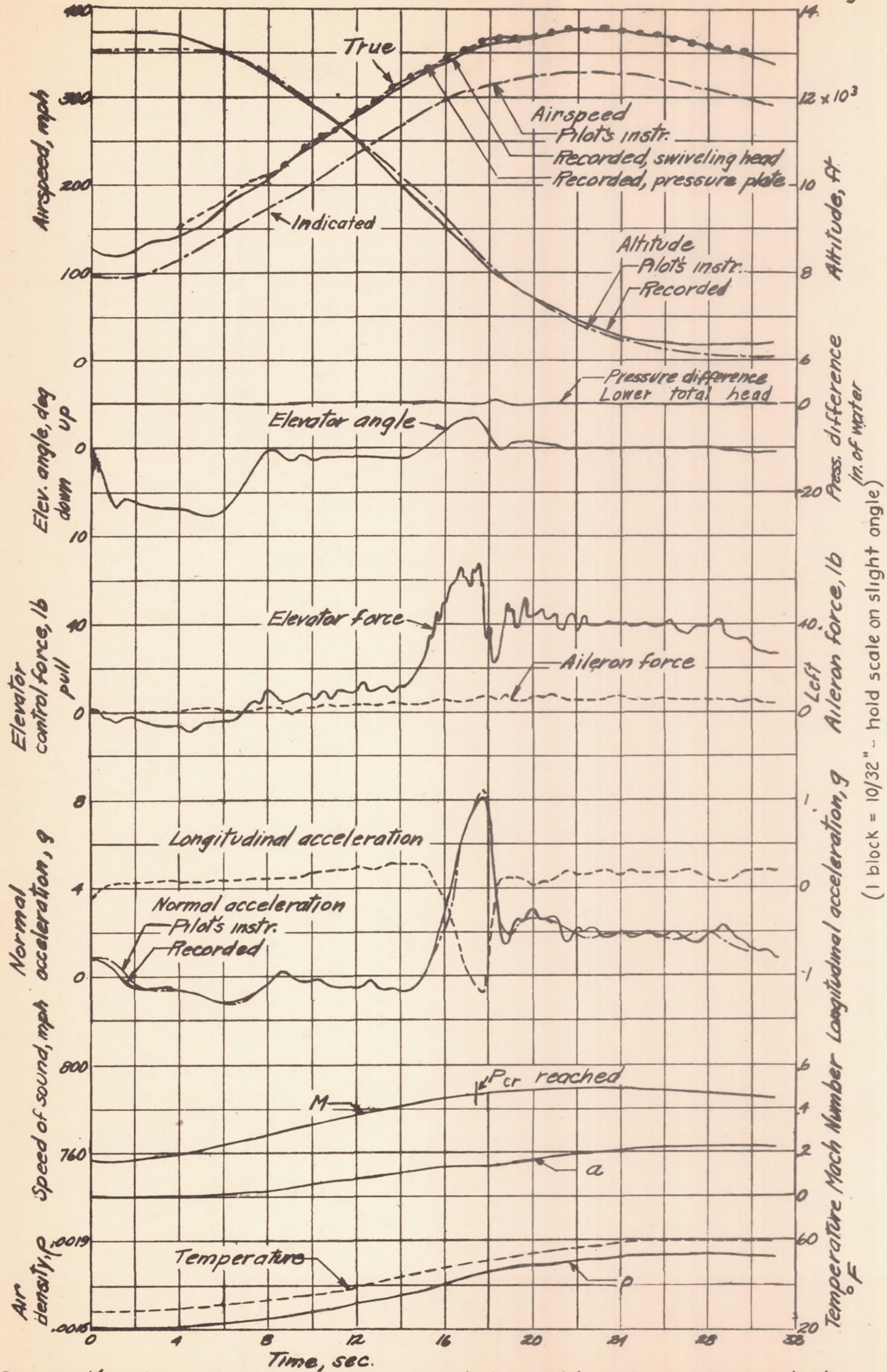


Figure 8.- Variation of measured and indicated quantities during 300 mph dive made on May 7, 1940.

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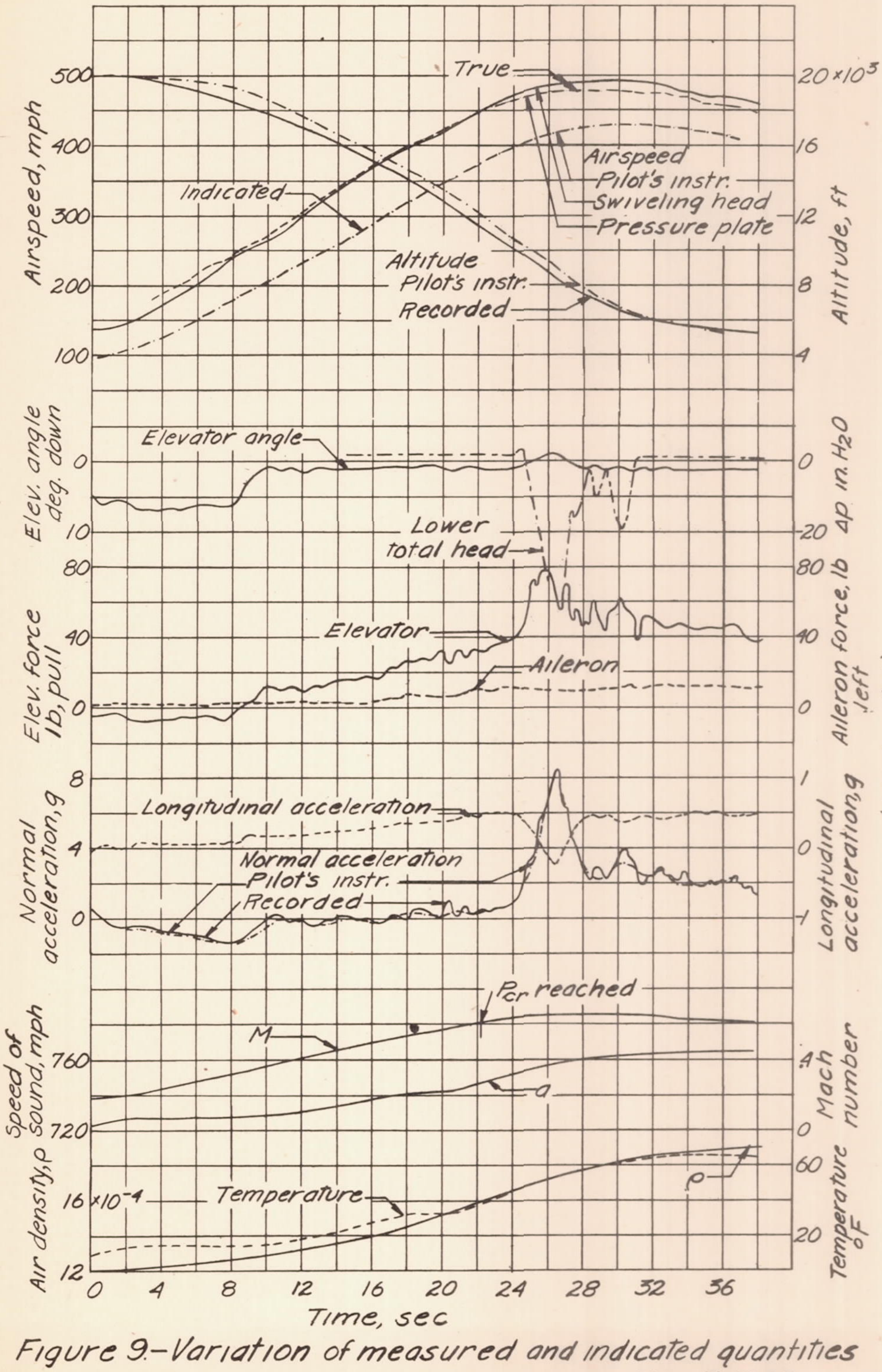


Figure 9.- Variation of measured and indicated quantities during 400 mph dive made on May 7, 1940.

