ARR No. 5G20

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WARTIME REPORT

ORIGINALLY ISSUED

September 1945 as
Advance Restricted Report 5G20

AN INVESTIGATION OF A THERMAL ICE-PREVENTION SYSTEM FOR A C-46 CARGO AIRPLANE

VII - EFFECT OF THE THERMAL SYSTEM ON THE WING-STRUCTURE STRESSES AS ESTABLISHED IN FLIGHT

By Alun R. Jones and Bernard A. Schlaff

Ames Aeronautical Laboratory
Moffett Field, California

WASHINGTON

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SUMMARY

As part of a comprehensive investigation of a thermal ice-prevention system for a Curtiss-Wright C-46 airplane, the change in stress at various locations in the wing outer panel caused by operation of the thermal system has been determined in flight.

Wire resistance-type strain gages and thermocouples were installed at numerous locations in the wing. Recordings of change in stress and temperature resulting from operation of the wing-heating system were obtained for two speeds in level flight and one speed in a 2g bank.

Curves are presented showing the chordwise variation of stress and temperature for two wing stations. In order to afford some indication of the magnitude and seriousness of the stress changes, they are compared with the allowable stresses for the unheated wing.

Although the test results are directly applicable to the C-46 wing only, two general conclusions of interest in the design of wing heating systems are presented. These are (1) the operation of a wing-leading-edge thermal ice-prevention system can result in stress changes which may (depending upon upon the unheated wing margins of safety) be negligible for regions aft of the double-skin region but will require investigation for the leading edge; and (2) local thermal stresses, possibly of critical magnitude, can be induced in sheet-stiffener combinations near the leading edge as a result

of temperature gradients in the order of 30° F between the stiffener and the adjacent skin.

INTRODUCTION

In the development of thermal ice-prevention systems for airplanes by the NACA, the principal objective has been to provide satisfactory and reliable systems for operation of the airplanes in icing conditions. Secondary problems, such as the effect of operation of the thermal systems on the airplane stresses, were afforded sufficient consideration to insure safe conduct of flight tests but were not examined in detail. The general acceptance of the practicability of thermal ice prevention for airplanes, however, has made necessary an investigation of these secondary problems in order to establish their magnitude and the degree of consideration to be assigned to them in future designs.

A preliminary examination of the effect of operation of a wing heating system upon the wing structure indicated that three factors were of importance: namely, (1) a reduction in strength and elasticity of the wing material caused by elevated temperatures, (2) increases in stress in the heated leading edge because of restrained thermal expansion, and (3) increases in stresses in the remainder of the wing, which is relatively unheated, caused by expansion of the leading-edge region. Although the first two items listed were the most obvious on first consideration of the problem, the incipient failure of a Lockheed 12A wing outer panel at the rear spar due to excessive expansion of the leading edge, as described in reference 1, was evidence that the last item might prove to be the most critical of the three.

The purpose of the investigation reported herein was to measure the changes in stress in a typical wing which result from the operation of a thermal ice-prevention system in order to establish the magnitude and seriousness of these changes in stress upon the structural integrity of the wing. The tests were conducted as the seventh part of a comprehensive investigation of a thermal ice-prevention system for a C-46 airplane. The first six parts of the investigation are presented as references 2 to 7.

The flight tests were conducted at the Ames Aeronautical Laboratory of the National Advisory Committee for Aeronautics, Moffett Field, Calif., at the request of, and in cooperation with, the Air Technical Service Command of the U. S. Army Air

Forces. The authors wish to acknowledge with appreciation the valuable assistance, during the ~~conduct of~~ the flight tests, of Mr. Robert Deland of the Curtiss-Wright Corporation.

Description of Equipment

The C-46 airplane (fig. 1) is a twin-engine, low-wing, transport-cargo monoplane powered by Pratt & Whitney model R-2800-51 engines having a sea-level rating of 2000 horsepower each. The wing span is 108 feet, the wing area 1360 square feet, and the normal gross weight 45,000 pounds.

The details of construction of the left-wing outer panel in which the stress measurements were obtained are shown in figure 2. The wing is of all-metal stressed-skin construction with spars at 30 and 70 percent of the chord. The skin is reinforced with spanwise hat sections and extruded stringers. At the wing root (station 0) the outer panel is attached to the inboard section through splice angles along the upper and lower surfaces of the wing. The wing profile varies from an NACA 23017 section at station 0 (chord = 198 in.) to an NACA 4410.5 section at station 412 (chord = 66 in.).

A complete description of the revisions to the C-46 airplane for thermal ice prevention is given in reference 4; however, a brief outline of the alterations as they affect the wing structure will also be given herein. The heated air for ice prevention is obtained from exhaust-gas-to-air heat exchangers installed on each side of the nacelles as may be seen in figure 1. The air from the outboard exchangers is directed to the outer-panel leading-edge heating system shown in figures 2 and 3. The details shown in figure 3 comprise the major revisions to the wing, the only other alterations being small reinforced holes in the spar webs (fig. 2) for circulation of the heated air.

The changes in stress were measured with standard Baldwin-Southwark wire resistance-type strain gages. In determining the location of the gages, consideration was given to the stress reports for the unheated wing and the measured temperature distribution in the wing as presented in references 5 and 7. The stress reports showed that the margins of safety were appreciably lower at the wing root than near the tip; while the temperature data indicated that, in general, the spanwise distribution of temperature rise for a given chord location was approximately constant. The assumption was made that, for a given chord location, the stress change along the span would be substantially constant with the application of heat; and, therefore, the gages were located at two stations relatively near the root (47 and 137 in. from station 0).

The individual locations of the strain gages and thermocouples are shown in figures 4 and 5. Two general types of gages were employed: namely, single-element gages on structural members such as hat sections and spar caps where the direction of stress was known, and triple-element gages on the wing skin where the direction, as well as the magnitude, of the maximum stress had to be established. All the single-element gages (hereinafter referred to as plain gages) are designated in figures 4 and 5 by the letter G before the gage number and were Baldwin-Southwark type A-7 gages. The triple-element gages (hereinafter referred to as rosette gages) are designated in figures 4 and 5 by the letters RG and were Baldwin-Southwark type ABR-4 gages forward of 10-percent chord and type AR-1 gages aft of that point. The ABR-4 gage is a high-temperature type required on the heated leading edge. The discontinuities in the strain-gage numbering system are the result of omitting from the list of gages those which failed.

The plain gages were installed with their strain-sensitive axes parallel to the longitudinal axes of the structural members. The rosette gages were oriented as shown in figure 6, which also gives the designation for the three strain elements. The fact that the high-temperature rosettes were supplied in a delta arrangement and the low-temperature rosettes in a 45° pattern has no particular significance. Aft of the double-skin region the rosette gages were mounted in pairs, back to back, and connected in series in order to eliminate strain indications caused by possible local skin buckling. This procedure was not feasible in the double-skin region but was considered unnecessary because of the stabilizing effect of the inner corrugations. Two typical plain-gage and one rosette-gage installations are shown in figure 7.

The strain-gage recording equipment is shown installed in the airplane in figure 8 and consisted basically of a 12-channel oscillograph, three bridges of four channels each, and three amplifiers of four channels each. The strain-gage switch box shown in figure 8 provided a means for connecting any 12 gages at one time to the recording oscillograph. A schematic diagram showing the complete circuit for one channel is presented in figure 9. The active and dummy gages were connected in two arms of a Wheatstone bridge circuit, the remaining two arms consisting of variable resistors in the four-channel bridge boxes. Any gage could be connected into any channel of the oscillograph by plugging it into the proper outlets of the strain-gage switch box. The change in resistance of the copper strain-gage lead wire caused by elevated temperatures in the wing was considered to be of sufficient magnitude to require the use of a compensation lead wire installed adjacent to the active lead wire of each gage. Means for calibration of all channels of the recording oscillograph at any time during the test program was provided by the installation of a variable-resistor switch and a channel-selector switch in parallel with the dummy gages.

Surface-type iron-constantan thermocouples, rolled to a thickness of 0.002 inch, were cemented to the aluminum surface with the junctions within one-quarter of an inch of the strain-gage elements. The thermocouples are designated by the same number as the corresponding strain gage and are prefixed with the letter T. (See figs. 4 and 5.) A thermocouple designation shown in figures 4 and 5 which is not accompanied by a corresponding gage number indicates the gage failed. In the case of the rosette gages aft of the double-skin region, where two gages were installed at each location, a thermocouple was placed on each side of the skin in the event that an appreciable temperature gradient might occur. The temperature readings were recorded with a Brown recording self-balancing potentiometer shown in figure 8. Additional thermocouples which had previously been used in the performance-test installation (reference 5) were also connected to the Brown recorder. These additional thermocouples permitted the measurement of the ambient-air temperature, the air temperature at the heat-exchanger outlet, and the air and skin temperatures in the double-skin region at stations 159 and 380.

Standard NACA instruments were installed to record airspeed and normal acceleration. An NACA timer, shown installed at the left of the airspeed recorder in figure 8, was used to synchronize the airspeed, accelerometer, and oscillograph records. The accelerometer was installed on the floor of the airplane at the center of gravity and oriented to record accelerations normal to the wing chord at station 0. The airspeed recorder was connected to the service Kollsman airspeed-head installation and the error in static-pressure reading was determined in flight for all the test conditions with a trailing static pressure head.

TEST PROCEDURE

Instrument Technique

When employing strain gages to measure stress changes in a structural member which is subjected to a temperature, as well as a stress variation, a question arises concerning the differentiation between movement due to stress and that due to thermal expansion. Superimposed on this complication is the unknown effect of elevated temperatures on the resistance of the strain-gage material. One installation method commonly used to overcome these problems consists of locating the dummy strain gage beside the active gage but cemented to a small piece of the structure metal which is free to expand. The compensation is based on the assumption that the piece of metal to which the dummy gage is attached will assume the same temperature as the surface upon which the active gage is installed. This method was considered unreliable in the case of the C-46 wing.

Laboratory calibrations were made to investigate the stress-strain curves which would result if the dummy gage were maintained at a constant temperature while the aluminum calibration specimen and the active gage were subjected to various stresses at different temperatures. The results of this calibration for a typical strain gage used in the investigation are presented in figure 10. The important fact to note in figure 10 is that the calibration curves form a series of parallel straight lines. This means that the sensitivity factor of the gage (ratio of unit change in resistance to unit strain causing this change) is independent of the initial stress and temperature throughout the calibration range.

The method of applying the calibration curves of figure 10 to the determination of stress changes in the heated wing was as follows: Assume that point a of figure 10 represents the stress and temperature conditions at a gage in the wing before the wing heat was applied. The temperature of point a was known but the stress was not; however, since only stress changes were to be evaluated, the strain-recording equipment was adjusted to zero reading (or balanced). The heat was then directed to the wing and the changes in temperature and strain-gage reading were recorded. If the temperature change had been 30° F and the strain-gage reading agreed with point b, pure expansion and no change in stress would be indicated. If the strain-gage reading corresponded to point c, however, a change in stress equal to the distance between points c and d (or an increase in tension) was indicated. It is evident that the same result would be obtained for any other initial point of e instead of a since the curve slopes were all equal. Hence the absolute value of the stress at the time of balancing the strain equipment had no bearing on the result.

Flight Procedure

All the flight tests were made at a pressure altitude of 10,000 feet and a take-off gross weight of 45,700 pounds. Data were obtained at two airspeeds (110 and 135 mph, indicated) in level flight and at 155 to 165 miles per hour in a 2g bank at constant altitude. The airspeed in the 2g bank was calculated to give the same lift coefficient as that existing in the level-flight condition at 110 miles per hour in order that the two wing-loading conditions would be directly comparable with respect to chordwise pressure distribution.

In the case of the level-flight tests, the airplane was stabilized in flight at the desired speed and with the wing-heating system in the off condition. Twelve gages were plugged into the strain-gage switch box, the equipment was balanced, and a calibration record was obtained. A 2-minute record was then taken with all the recording instruments. The length of the record was established by the time required for the thermocouple recorder to complete one cycle. At the completion of the heat-off recording, the heated air was directed to the left-wing outer panel. Preliminary tests indicated that 4 minutes was an adequate period of time for stabilization of the wing temperatures. At the end of 4 minutes, therefore, a second 2-minute record was taken with all recording instruments. This procedure, the heat-off recording and heat-on recording, was then repeated with 12 new gages until the entire gage installation had been included. The stress changes due to wing heating were taken as the difference between the stress records for the two flight conditions just outlined with corrections being made for temperature effect.