L-417

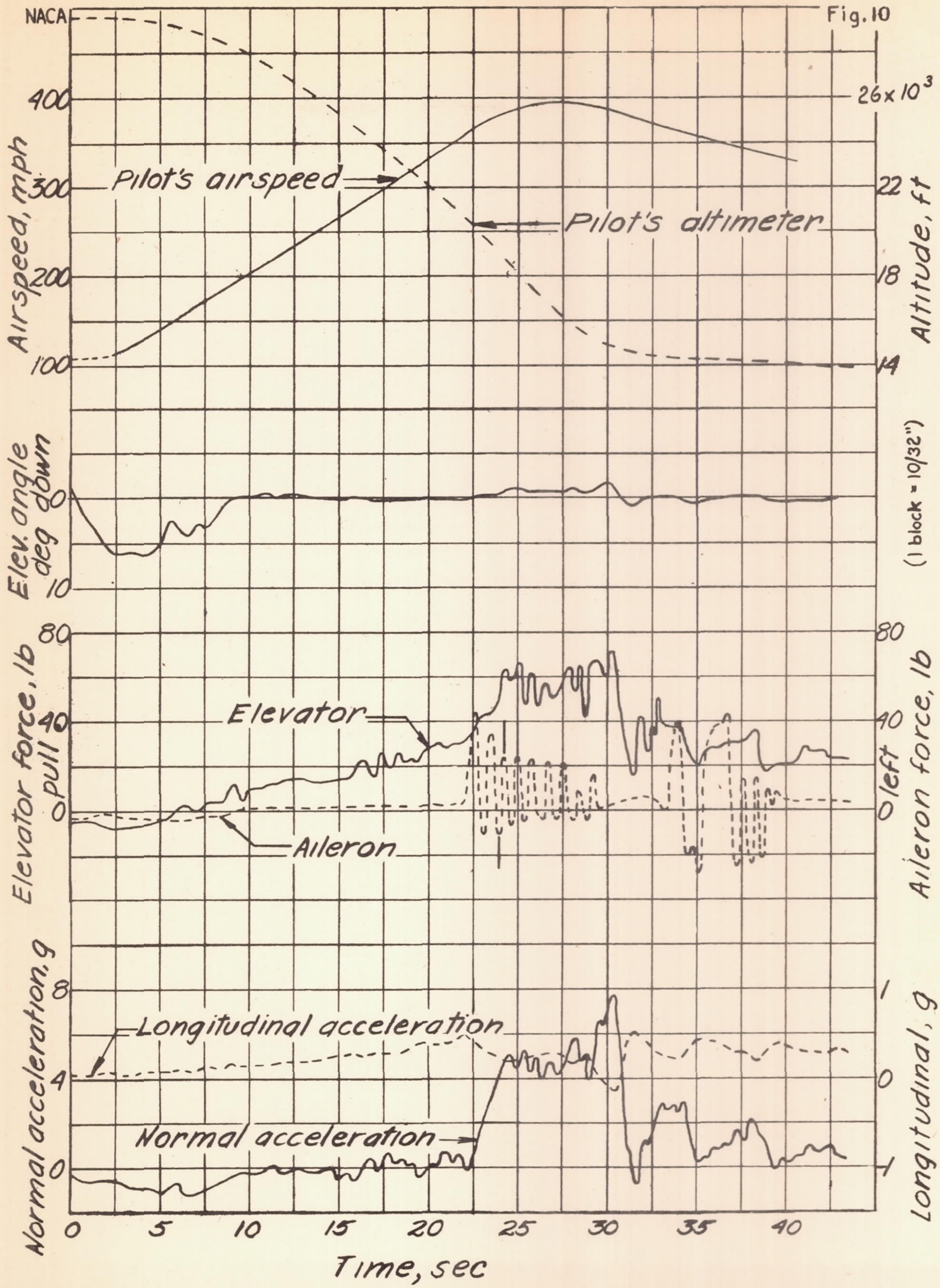


Figure 10 - Variation of measured and indicated quantities during attempted terminal dive.

L-471

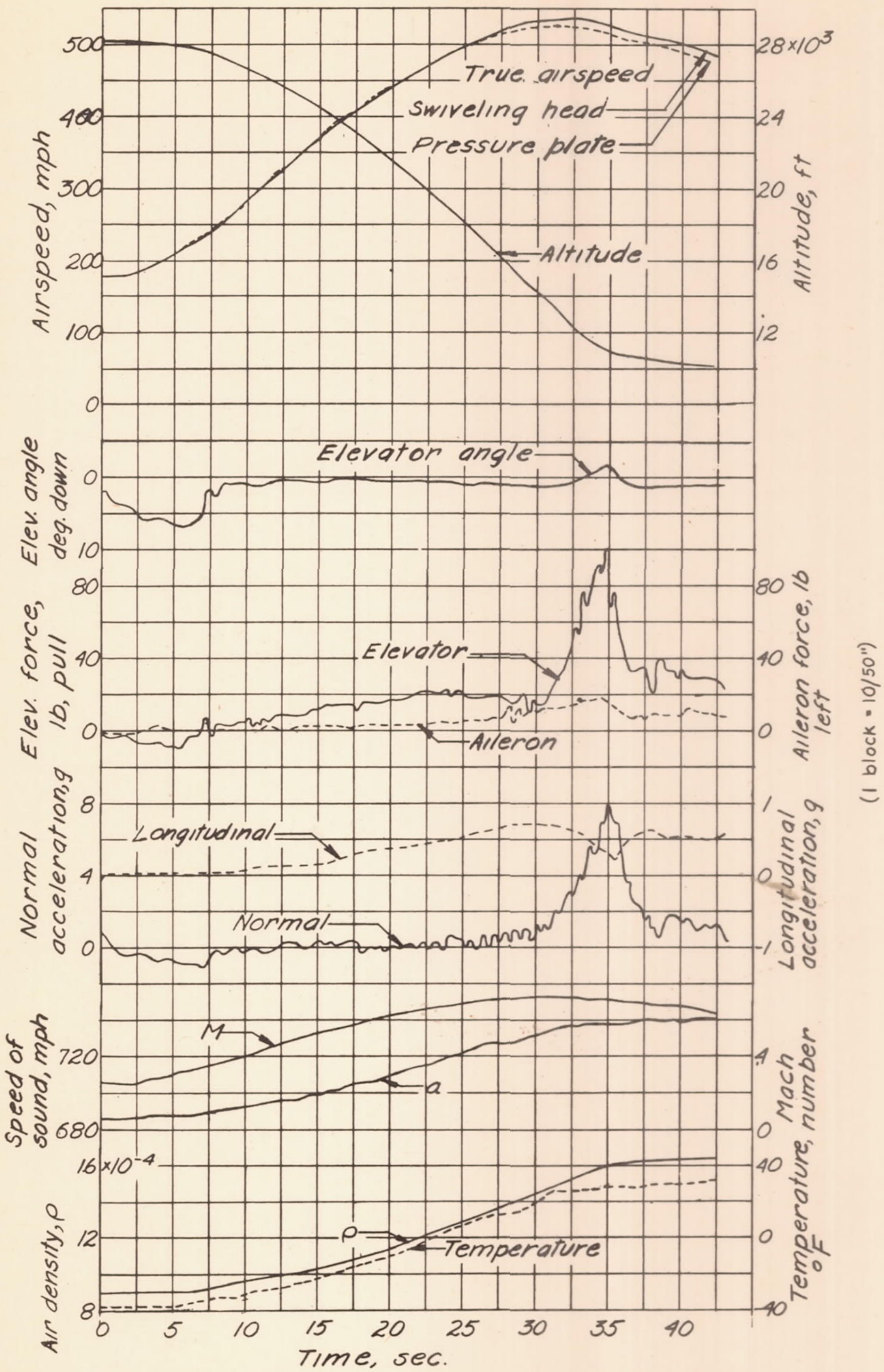


Figure 11- Variation of measured and indicated quantities during terminal dive made on May 13, 1940.

L-411

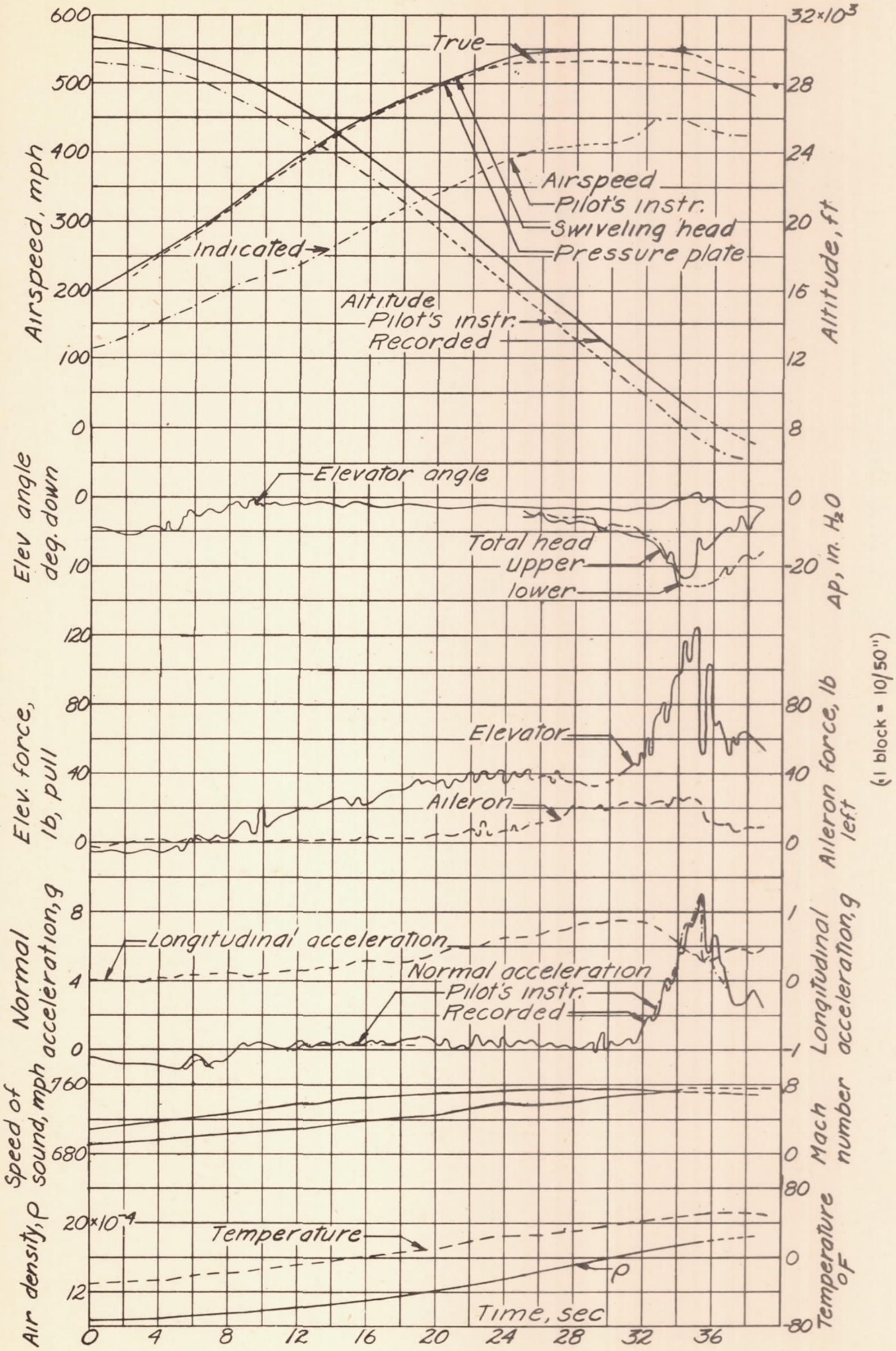
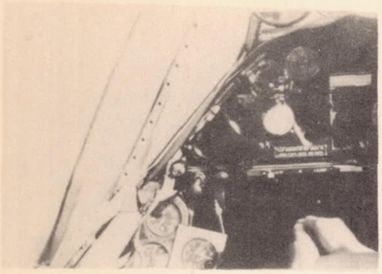
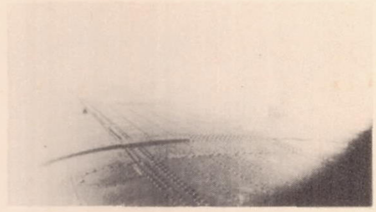


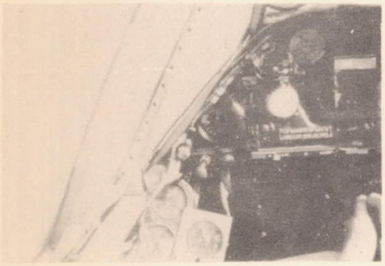
Figure 12.- Variation of measured and indicated quantities during terminal dive made on June 5, 1940.