In the case of the 2g banks, a slightly different procedure was followed because of the difficulties involved in maintaining a reasonably constant acceleration over the entire 2-minute period required by the thermal recorder. The airplane was flown at an indicated airspeed of 110 miles per hour in a level attitude with the heat off, and the strain-gage equipment was balanced. Records were obtained for this condition with all recording equipment. The airplane was then banked to produce a 2g normal acceleration at an indicated airspeed of 155 to 165 miles per hour and constant altitude. At a signal from the pilot indicating stable flight conditions, a 1-minute record was obtained. The airplane was returned to the level attitude (thermal system still heat off), the airspeed was set at 110 miles per hour, the strain-gage equipment was rebalanced, and complete recordings were taken. The engine power was then increased to that required in the banked condition (approx. 32 in. Hg manifold pressure and 2200 rpm, engine speed), the thermal system switched to heat on, and the wings allowed to heat for 4 minutes in level flight. A complete temperature record was taken at the end of this period, and the airplane was then banked to the 2g attitude. Again at a signal from the pilot, a 1-minute record of all recording instruments was obtained. Although this 1-minute period was adequate for recording the stress records, only half the temperature data could be recorded in this interval. The temperature readings in banked flight, however, agreed within the limits of experimental error with the temperatures recorded immediately before the bank. Hence the level-flight-temperature data were considered satisfactory for the banked condition.

The differences between the stress records for level flight at 110 miles per hour, heat off, and banked flight, heat off, supplied the stress changes due to increased wing loading with no increase in temperature. The differences between the stress records for level flight at 110 miles per hour, heat off, and banked flight, heat on, supplied the stress changes due to increased wing loading plus those due to wing heating. The stresses due to wing heating alone were determined by subtracting the stresses due to increased wing loading from those due to increased wing loading plus heating.

PRECISION OF DATA

An exact determination of the accuracy of measurement of the stress changes was not possible because the evaluation of some of the factors involved was not practicable. A laboratory check of the precision of the entire strain-gage circuit and the flight-test procedure, however, presented some indication of the degree of accuracy of the final results. This laboratory check was undertaken with the equipment shown in figure 11. A strain gage and a thermocouple, each identical to those installed in the C-46 wing, were installed on a tension specimen of aluminum alloy 24S-T alclad. The specimen was subjected to various known stresses at different temperatures (heating being obtained with the lamps shown in fig. 11) and the stress and temperature changes recorded with the flight-test equipment.

These laboratory records were evaluated using the same calibration curves employed in establishing the flight-test results and were compared to the known stress changes. A precision of measurement of ±200 psi was established. Adding to this the unknown errors in constancy of flight conditions, effects of vibration on instruments, and so forth, an accuracy of ±400 psi for the stress-change data presented herein is believed to be reasonable.

Based on laboratory calibrations of the iron-constantan thermocouple wire and the Brown recording potentiometer, plus an unknown installation error, the temperatures presented are considered accurate to ±3° F for the strain-gage locations and ±5° F for the remaining installations. The 25° F correction for surface thermocouples forward of the baffle (5 percent chord) mentioned in reference 5 has been applied to the data presented for stations 159 and 380.

The airspeed data are considered accurate to ±1.5 miles per hour for the level-flight tests and to ±3 miles per hour for the banked condition. The error in recorded acceleration may be taken as ±0.02g.

RESULTS

The test conditions and the resulting total heat flow to the left wing are given in table I. The stress changes measured by each gage element are presented in table II, part 1, for the plain gages and in table II, part 2, for the rosette gages. The data presented in table II have been corrected for temperature effect by the method previously outlined in the discussion of flight procedure and figure 10, and represent actual changes in stress for the plain gages. For the rosette-gage data (pt. 2 of table II) a further correction based on the Poisson ratio effect on the gage readings must be applied.

The maximum and minimum changes in normal stress, the maximum change in shear, and the direction of action of these stress changes were computed from the rosette-gage data (table II, pt. 2) and are presented in table III. The terms "maximum normal" and "minimum normal" are used here in their algebraic sense. The Poisson ratio correction has been applied, and the data presented in table III represent actual stress changes. The angle θ in the table gives the direction of the line of action of the maximum change in normal stress and is measured from the wing chord with the positive direction as shown in figure 6. The line of action of the minimum change in normal stress is at right angles to that of the maximum normal stress, and the maximum change in shear is at an angle of 45° to either of these axes. The rosette-gage stresses were also resolved in the spanwise and chordwise directions and these data are presented in table IV.

Representative temperature data are presented in table V. A study of the temperature data indicated that the temperature differences between heat on and heat off at a given flight condition were substantially the same and, therefore, only one set of data is presented for each flight condition. In evaluating the strain-gage recordings, however, the actual temperature recorded at the gage for that particular run was used.

DISCUSSION

The anticipated effect of heating the wing leading edge on the chordwise wing stress distribution was (1) an increase in compression for the double-skin region as a result of resistance to thermal expansion, (2) an increase in tension in the region between the double skin and the 30-percent spar caused by the expanded leading edge pulling on the relatively cool afterbody, and (3) an increase in compression at the 70-percent spar caused by the two spars and the wing skin acting as a box beam to resist the moment imposed by the expanding leading edge. This general trend is evident in all the curves showing the chordwise distribution of stress change. (See fig. 12.) These curves are based on the values of stress change normal to the chord presented in tables II and IV. The chordwise temperature distribution has been added to the stress-distribution curves in order to facilitate the interpretation and explanation of the test data.

A comparison of the stress changes for the three flight conditions is presented in figures 12(g) and 12(h). Although the increased heat supplied to the wing in the 2g bank is evidenced by increased compression at the leading edge and some deviation of the 2g curve from the level-flight curves at other chord positions; the over-all agreement between the stresses for the three conditions is considered sufficient to allow them to be discussed as one general trend.

The expected compression at the leading edge and 70-percent spar is evident in figure 12. From 10-to 30-percent chord, however, considerable variation in the data is noted. This apparent discrepancy at first appears to refute the anticipated general trend, but on further examination is seen to be the result of local conditions superimposed upon the over-all pattern. An example of this effect is shown by the data presented for gages 41, 51, and 52 in figure 12(d). The expansion of the leading edge would be expected to pull on the hat section containing gage 51 and exert tension similar to gage 36 on the lower surface. Apparently, however, the heated air discharging from the double skin heated the hat section considerably (note temperature distribution) and the restrained expansion induced compressive stresses which were larger than the induced tension. The expansion forces of the hat section, in turn, placed the colder skin in tension as signified by the indication of gage 41. At the location of gage 52 the stress has again reversed, the actual value at gage 52 being the result of the combined effects of several factors of unknown magnitude.

Another interesting example of large stress changes caused by local conditions (temperature gradient between stiffener and skin) is presented by the indications of gages 50 and 60 in figure 12(d). Although the two gages are located within 1 inch of each other, the stress in the angle was about 3000-psi compression; whereas the skin was practically unstressed. An examination of the temperature-distribution curve shows that the angle temperature was approximately 10° F greater than the skin temperature. For a completely restricted aluminum member the increase in compressive stress for a 10° F temperature rise is about 1300 psi. It appears reasonable, therefore, to picture the stress changes occurring at gages 50 and 60 as a uniform increase in stress in both the angle and skin until a value of about 1500-psi compression has been achieved. This is followed by an increase in compressive stress in the angle (caused by an increase in temperature of the angle only) and a decrease in stress in the skin because of the stretching action of the angle.

Because local conditions in some cases caused large stress differences at a given chord location (the actual deviation depending upon whether the strain gage was mounted on the skin or on a longitudinal stiffener) the stress curves presented for station 137 represent some mean values of stress for the region from 10- to 30-percent chord. Although the same local heating existed at station 47 (note temperature distribution for upper surface, figs. 12(a), 12(b), and 12(c)), the scatter of data was not obtained because all the gages in the local heating region were mounted on the stiffeners. The stress curve for the upper surface is, therefore, more representative of stress in the stiffeners than in the skin.

The belief that the scatter of data between 10- and 30-percent chord is largely attributable to localized heating rather than inaccurate measurements is verified by an inspection of figures 12(g) and 12(h). These curves show consistency of the data between the three flight conditions for the regions where temperature gradients between the internal structure and skin were negligible. Further examples of this local heating effect between 10- and 30-percent chord could be cited; however, the purpose of this report is to determine, in general, the magnitude and importance of the stress changes rather than to present a detailed investigation of the C-46 wing.

In order to obtain some indication of the seriousness of the stress changes, the test data were compared with the critical values as specified in the wing outer-panel stress analysis as prepared by the airplane manufacturer. The flight conditions considered critical for the wing outer panel in the stress analysis were conditions I, II, III_1, and VII as presented in reference 8. The assumption is made that the thermal system might be in operation during any of these flight conditions; and, therefore, the measured stress changes are compared with (1) the allowable stress and (2) the critical margins of safety. A complete and precise comparison was not practicable because (1) some of the strain-gage locations were not considered critical in the stress analysis and, therefore, no margin of safety was presented, (2) the nearest station to 137 covered by the stress analysis was station 112 (or 25 in. inboard of 137), and (3) in the leading-edge region, the type of wing construction was considerably different from that used in the stress analysis.

The stress changes in the longitudinal stiffeners and spar caps ranged from 1 to 5 percent of the allowable stress with the exception of a few stiffeners near the leading edge. Whether or not such changes are critical depends, of course, upon the particular margin of safety for each stiffener. For the specific case of the C-46 wing the margins were sufficiently high in most cases to absorb an increase of 5 percent of the allowable. It would be inadvisable, however, to make the general statement that increases in stress of this magnitude are not critical because of the specific nature of the problem. For example, the increase in stress of the T-section containing gage 52 at station 137 was only 800-psi compression, or about 2 percent of the allowable stress for that member. This increase, however, was sufficient to reduce the already critical margin of safety from 0.06 to 0.03. The spar caps, on the other hand, because of their large margins of safety may be considered as unaffected by the thermal stress changes.

In the case of longitudinal stiffeners located near the point of discharge of the heated air from the double-skin region (gages 50 and 51), the changes in stress ranged from 10 to 16 percent of the allowable stress. The value of 16 percent was obtained with gage 50 and the increased stress (2800-psi compression) would be sufficient to change the margin of safety for the angle in compression from 0.08 to -0.08. Although this negative margin is small the important fact to note is that impingement of the heated air directly upon a structural member may induce stresses which are large enough to change the margin of safety from a satisfactory to an unacceptable value. In this connection it is of interest to note the effect of stringer configuration on the temperature distribution. An inspection of the upper-surface temperatures for either station presented in figure 12 shows that a larger temperature gradient between the skin and the adjacent stiffeners, and hence larger local stress changes, was measured for the hat sections than for the T-sections. In regions where heated-air discharge is apt to cause local overheating of the wing structure, therefore, the air should be directed away from the structure by nonstructural vanes and, if practicable, the structural member should have a maximum of area in contact with the skin.

The maximum shear-stress changes in the skin aft of the double-skin region, as measured by rosette gages 4, 32, 41, and 60 and presented in table III, were all of low magnitude (about 2 percent of the allowable shear stress). The margins of safety at these locations were all large and quite capable of absorbing the 2-percent increase in stress. As mentioned previously, however, this general conclusion should not be applied indiscriminately to other wing structures.

The largest changes in stress were measured at the wing leading edge, but are particularly difficult to interpret because of the lack of data on the allowable leading-edge stresses. Some test data are available which can be reasonably applied to the C-46 leading-edge construction, prior to revisions to incorporate the thermal system, but information on the double-skin type of construction did not appear to be available. In order to obtain some indication of the seriousness of the measured leading-edge stress changes, the revised wing stresses will be compared to those allowable for the unaltered leading edge. The structural effect of the inner corrugations will of necessity remain an unknown factor in this report.