(a)

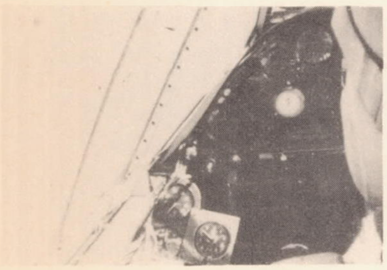
STICK LEFT
AILERON DOWN 5 DEG.
FORCE RIGHT = 10 LBS.
TIME = 23.8 SEC.





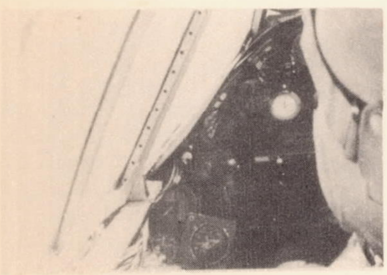
(b)

STICK MOVING RIGHT
AILERON MOVING UP



(c)

STICK NEUTRAL
AILERON NEUTRAL
FORCE LEFT = 4 LBS.
TIME = 24.0 SEC.



(d)

STICK MOVING RIGHT
AILERON MOVING UP

(e)

STICK RIGHT
AILERON UP 10 DEG.
FORCE LEFT = 24 LBS.
TIME = 24.2 SEC.

COCKPIT CAMERA RIGHT WING CAMERA

Figure 13.- Sequence of events showing aileron instability in attempted terminal velocity dive on 5-7-40.

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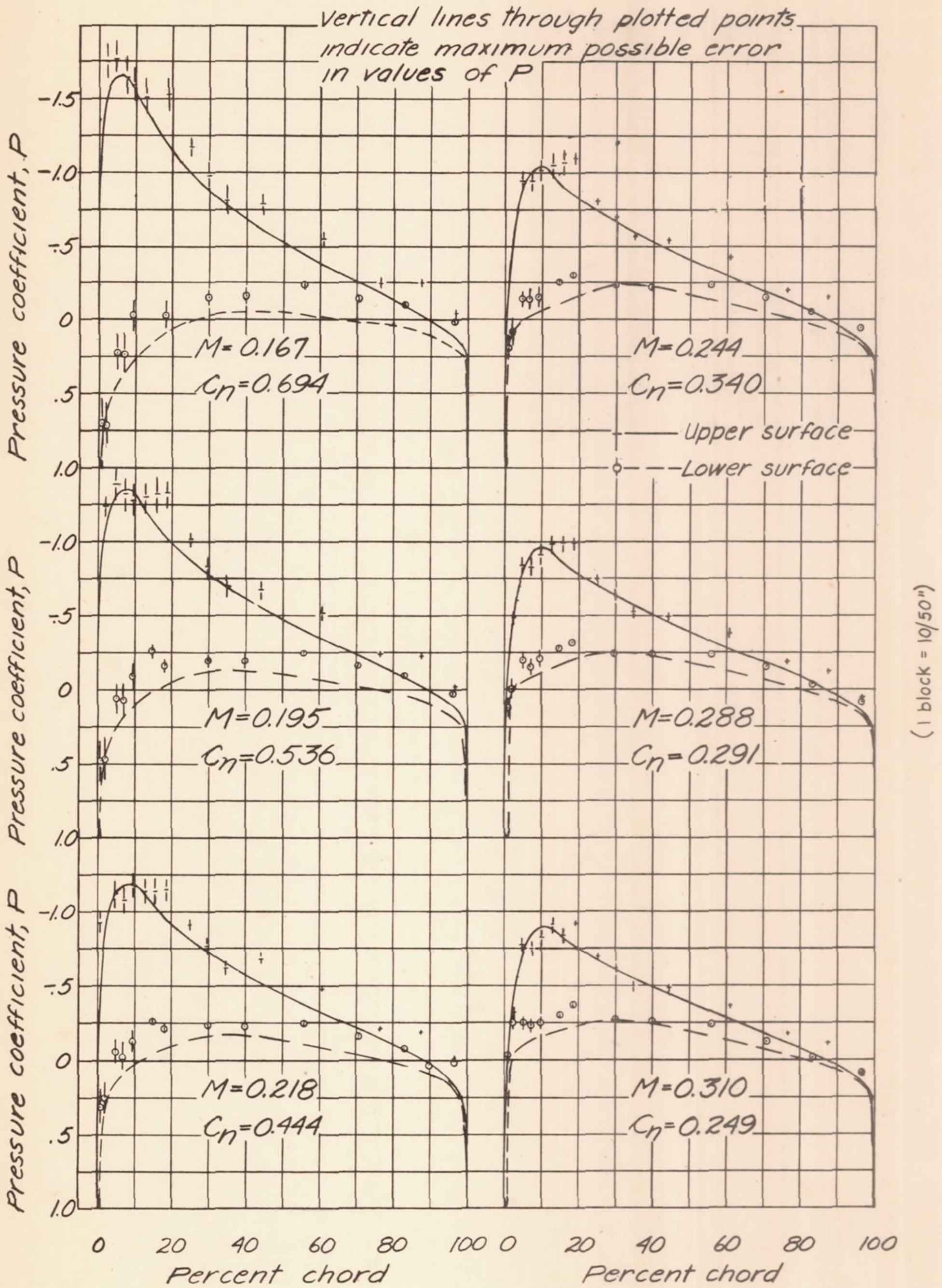

Figure 14.- Comparison between theoretical and actual rib pressure distributions measured in steady level flight.

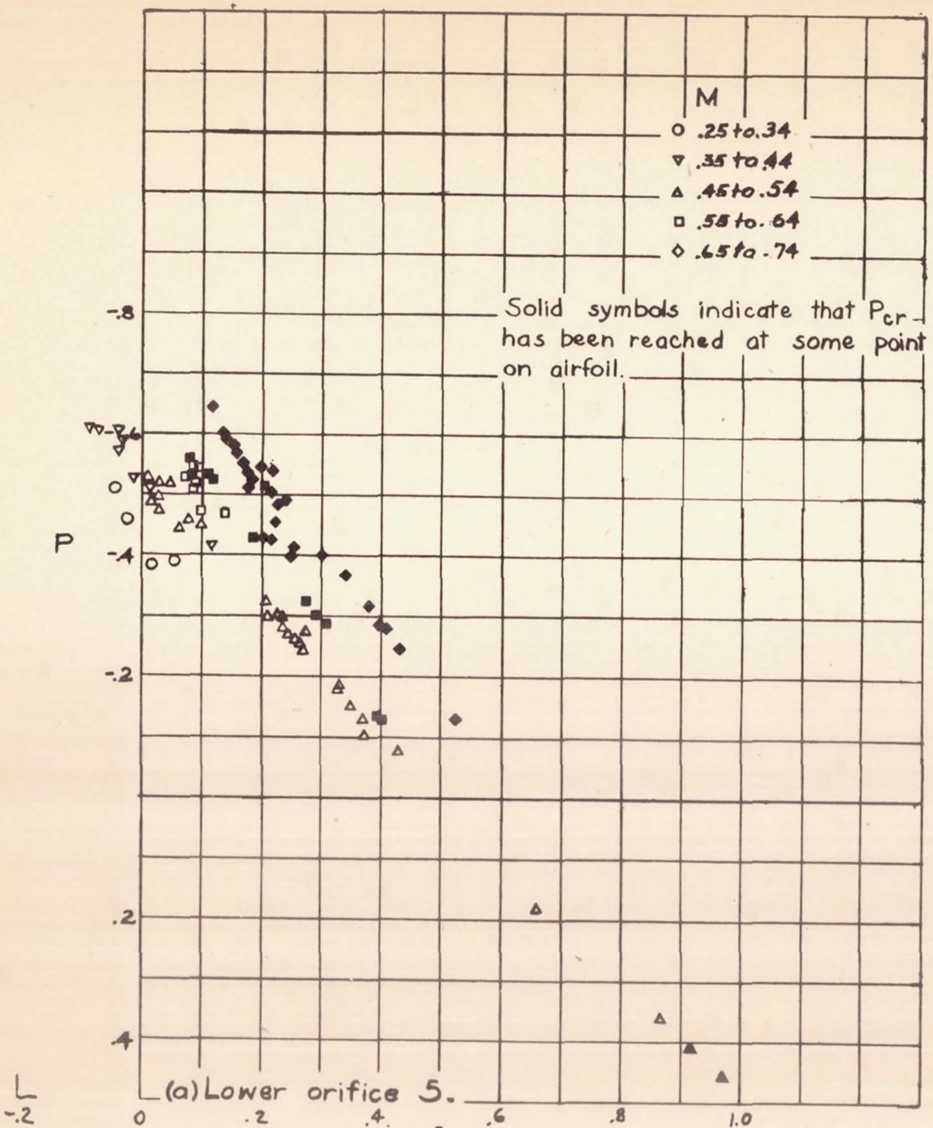

Figure 15.-Variation of local pressure coefficient with section normal-force coefficient.

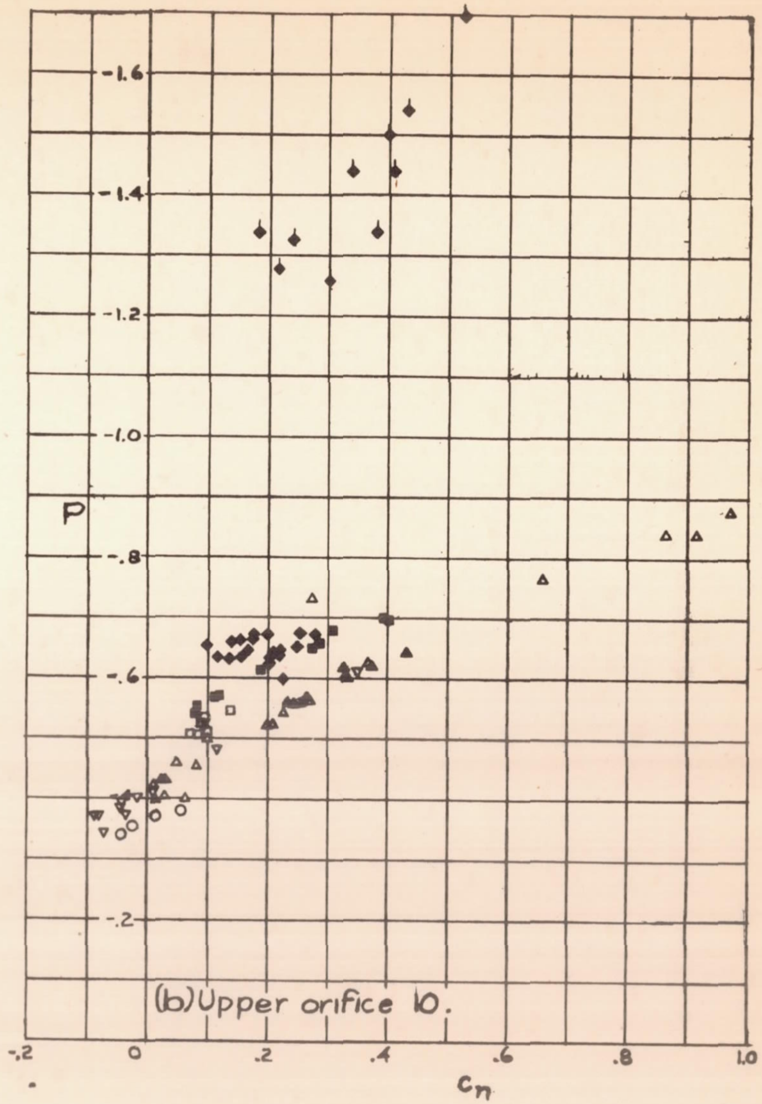

Figure 15.- Concluded
Variation of local pressure coefficient with section normal-force coefficient