In the wing structural analysis for the unrevised leading edge the skin was considered to carry only shear due to torsion and beam bending. A stress element on the leading edge at station 47 (location of gage 7) would therefore be represented as shown in figure 13(a). The chordwise shear stress of 7300 psi represents the maximum value investigated in the stress analysis, and compression in the leading edge due to chord bending was considered negligible. Operation of the thermal system produced additional normal stresses of approximately 4000-psi spanwise compression, and 5000-psi chordwise tension (data for gage 7, table IV), as shown in figure 13(b). Since these normal stresses can be considered the principal stresses (table III, angle θ, approx. 0°), no additional shear is added to the stress element in a chordwise direction. The actual stresses existing at gage 7 during operation of the thermal system, therefore, may be taken as shown in figure 13(b).

The allowable stress for the unheated leading edge was presented in the stress analysis as the stress at failure in an unstiffened thin-walled cylinder in pure torsion. (See reference 9.) From reference 10, the heated leading edge can be approximated by a cylinder in combined loading for which the three allowable stresses are related by the equation as follows:

$$\left(\frac{f_c}{F_c}\right)^a + \left(\frac{f_s}{F_s}\right)^b + \left(\frac{f_t}{F_t}\right)^c = 1 \qquad (1)$$

where

f_c allowable compressive stress in combined loading

F_c allowable compressive stress for pure compression

f_s allowable shear stress in combined loading

F_s allowable shear stress for pure shear

f_t allowable tension stress in combined loading

F_t allowable tension stress for pure tension

a, b, c empirical exponents

No data were available on tests of cylinders in combined compression, tension, and shear from which exponents a, b, and c could be evaluated. If the skin is assumed capable of developing the ultimate tension stress in pure tension (i.e., if F_t = 56,000 psi), the factor f_t/F_t may be neglected. A consideration of references 10 and 11 relative to the aerodynamic shear stresses and the thermal compressive stresses indicated that equation (1) could be expressed as

$$\frac{f_c}{F_c} + \left(\frac{f_s}{F_s}\right)^2 = 1 \qquad (2)$$

The allowable shear stress due to pure torsion F_s may be taken as the same value presented for the unheated leading edge or 9700 psi. This value was obtained from reference 9 for values of $l/r = 2.37$ and $r/t = 158$

where

l length of cylinder

r radius of cylinder

t skin thickness

In a like manner the value of F_c is taken from reference 12 (curve c of fig. 7) to be 18,000 psi. In a rigorous analysis these allowable stresses would be reduced by about 10 percent because of the effect of elevated temperatures on the tangent modulus of elasticity. (See reference 13.) This factor has been neglected, however, because of the general nature of the discussion. If the torsional and bending shear due to aerodynamic loading is assumed to remain constant, then f_s = 7300 psi. (See fig. 13.) Substitution of the foregoing values in equation (2) provides an allowable compression stress in combined loading of f_c = 7800 psi.

A comparison of the actual compression stress (4000 psi) with the allowable stress (7800 psi) indicates that the thermal system has not resulted in a critical stress condition. Such a statement, however, should be qualified with a few remarks which are of interest. The compression load in the leading edge due to chord bending, although negligible for the unrevised wing, cannot be neglected for the heated wing. The wing structural analysis indicates that the compression due to chord bending is about 5000 psi, which, when added to the thermal compression, gives a total value of 9000 psi. This value is 1200 psi in excess of the allowable. It is not intended that these figures should be taken as a precise indication that the allowable has been exceeded, because of the uncertanties involved in equation (2) and the unknown increase in allowable stress afforded by the inner corrugated skin. It is intended, however, to point out that the thermal stresses superimposed on the aerodynamic stresses might produce a critical condition in combined loading, particularly in cases where the inner skin cannot appreciably increase the allowable stress of the outer skin.

The chordwise tension stress was not anticipated prior to the investigation and was probably the result of larger expansion forces in the corrugated inner skin than in the cooler outer skin. Although the tension stress was considered negligible in the previous discussion, it might prove of importance in some instances of combined loading, particularly if the inner skin has considerable resistance to buckling under chordwise compression. It is evident from the foregoing discussion that the general acceptance of thermal ice prevention has resulted in a need for structural-test data on the strength of internally stiffened cylinders, which are representative of practical double-skin leading-edge designs, under combined loading.

The effect of elevated temperatures on the strength and elasticity of the wing material is a subject of considerable interest in the design of thermal ice-prevention equipment, but a detailed discussion is regarded as beyond the scope of this report. The generally accepted temperature limit for aluminum-alloy structural members is 200° F. Above this temperature, reductions in the allowable stresses are required. An examination of table V indicates that certain leading-edge components such as nose ribs may exceed this critical temperature, resulting in a lowering of the allowable stress, but structure aft of the leading edge can be maintained at subcritical temperatures. Although researches such as that presented in reference 13 are providing valuable information on the strength of aircraft structural materials at elevated temperatures, further investigations appear desirable to establish the effect of cyclic heating over a long period of time on the strength of aircraft materials and to examine the phenomena associated with metal "creep" at increased temperatures. The extension of airplane speeds into the sonic range will result in aerodynamic heating of the airplane surfaces of appreciable magnitude, and thermal stress problems will be unavoidable. Thus additional research on wing thermal stresses, particularly in the leading-edge region, will be of value in the development of wings for high-speed airplanes as well as in the development of thermal ice-prevention systems.

CONCLUSIONS

From flight tests of the change in stress in wing structure resulting from the operation of the thermal ice-prevention system, the following conclusions are made:

1. The operation of a wing-leading-edge thermal ice-prevention system can result in stress changes which may (depending upon the unheated wing margins of safety) be negligible for regions aft of the double-skin region but will require investigation for the leading-edge region.

2. Local thermal stresses, possibly of critical magnitude, can be induced in sheet-stiffener combinations near the leading edge as a result of temperature gradients in the order of 30° F between the stiffener and the adjacent skin.

Ames Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Moffett Field, Calif.

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TABLE I.- TEST CONDITIONS FOR DETERMINATION OF STRESS CHANGES IN C-46 WING RESULTING FROM OPERATION OF THERMAL ICE-PREVENTION SYSTEM

Flight condition	Gross weight at take-off, pounds	Pressure altitude, feet	Ambient-air temperature, °F	Engine speed, rpm	Manifold pressure, inches of mercury	Air temperature at outlet of heat exchanger, °F	Air flow to wing, pounds per hour	Heat flow to wing, Btu per hour
110 mph correct I.A.S. level flight	45,700	10,000	44	1900	24	355	3,485	261,000
135 mph correct I.A.S. level flight	45,700	10,000	44	1900	26	335	4,020	282,000
155 to 165 mph correct I.A.S. 2g bank	45,700	10,000	44	2200	32	353	4,960	370,000

TABLE II.- CHANGES IN STRESS IN WING OUTER PANEL OF C-46 AIRPLANE RESULTING FROM OPERATION OF THE THERMAL ICE-PREVENTION SYSTEM.
PART 1 - PLAIN-GAGE LOCATIONS

Station 47											
Flight condition \ gage	G15	G17	G18	G19	G22	G24	G25	G26	G27	G30	G31
110 mph correct I.A.S. level flight	+770	+1100	+1040	+1480	+860	-450	+60	+530	+980	-710	-570
135 mph correct I.A.S. level flight	+600	+810	+1130	+1520	+800	-650	-100	+580	+950	-780	-750
155 to 165 mph correct I.A.S. 2g bank	+750	+110	+580	+580	-20	-1650	-470	+350	+970	-770	-900
Station 137											
Flight condition \ gage	G45	G47	G48	G50	G51	G52	G54	G55	G59	G57	G58
110 mph correct I.A.S. level flight	-520	+1280	+1610	-2500	-2070	-490	+100	+420	+310	-1140	-1290
135 mph correct I.A.S. level flight	-750	+1070	+1390	-2100	-2230	+80	-160	+460	+330	-1280	-1410
155 to 165 mph correct I.A.S. 2g bank	-1990	+630	+590	-2340	-2650	-810	-270	+70	+540	-1560	-1490

Note: + denotes tension stress, pounds per square inch.
- denotes compression stress, pounds per square inch.

TABLE II.- CHANGES IN STRESS IN WING OUTER PANEL OF C-46 AIRPLANE RESULTING FROM OPERATION OF THE THERMAL ICE-PREVENTION SYSTEM

PART 2 - ROSETTE-GAGE LOCATIONS

High-temperature delta rosettes at station 47

Flight condition \ gage	RG5			RG6			RG7			RG8		
	f_1	f_2	f_3	f_1	f_2	f_3	f_1	f_2	f_3	f_1	f_2	f_3
110 mph correct I.A.S. level flight	+6490	+3270	+5400	+4780	-2250	-2780	+6820	-3450	-2100	+5870	-2120	-2000
135 mph correct I.A.S. level flight	+7060	+2320	+5600	+4260	-1810	-2510	+6030	-3580	-2260	+5680	-1890	-3060
155 to 165 mph correct I.A.S. 2g bank	+6890	+2830	+4860	+4470	-1650	-3430	+6870	-5200	-2650	+4930	-2820	-2520

High-temperature delta rosettes at station 137

Flight condition \ gage	RG36			RG37			RG38			RG39		
	f_1	f_2	f_3	f_1	f_2	f_3	f_1	f_2	f_3	f_1	f_2	f_3
110 mph correct I.A.S. level flight	+2420	-----	-1040	+6580	-2180	-2550	+5530	-5120	-4120	+4880	-2400	-4420
135 mph correct I.A.S. level flight	+2760	+3230	-960	+6130	-1420	-1500	+5020	-4500	-3700	+3850	-3310	-3110
155 to 165 mph correct I.A.S. 2g bank	+2800	+3290	+250	+8420	-980	-1180	+5750	-5940	-4550	+1800	-2270	-2190

Low-temperature 45° rosettes at stations 47 and 137

Flight condition \ gage	RG4 (station 47)			RG32 (station 137)			RG41 (station 137)			RG60 (station 137)		
	f_1	f_2	f_3	f_1	f_2	f_3	f_1	f_2	f_3	f_1	f_2	f_3
110 mph correct I.A.S. level flight	+290	+1350	+140	+500	-680	-150	-580	+2050	+580	+870	-490	+660
135 mph correct I.A.S. level flight	+380	+1010	+90	+530	-550	-90	-680	+1970	+300	+800	-40	+600

Note: + denotes tension stress, pounds per square inch.
- denotes compression stress, pounds per square inch.

Data uncorrected for Poisson ratio effect, see text.

TABLE III.- MAXIMUM AND MINIMUM CHANGES IN NORMAL STRESS AND MAXIMUM CHANGE IN SHEAR STRESS AT ROSETTE-STRAIN-GAGE LOCATIONS IN WING OUTER PANEL OF THE C-46 AIRPLANE RESULTING FROM OPERATION OF THE THERMAL ICE-PREVENTION SYSTEM

High-temperature delta rosettes at station 47

Gage / Flight condition	RG5				RG6				RG7				RG8			
	Maximum normal	Minimum normal	Maximum shear	[1] θ (deg)	Maximum normal	Minimum normal	Maximum shear	[1] θ (deg)	Maximum normal	Minimum normal	Maximum shear	[1] θ (deg)	Maximum normal	Minimum normal	Maximum shear	[1] θ (deg)
110 mph correct I.A.S. level flight	+8960	+6150	1420	$+20\frac{1}{4}$	+3410	-3660	3540	$-1\frac{3}{4}$	+5480	-4210	4840	$+3\frac{1}{2}$	+4750	-3100	3930	$+\frac{1}{4}$
135 mph correct I.A.S. level flight	+9560	+5340	2110	$+21\frac{1}{4}$	+3200	-3260	3230	$-2\frac{3}{4}$	+4610	-4430	4520	$+3\frac{1}{2}$	+4480	-3750	4120	$-3\frac{1}{2}$
155 to 165 mph correct I.A.S. 2g bank	+9010	+5490	1760	+15	+3290	-3900	3590	$-6\frac{1}{4}$	+5040	-6010	5520	$+5\frac{3}{4}$	+3600	-4020	3810	+1

High-temperature delta rosettes at station 137

Gage / Flight condition	RG36				RG37				RG38				RG39			
	Maximum normal	Minimum normal	Maximum shear	[1] θ (deg)	Maximum normal	Minimum normal	Maximum shear	[1] θ (deg)	Maximum normal	Minimum normal	Maximum shear	[1] θ (deg)	Maximum normal	Minimum normal	Maximum shear	[1] θ (deg)
110 mph correct I.A.S. level flight	-------	-------	-------	-----	+ 5400	-3570	4480	-1	+3250	-6950	5100	$+2\frac{1}{2}$	+3280	-5210	4250	-6
135 mph correct I.A.S. level flight	+4500	+520	1990	-33	+5300	-2210	3750	$-\frac{1}{4}$	+3000	-6170	4580	$+2\frac{1}{4}$	+2260	-4810	3540	$+\frac{3}{4}$
155 to 165 mph correct I.A.S. 2g bank	+4570	+1740	3150	$-34\frac{1}{4}$	+7880	-1650	4760	$-\frac{1}{2}$	+3200	-7900	5540	+3	+700	-3340	2020	$+\frac{1}{4}$

Low-temperature 45° rosettes at stations 47 and 137

Gage / Flight condition	RG4				RG32				RG41				RG60			
	Maximum normal	Minimum normal	Maximum shear	[1] θ (deg)	Maximum normal	Minimum normal	Maximum shear	[1] θ (deg)	Maximum normal	Minimum normal	Maximum shear	[1] θ (deg)	Maximum normal	Minimum normal	Maximum shear	[1] θ (deg)
110 mph correct I.A.S. level flight	+1870	+580	640	+65	+310	-580	440	$+3\frac{1}{2}$	+2110	+ 90	1010	$-93\frac{1}{2}$	+900	-330	620	-17
135 mph correct I.A.S. level flight	+1550	+ 530	510	+59	+400	-430	420	+4	+2000	-50	1020	$-97\frac{1}{4}$	+940	+210	370	-15

[1] For designation of positive θ see figure 6.

+ Denotes tension stress, pounds per square inch.

- Denotes compression stress, pounds per square inch.

Data corrected for Poisson ratio effect.

TABLE IV.- SPANWISE AND CHORDWISE STRESS CHANGES AT ROSETTE STRAIN-GAGE LOCATIONS IN WING OUTER PANEL OF THE C-46 AIRPLANE RESULTING FROM OPERATION OF THE THERMAL ICE-PREVENTION SYSTEM

High-temperature delta rosettes at station 47

Flight condition \ gage	RG5		RG6		RG7		RG8	
	Spanwise	Chordwise	Spanwise	Chordwise	Spanwise	Chordwise	Spanwise	Chordwise
110 mph correct I.A.S. level flight	+6390	+8550	-3660	+3410	-4180	+5470	-3100	+4750
135 mph correct I.A.S. level flight	+5890	+8980	-3250	+3190	-4400	+4580	-3720	+4460
155 to 165 mph correct I.A.S. 2g bank	+5710	+8750	-3820	+3250	-5900	+4950	-4020	+3600

High-temperature delta rosettes at station 137

Flight condition \ gage	RG36		RG37		RG38		RG39	
	Spanwise	Chordwise	Spanwise	Chordwise	Spanwise	Chordwise	Spanwise	Chordwise
110 mph correct I.A.S. level flight	-----	-----	-3570	+5400	-6940	+3230	-5040	+3180
135 mph correct I.A.S. level flight	+1700	+3310	-2210	+5300	-6170	+2990	-4810	+2260
155 to 165 mph correct I.A.S. 2g bank	+2630	+3665	-1650	+7880	-7880	+3180	-3340	++700

Low-temperature 45° rosettes at stations 47 and 137

Flight condition \ gage	RG4		RG32		RG41		RG60	
	Spanwise	Chordwise	Spanwise	Chordwise	Spanwise	Chordwise	Spanwise	Chordwise
110 mph correct I.A.S. level flight	+1630	+830	-570	+310	+2080	+120	-230	+790
135 mph correct I.A.S. level flight	+1280	+810	-420	+390	+1960	--30	+260	+890

Note: + denotes tension stress, pounds per square inch.
- denotes compression stress, pounds per square inch.

Data corrected for Poisson's ratio effect.

TABLE V.- TEMPERATURE CHANGES IN WING OUTER PANEL OF THE C-46 AIRPLANE RESULTING FROM OPERATION OF THERMAL ICE-PREVENTION SYSTEM. PART 1 - STATION 47.

Flight conditions / Thermocouples	Temperature °F					
	110 mph I.A.S. level flight		135 mph I.A.S. level flight		155 to 165 mph I.A.S. 2g bank	
	Heat off	Heat on	Heat off	Heat on	Heat off	Heat on
T1-1	45	46	46	47	46	50
T1-2	45	46	45	47	46	45
T2-1	46	49	46	50	46	48
T2-2	46	49	47	50	47	52
T3-1	47	53	47	54	47	56
T3-2	46	53	46	54	46	56
T4-1	47	53	47	54	47	57
T4-2	46	54	47	53	46	57
T6	51	129	51	120	48	132
T7	56	151	63	157	47	158
T8	54	124	60	124	50	132
T9	49	74	50	75	49	75
T10-1	50	66	50	67	50	76
T10-2	50	63	50	65	50	70
T11-1	51	64	51	66	51	77
T11-2	50	56	49	56	51	65
T12-1	49	53	49	55	51	60
T12-2	48	52	48	53	50	57
T13-1	47	49	48	51	54	55
T13-2	47	49	47	49	54	54
T14	46	48	46	48	47	51
T15	47	50	46	51	48	54
T16	47	50	48	52	48	55
T17	49	53	48	54	48	58
T18	48	55	48	56	48	61
T19	49	57	49	58	48	65
T20	49	68	49	67	47	75
T21	54	94	53	97	50	106
T22	51	69	51	70	50	80
T23	53	75	54	77	51	88
T24	53	73	53	76	52	86
T25	52	65	52	70	52	80
T26	52	59	53	63	53	71
T27	49	53	50	55	52	60
T28	48	53	49	55	53	59
T29	48	53	49	56	54	60
T30	47	50	49	53	54	58
T31	47	49	48	52	54	56

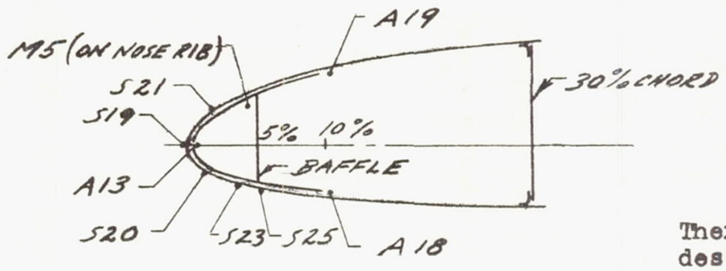
Note: 1 denotes thermocouple on inner surface of skin.
2 denotes thermocouple on outer surface of skin.

TABLE V.- TEMPERATURE CHANGES IN WING OUTER PANEL OF THE C-46 AIRPLANE RESULTING FROM OPERATION OF THERMAL ICE-PREVENTION SYSTEM. PART 2 - STATION 137.

Flight condition / Thermocouples	Temperature °F					
	110 mph I.A.S. level flight		135 mph I.A.S. level flight		155 to 165 mph I.A.S. 2g bank	
	Heat off	Heat on	Heat off	Heat on	Heat off	Heat on
T32-1	54	55	50	55	40	45
T32-2	55	56	50	55	41	46
T33-1	54	61	50	60	42	50
T33-2	54	61	50	60	43	52
T34-1	54	71	50	68	44	61
T34-2	54	68	50	68	43	59
T36	47	112	47	106	46	120
T38	51	157	51	152	45	163
T39	48	105	49	106	45	108
T40	47	89	48	93	47	98
T41-1	49	76	49	80	49	86
T41-2	48	74	48	76	48	80
T42-1	48	64	48	66	49	71
T42-2	48	60	48	63	49	76
T43-1	47	56	48	58	50	62
T43-2	46	54	47	56	50	60
T44-1	46	50	47	51	51	56
T44-2	46	50	47	51	52	55
T45	46	49	46	48	46	51
T46	46	53	46	54	47	57
T47	46	54	46	55	47	60
T48	48	63	47	64	48	70
T49	48	71	48	73	47	80
T50	48	97	48	95	47	105
T51	48	98	49	101	47	108
T52	50	83	49	86	48	95
T53	51	86	50	88	50	91
T54	50	77	49	81	51	84
T55	50	65	49	68	50	70
T56	48	58	48	60	50	61
T57	47	54	46	55	51	56
T58	47	50	46	51	51	53
T59	47	59	48	61	51	63
T60-1	47	90	47	85	46	86
T60-2	46	82	47	78	46	80

1 Denotes thermocouple on inner surface of skin,
2 Denotes thermocouple on outer surface of skin.

TABLE V.- TEMPERATURE CHANGES IN WING OUTER PANEL OF C-46 AIRPLANE RESULTING FROM OPERATION OF THERMAL ICE-PREVENTION SYSTEM. PART 3 - STATION 159

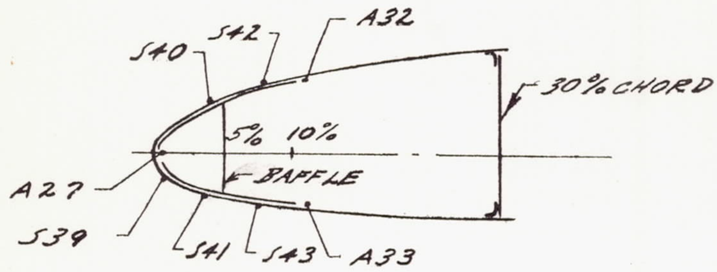


Thermocouple designation

A - air
S - skin
M - structure

Flight conditions / Thermo-couples	Temperature °F					
	110 mph I.A.S. level flight		135 mph I.A.S. level flight		155 to 165 mph I.A.S. 2g bank	
	Heat off	Heat on	Heat off	Heat on	Heat off	Heat on
S21	64	151	62	147	52	157
S19	59	152	58	147	47	158
S20	55	147	54	124	48	161
S23	54	150	52	148	49	164
S25	53	149	51	150	50	160
A19	52	125	54	134	50	151
A13	58	256	56	247	50	269
A18	51	131	50	134	50	142
M5	59	216	56	214	52	244

TABLE V.- TEMPERATURE CHANGES IN WING OUTER PANEL OF C-46 AIRPLANE RESULTING FROM OPERATION OF THERMAL ICE-PREVENTION SYSTEM. PART 4 - STATION 380



Thermocouple designation

A - air
S - skin

Flight conditions / Thermocouples	Temperature °F					
	110 mph I.A.S. level flight		135 mph I.A.S. level flight		155 to 165 mph I.A.S. 2g bank	
	Heat off	Heat on	Heat off	Heat on	Heat off	Heat on
S42	49	112	54	145	49	157
S40	51	137	57	164	50	176
S39	47	169	41	163	47	180
S41	47	161	49	157	48	167
S43	47	155	46	146	48	138
A32	47	149	47	152	48	160
A27	48	278	50	264	47	286
A33	47	180	46	174	48	179