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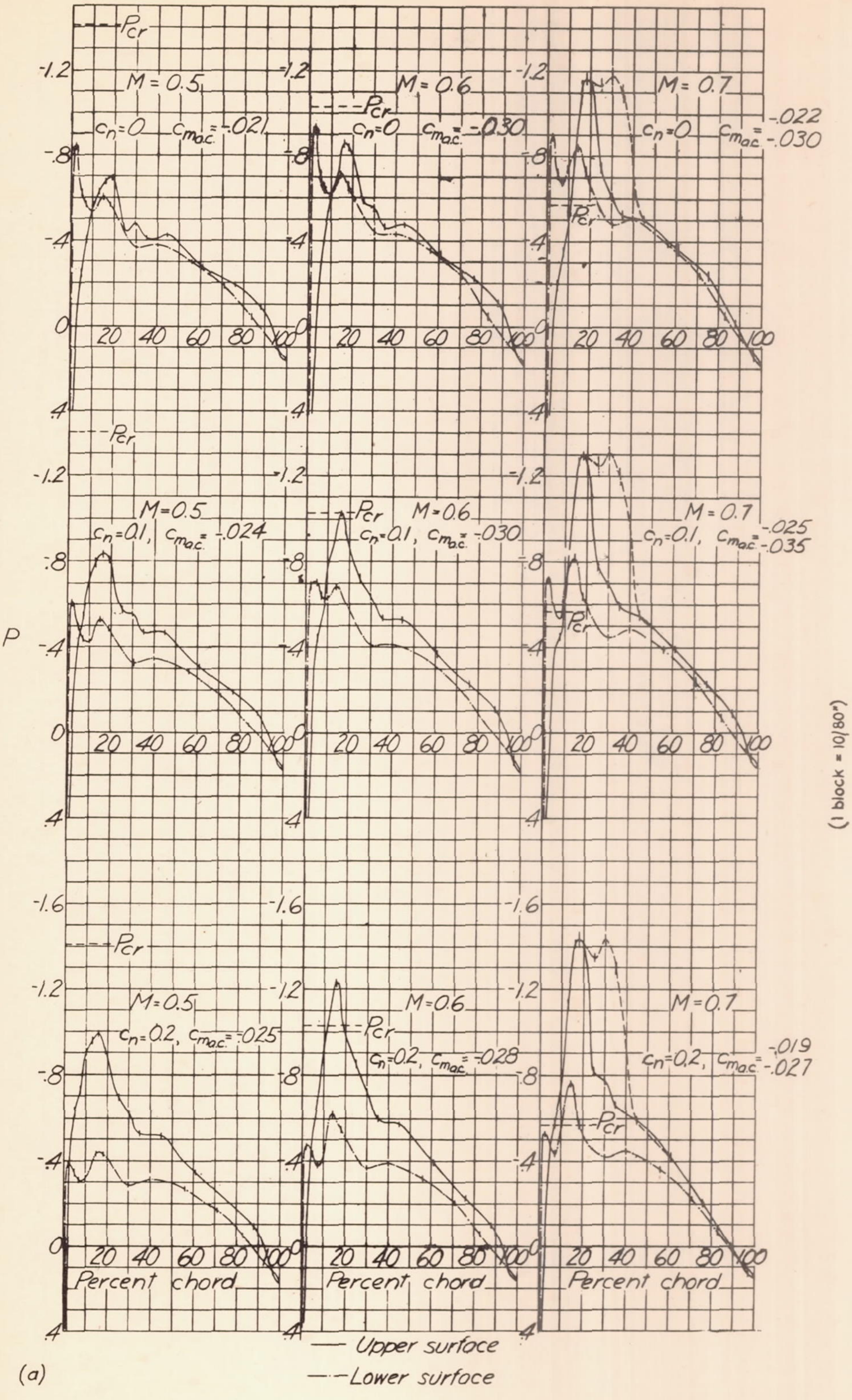
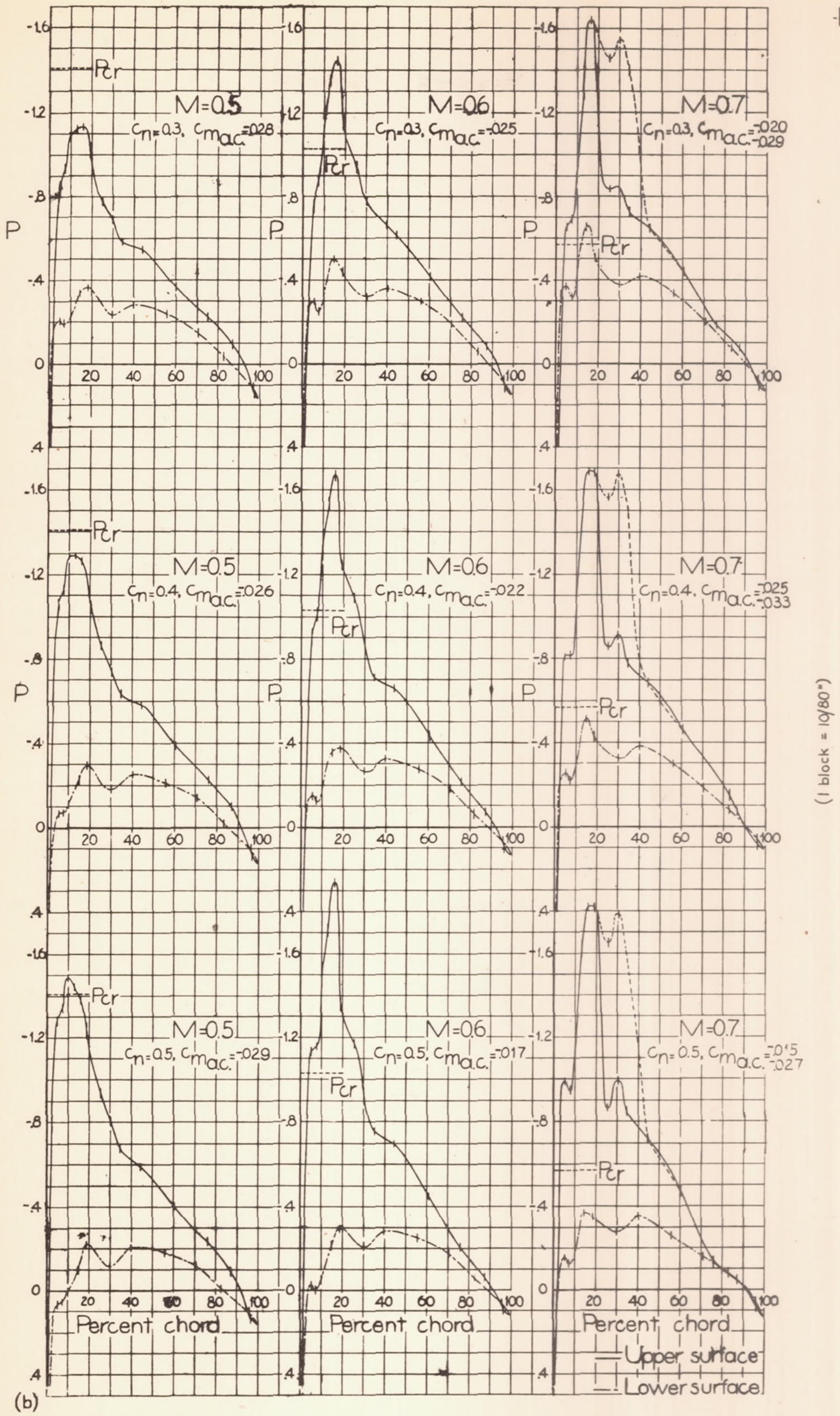
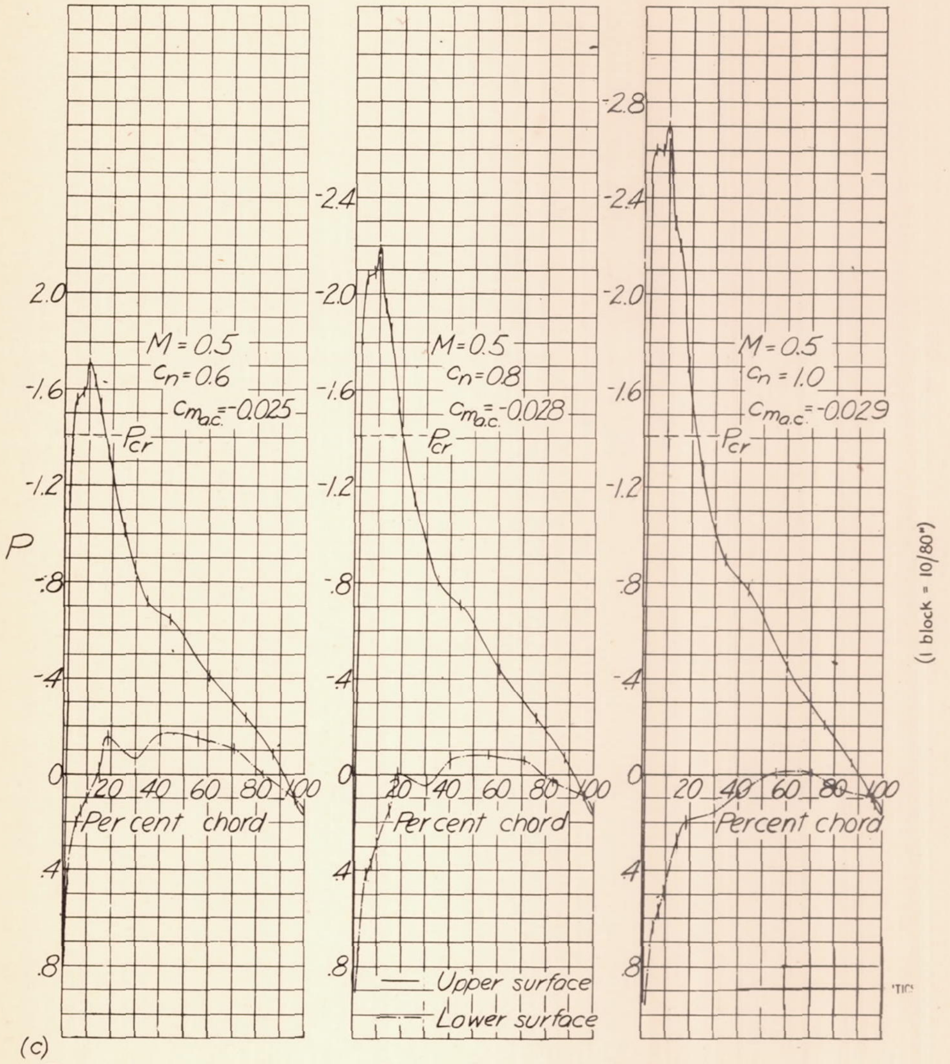