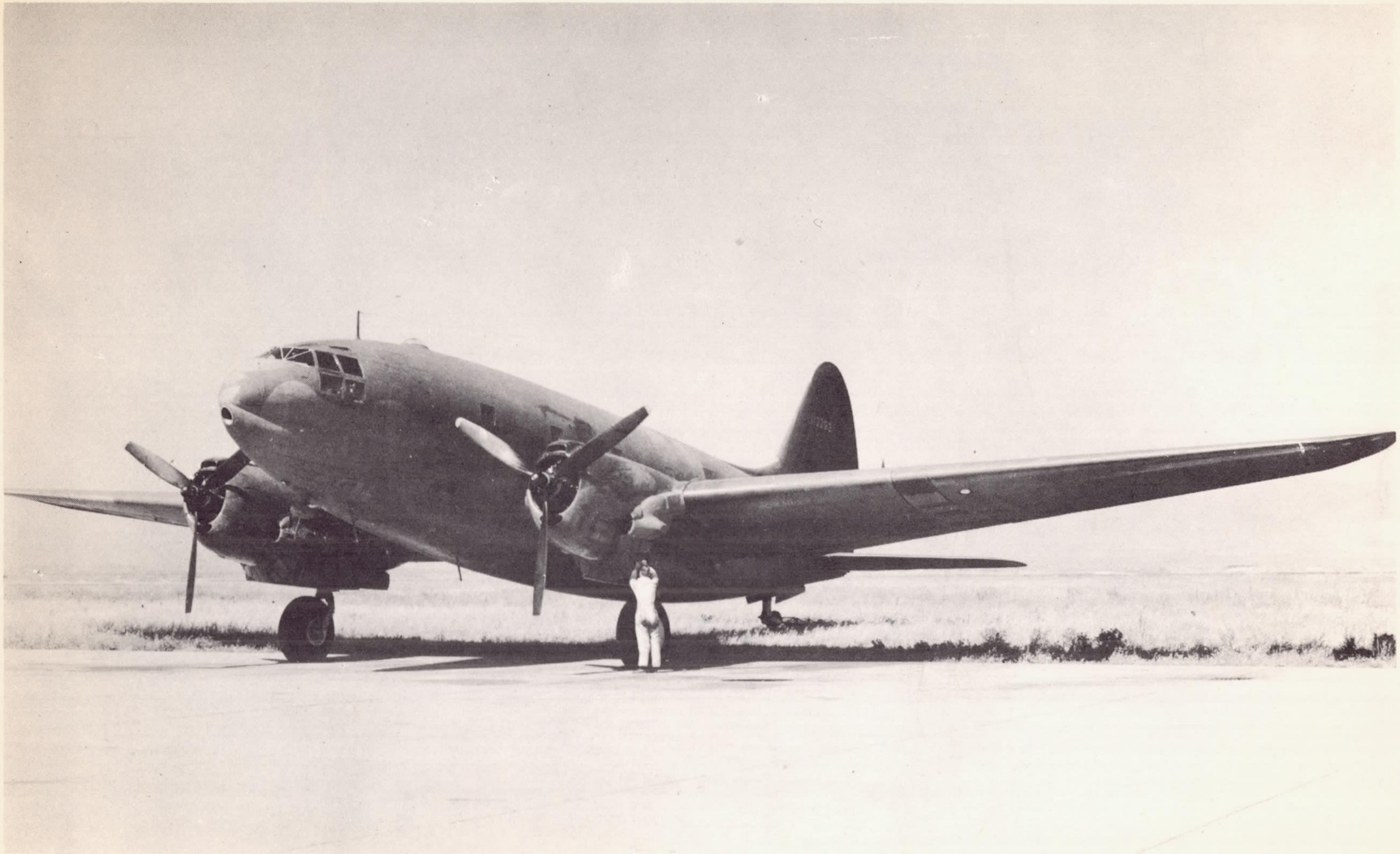


Figure 1.- The C-46 airplane equipped with a thermal ice-prevention system.

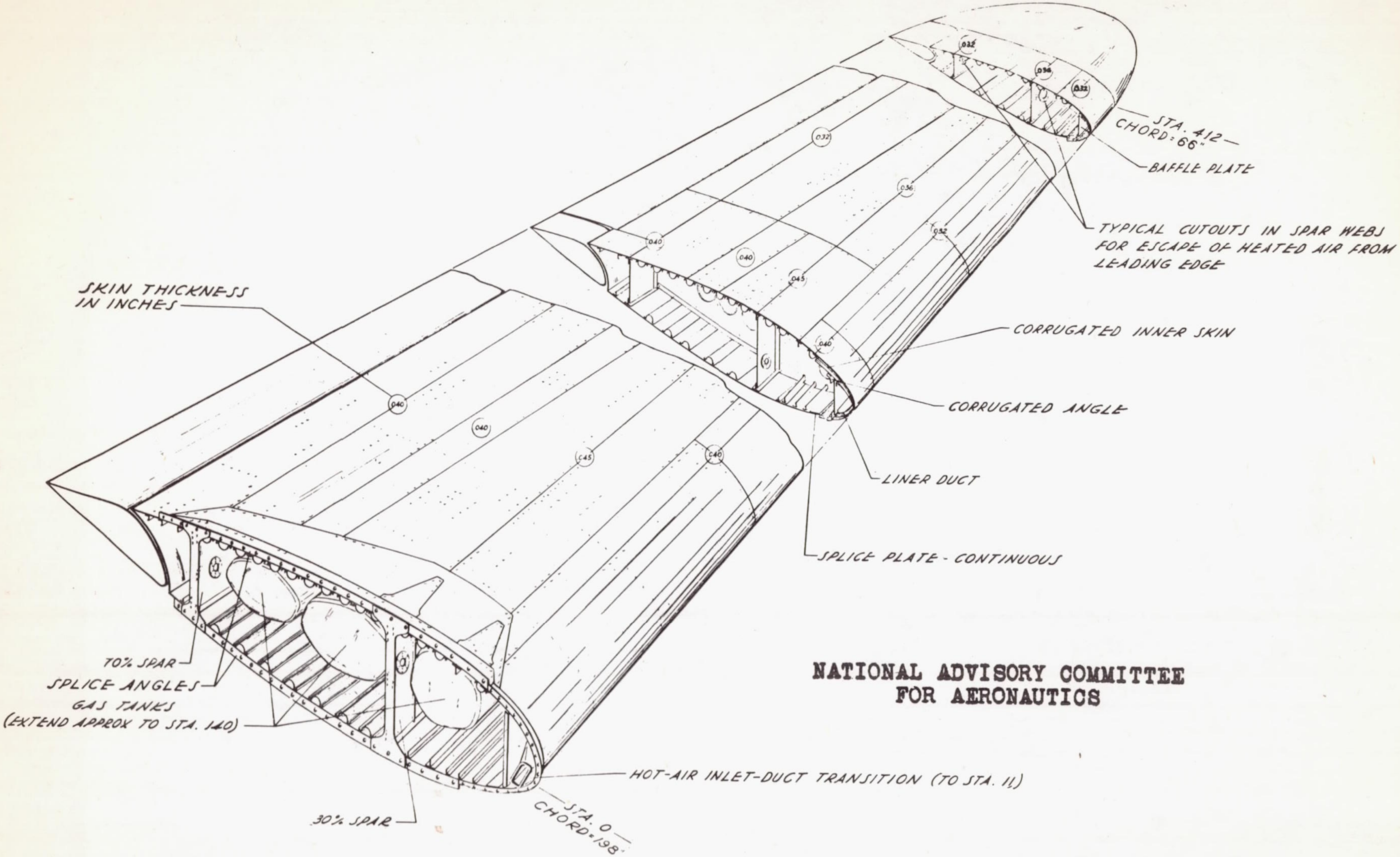


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Figure 2.- Structural details of the left-wing outer panel of the C-46 airplane with leading edge revised for thermal ice prevention.

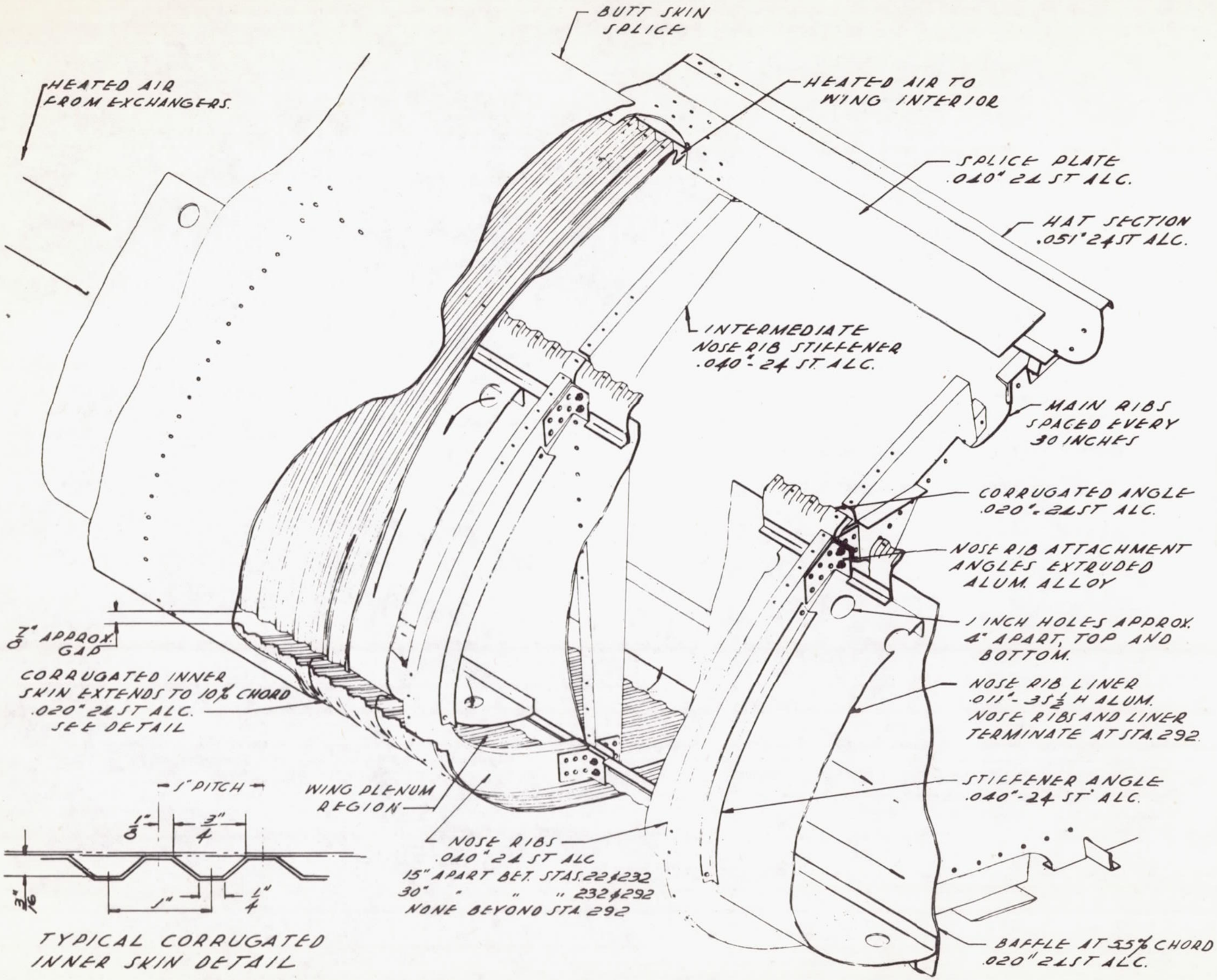

Figure 3.- Typical wing outer-panel leading-edge section as revised for thermal ice-prevention, C-46 airplane.