Figure 16: Average pressure distribution at several Mach numbers.



(b)

Figure 16. Continued Average pressure distribution at several Mach numbers.



(c)

Figure 16.- Concluded. Average pressure distribution at several Mach numbers.

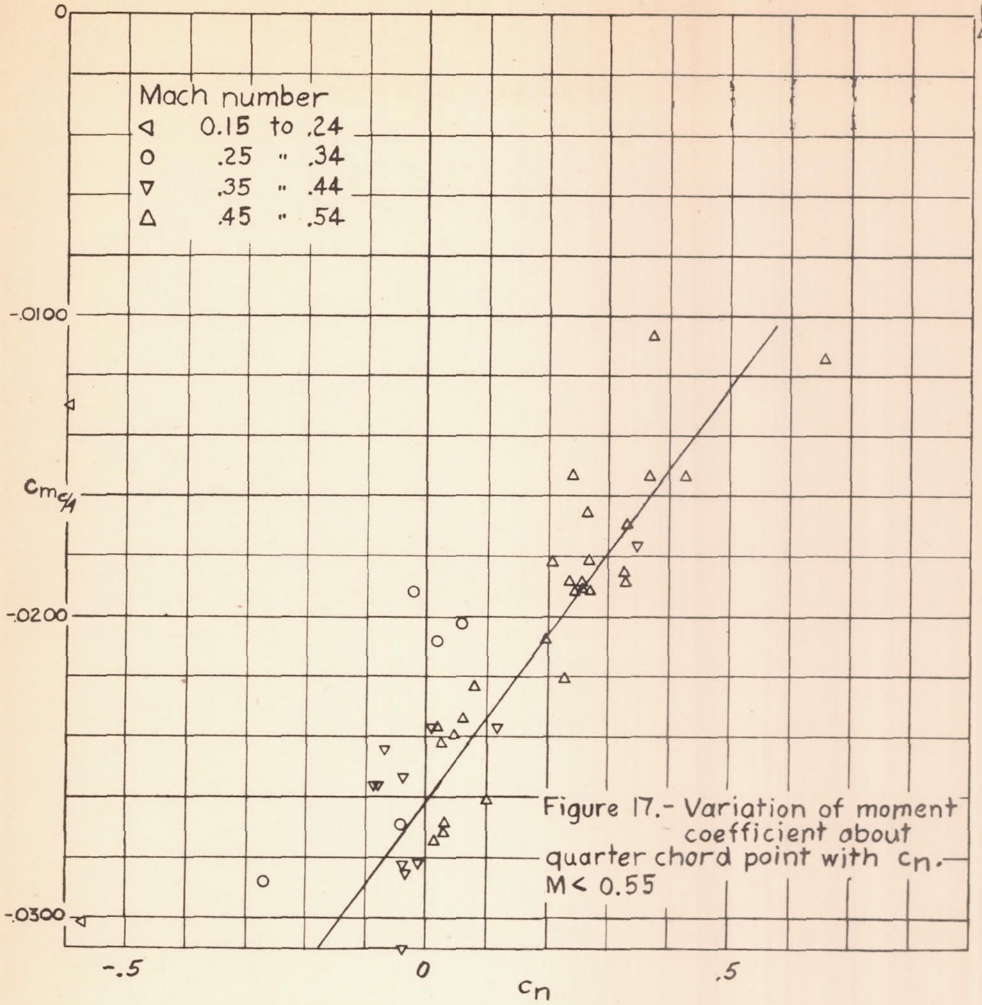


Figure 17.- Variation of moment coefficient about quarter chord point with c_n. M< 0.55