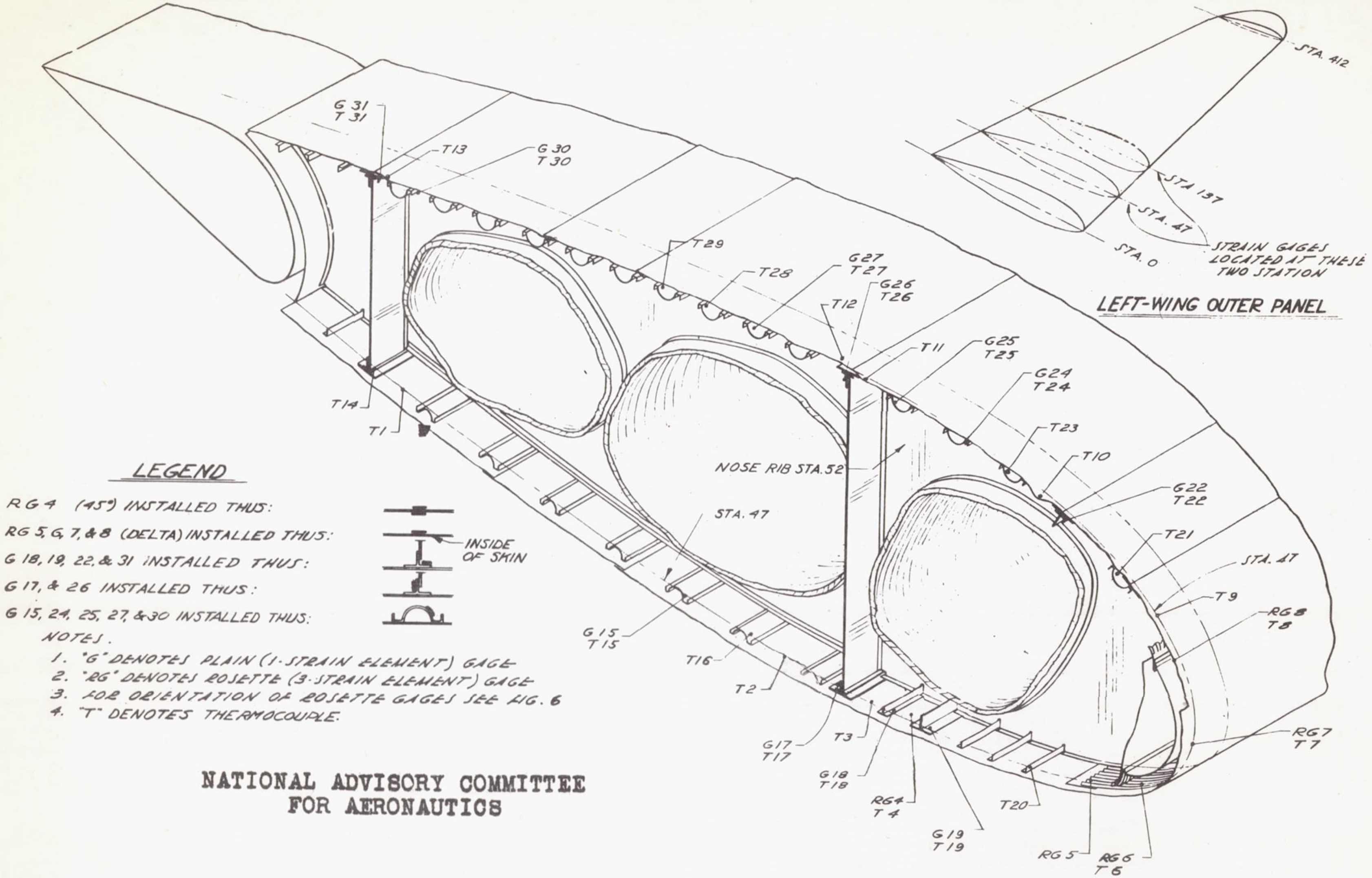

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Figure 4.- Location of strain gages and thermocouples at wing outer-panel station 47, C-46 airplane.

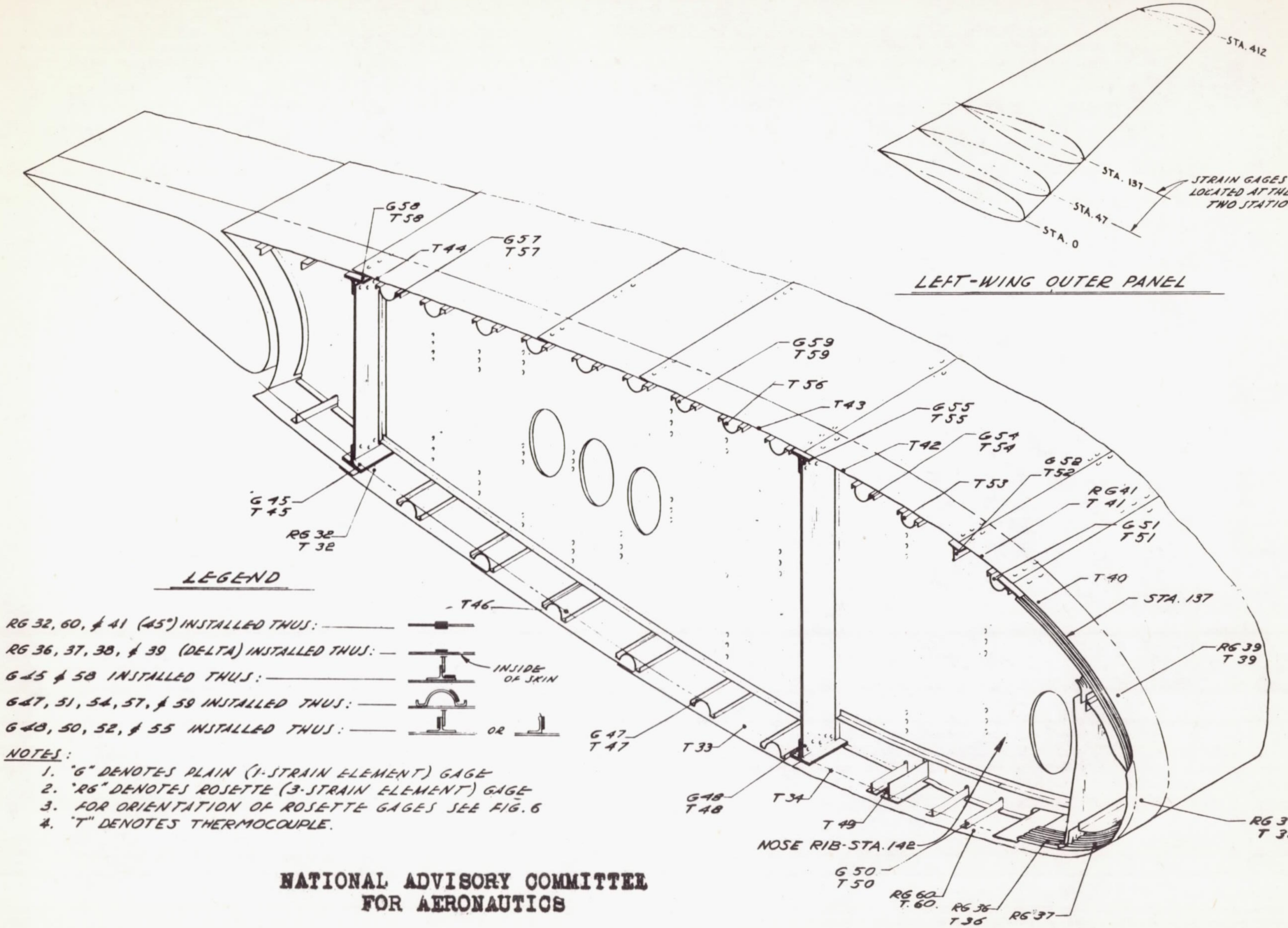

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Figure 5.- Location of strain gages and thermocouples at wing outer-panel station 137, C-46 airplane.

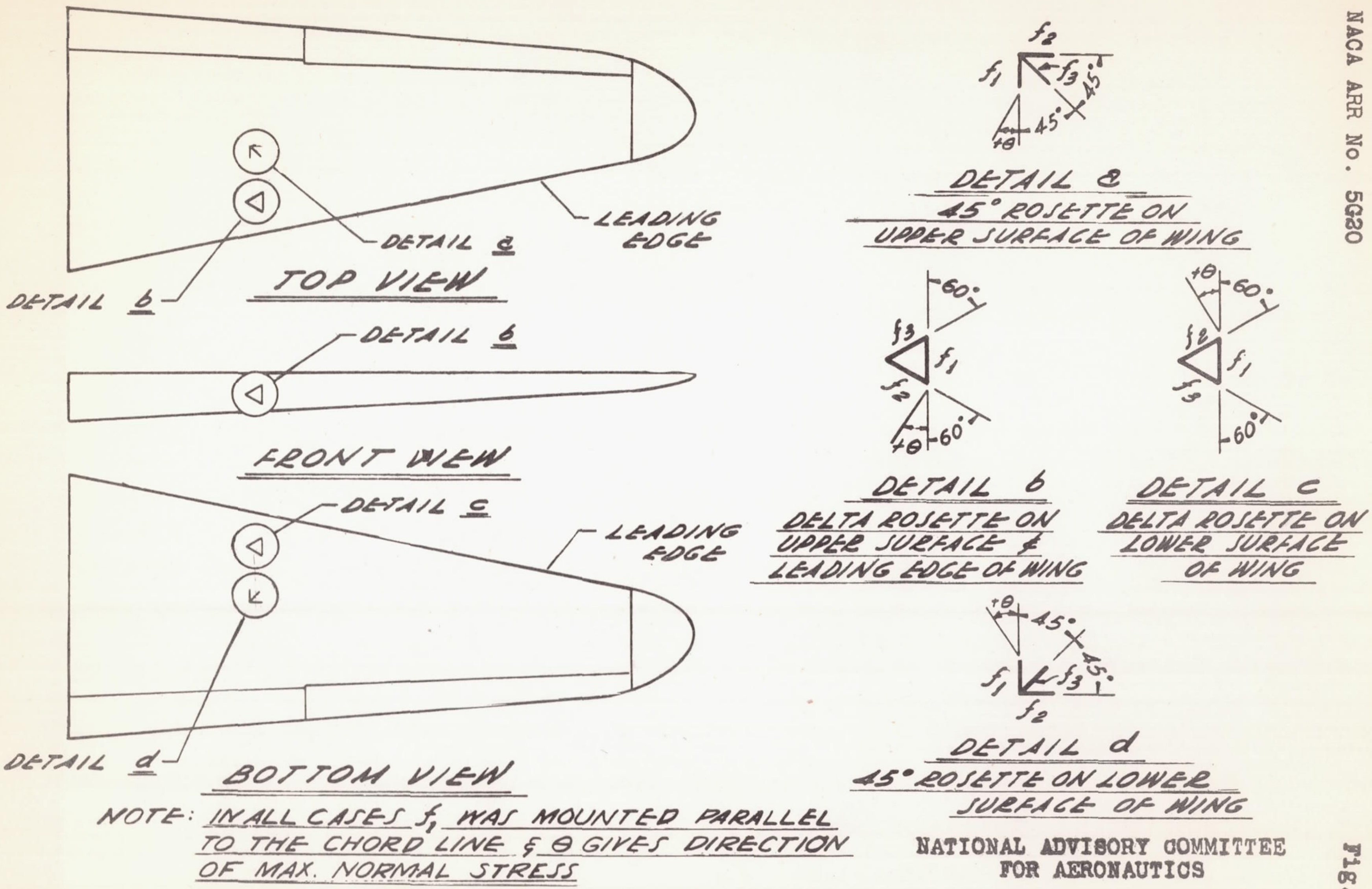

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Figure 6.- Orientation of rosette gages on the left-wing outer panel of the C-46 airplane.