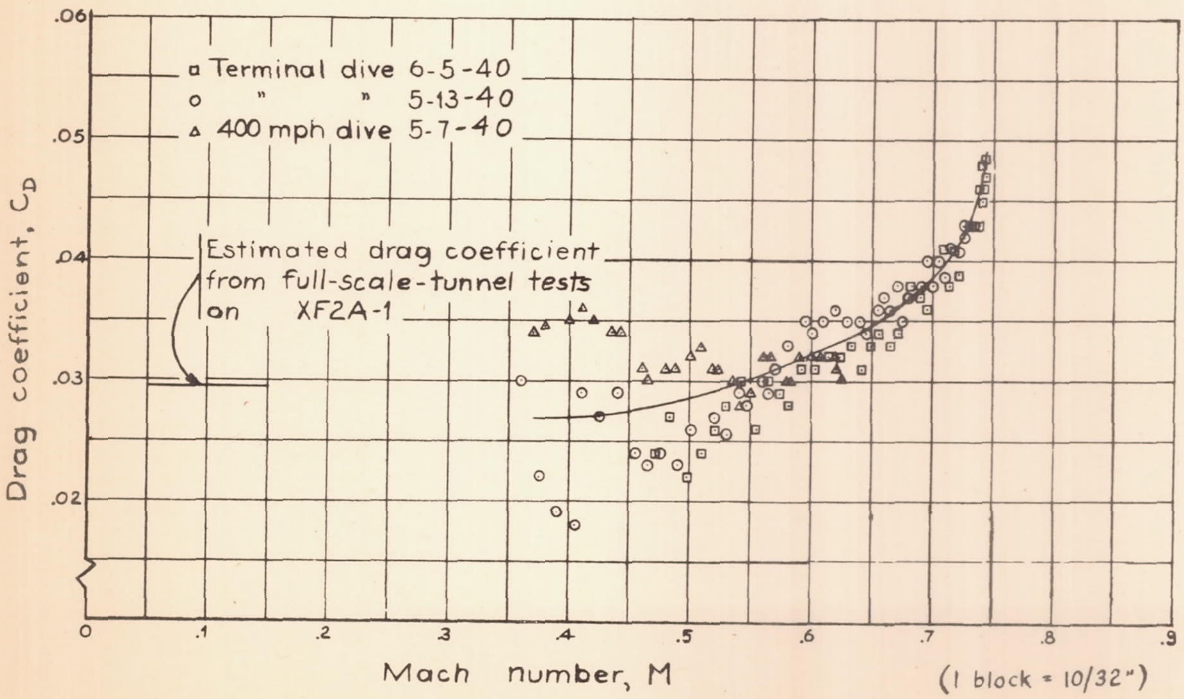


Figure 18.- Variation of drag coefficient with Mach Number as determined from accelerometer measurements (XF2A-2).

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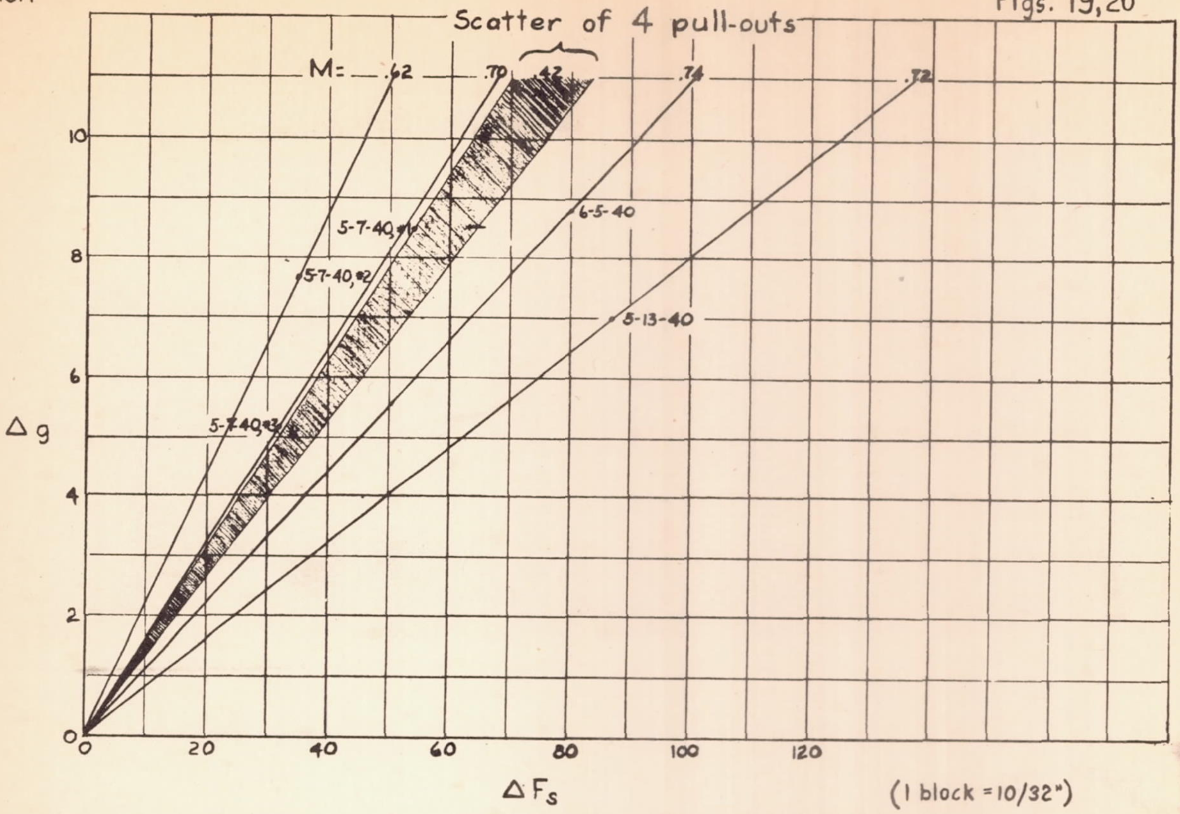

Figure 19.- Variation of load-factor increment with increment of stick force.

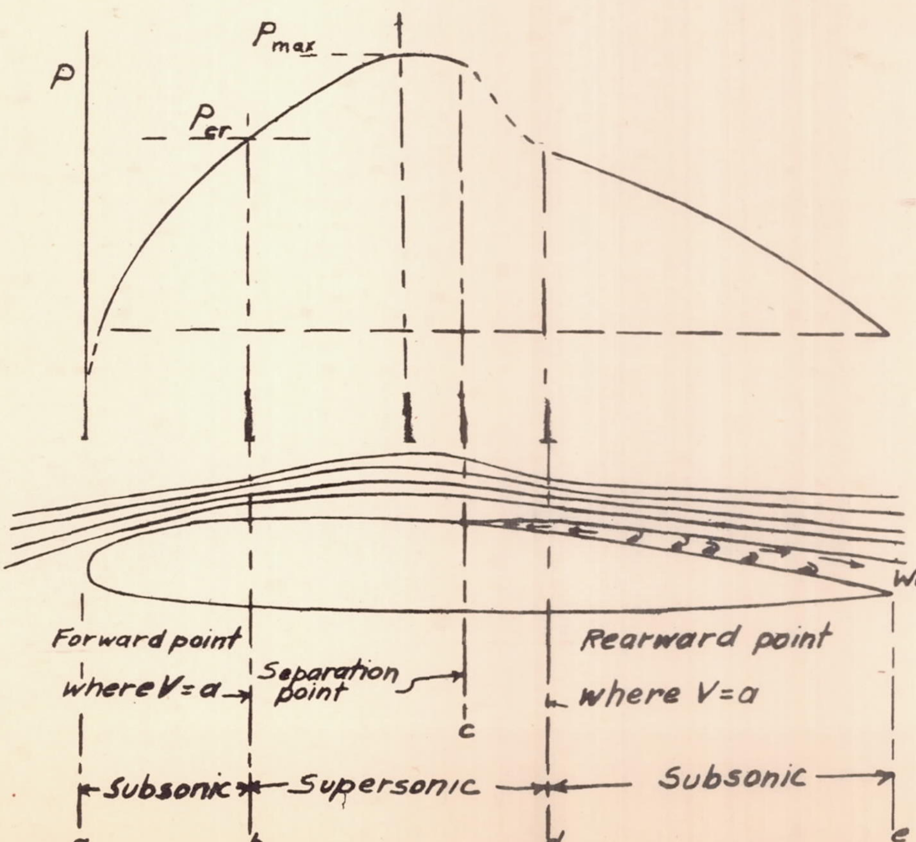

Figure 20.- Assumed flow at compressibility burble.