A-48

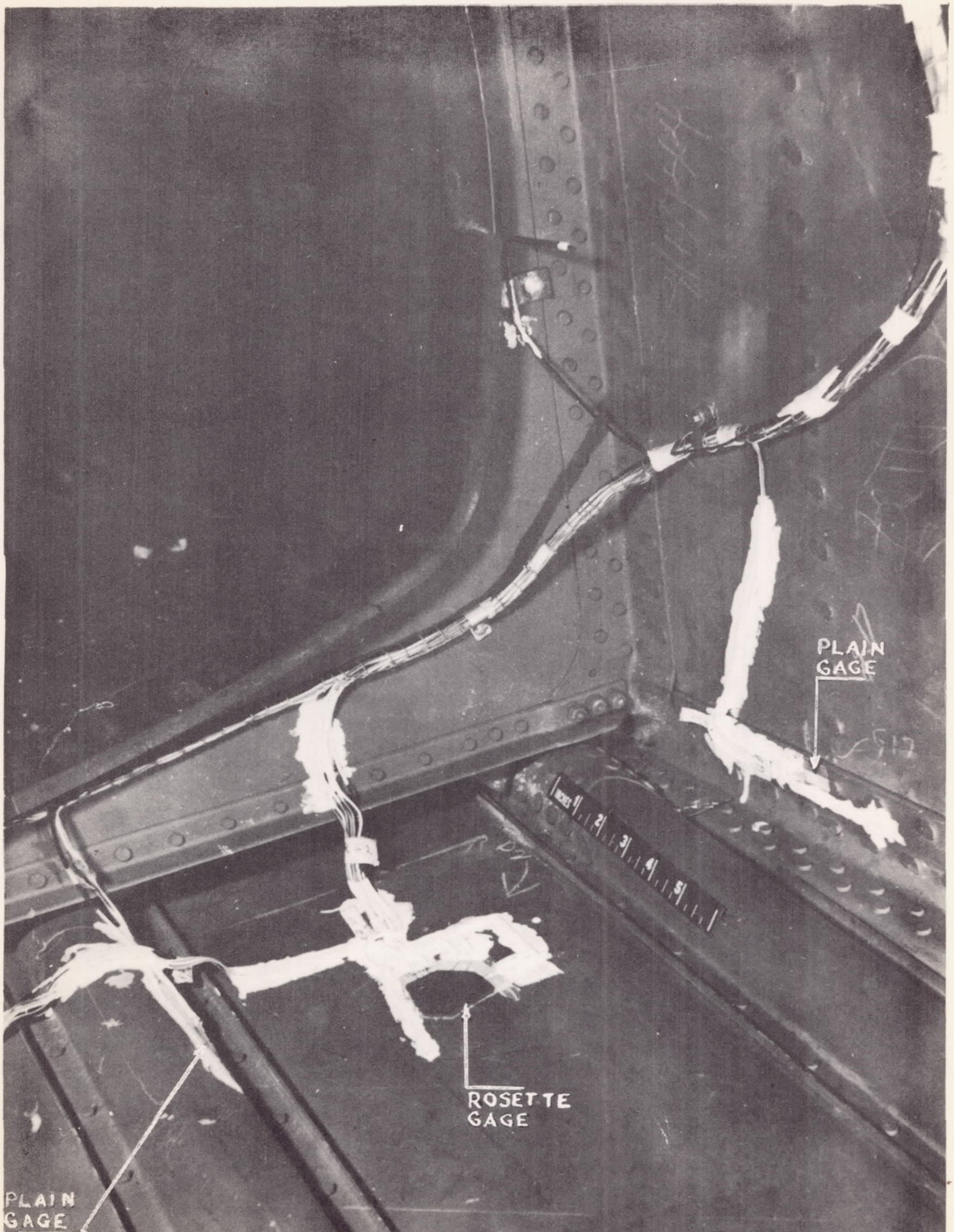

Figure 7.- Typical strain-gage and thermocouple installation in left-wing outer panel, C-46 airplane.

Figure 8.- Installation of instrumentation in airplane cabin for the wing thermal-stress investigation, C-46 airplane.

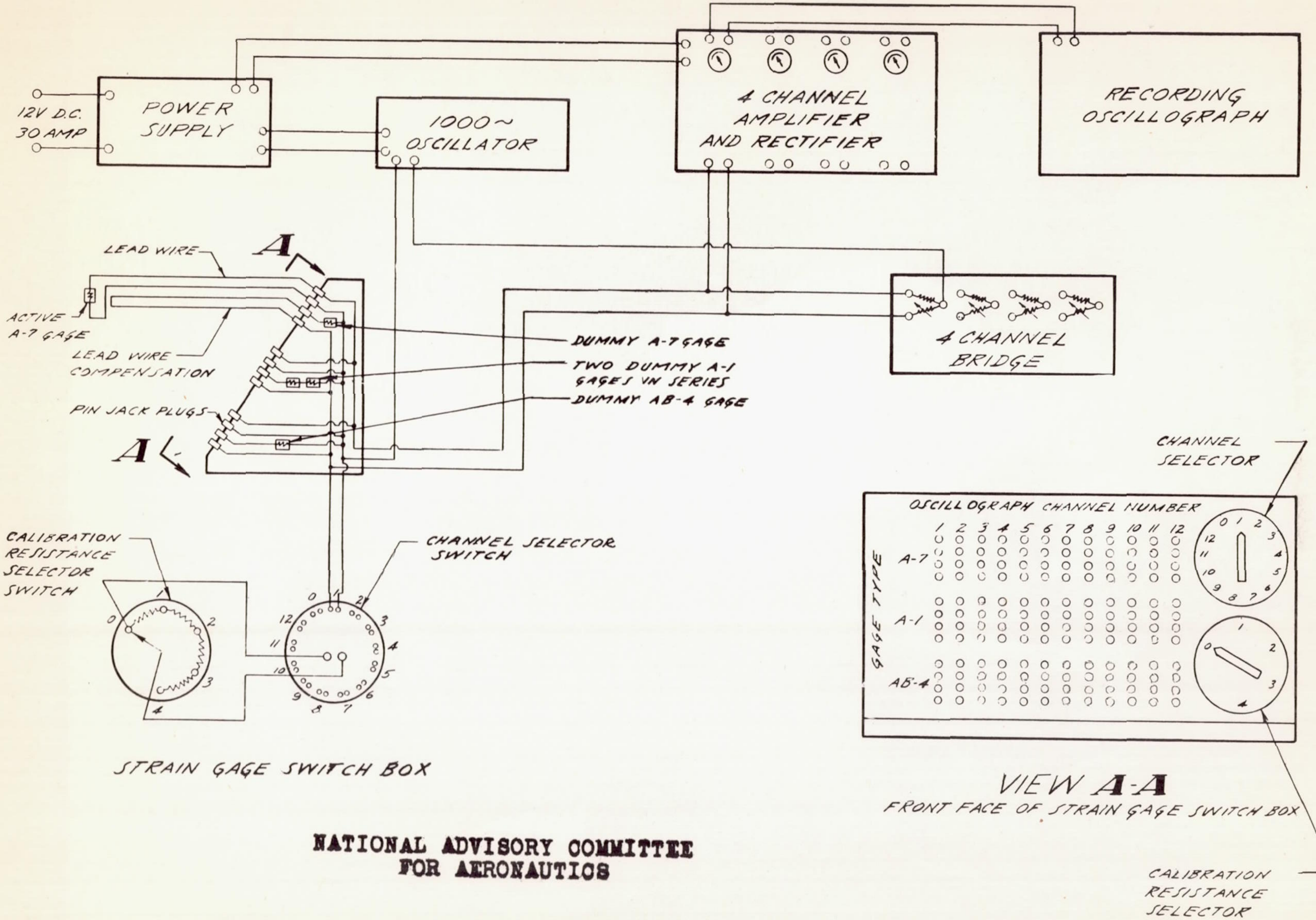

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Figure 9.- Schematic diagram of the strain-gage recording circuit. Complete circuit is given for recording A-7 gage readings on oscillograph channel number 1.

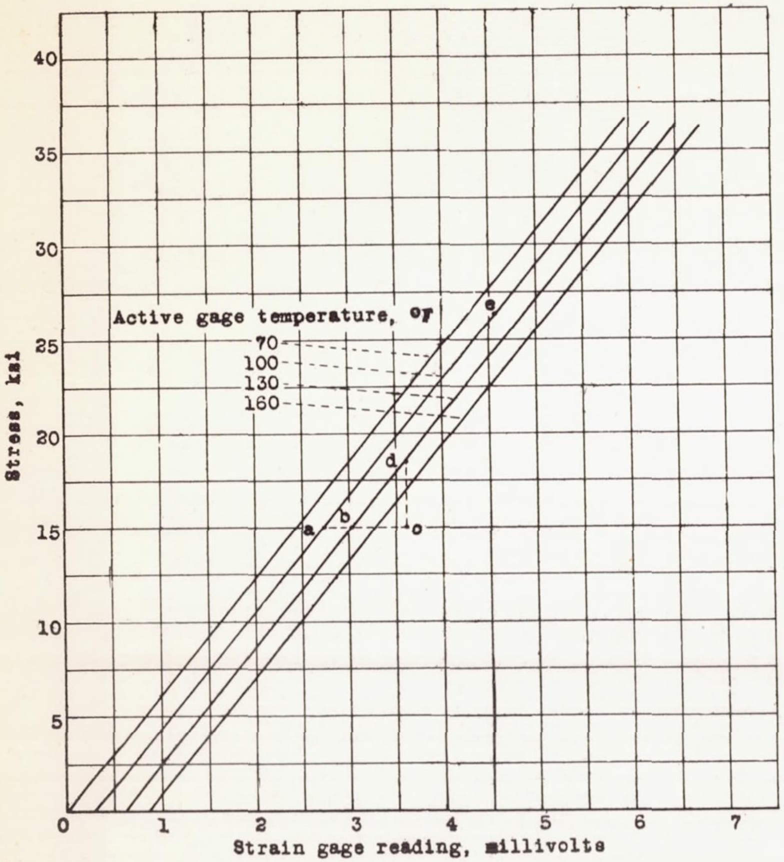
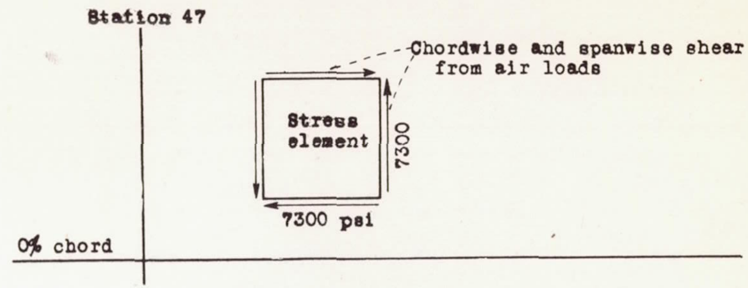
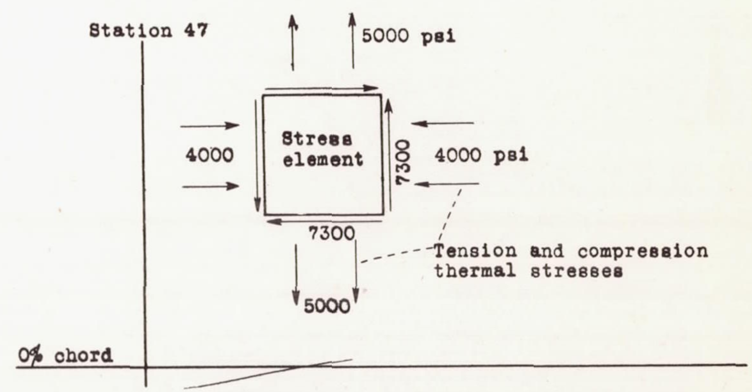


Figure 10.- Typical laboratory calibration of strain gage showing effect of temperature on gage reading. Dummy gage temperature constant at 70° F.



(a) Initial stress due to air loads as obtained from wing structural analysis.



(b) Combined stress resulting from air loads plus wing heating.

Figure 13.- Typical change in stress at the leading edge of the C-46 wing outer panel upon operation of the thermal ice-prevention system.

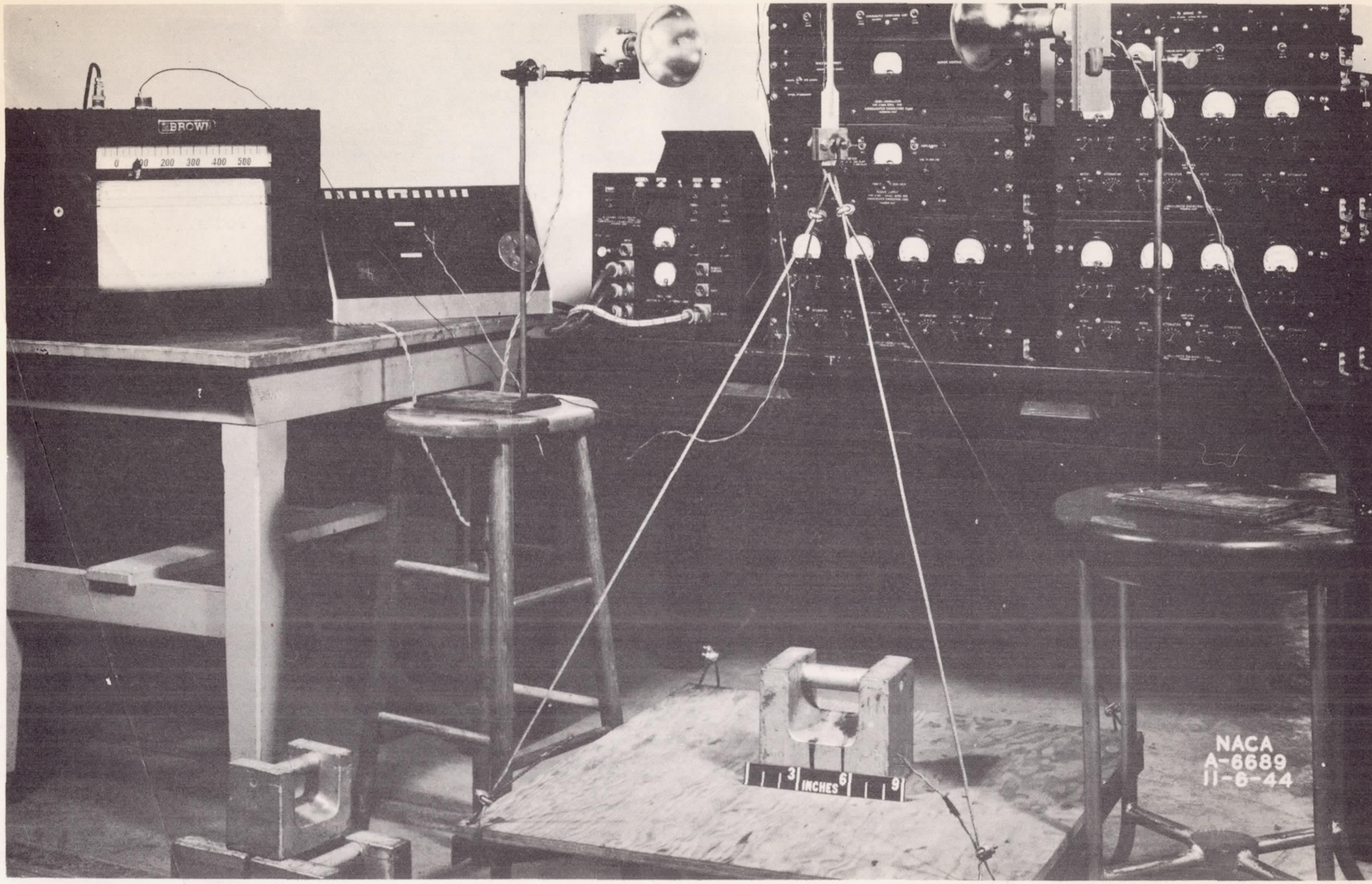
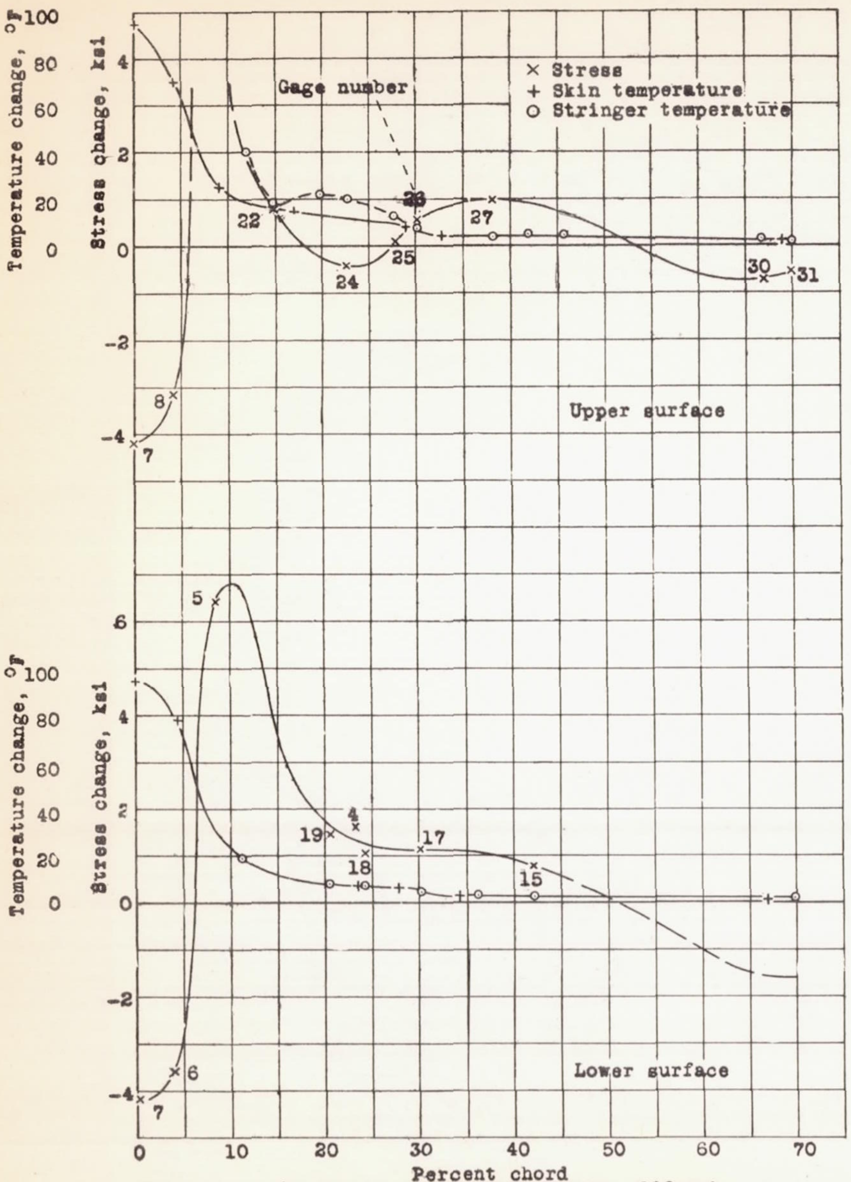


Figure 11.- Laboratory test installation used to determine the accuracy of stress-change-measurement equipment and flight-test procedure.



(a) Station 47, level flight, 110 mph

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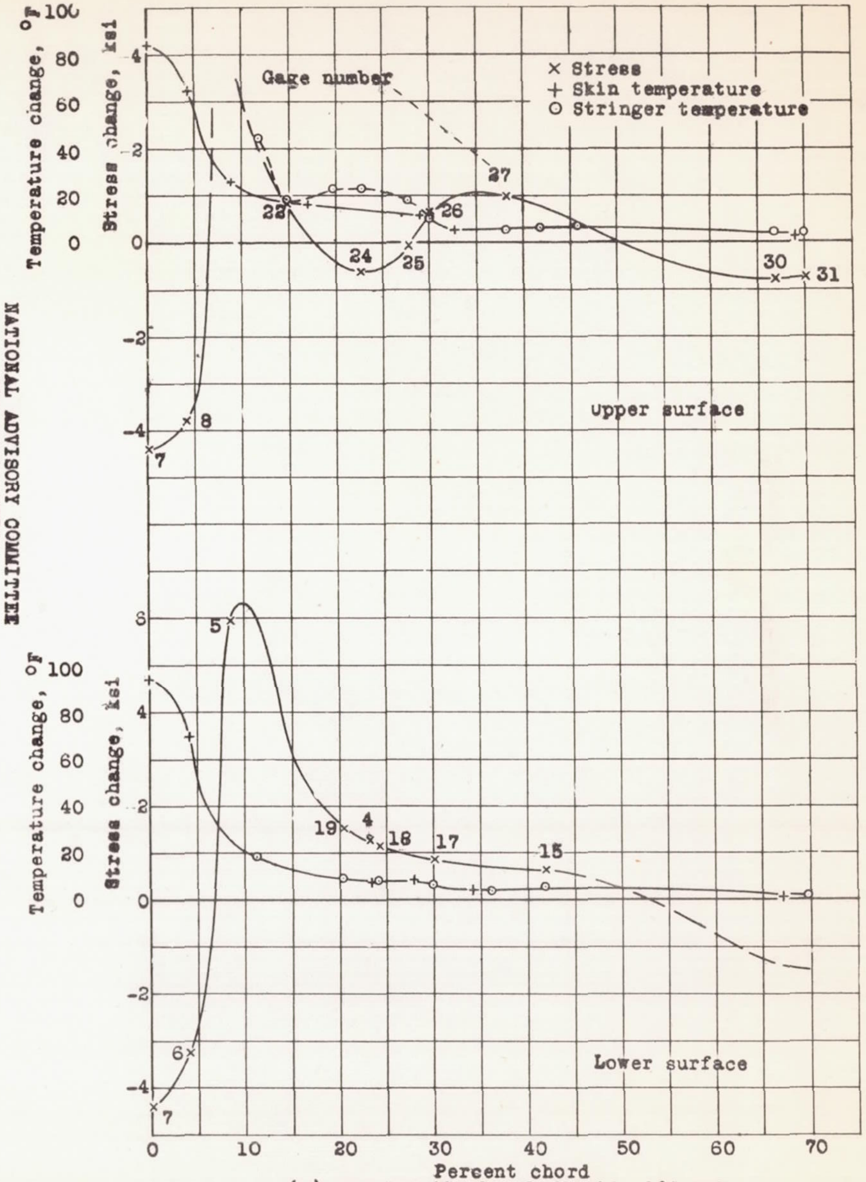
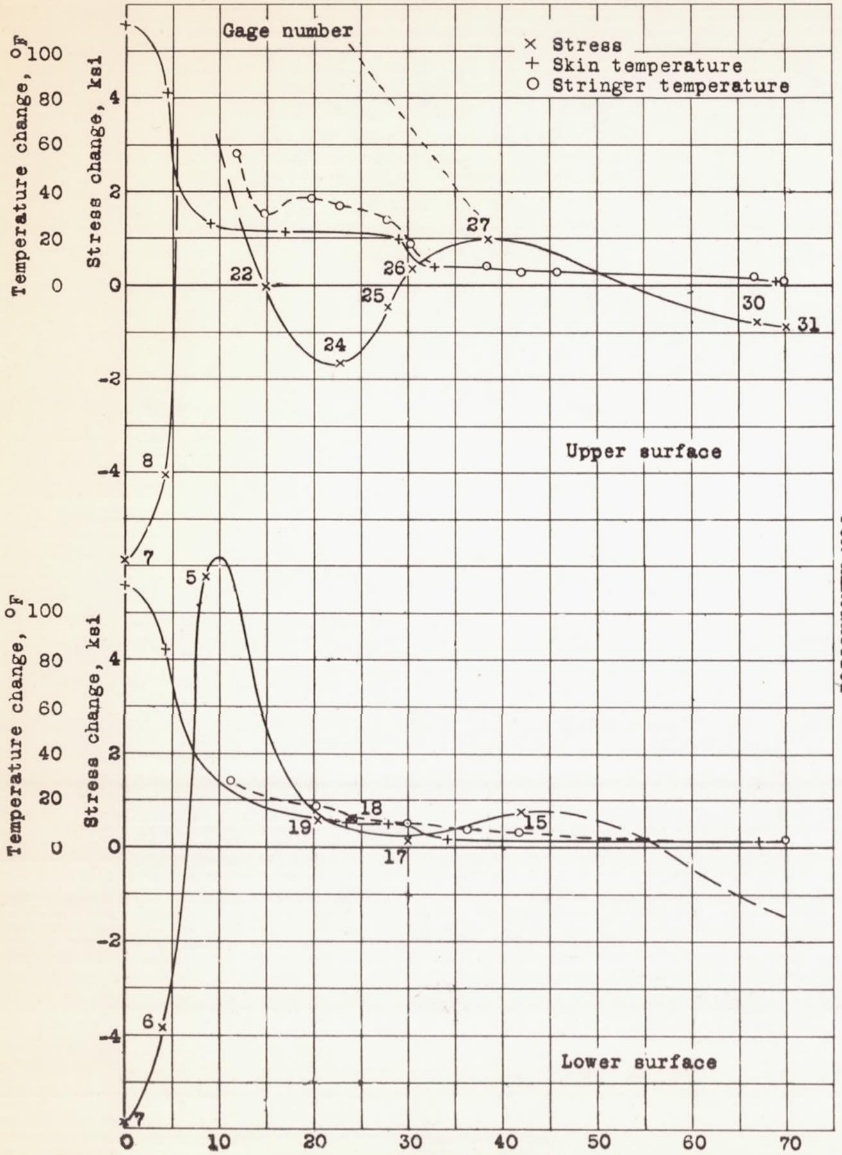
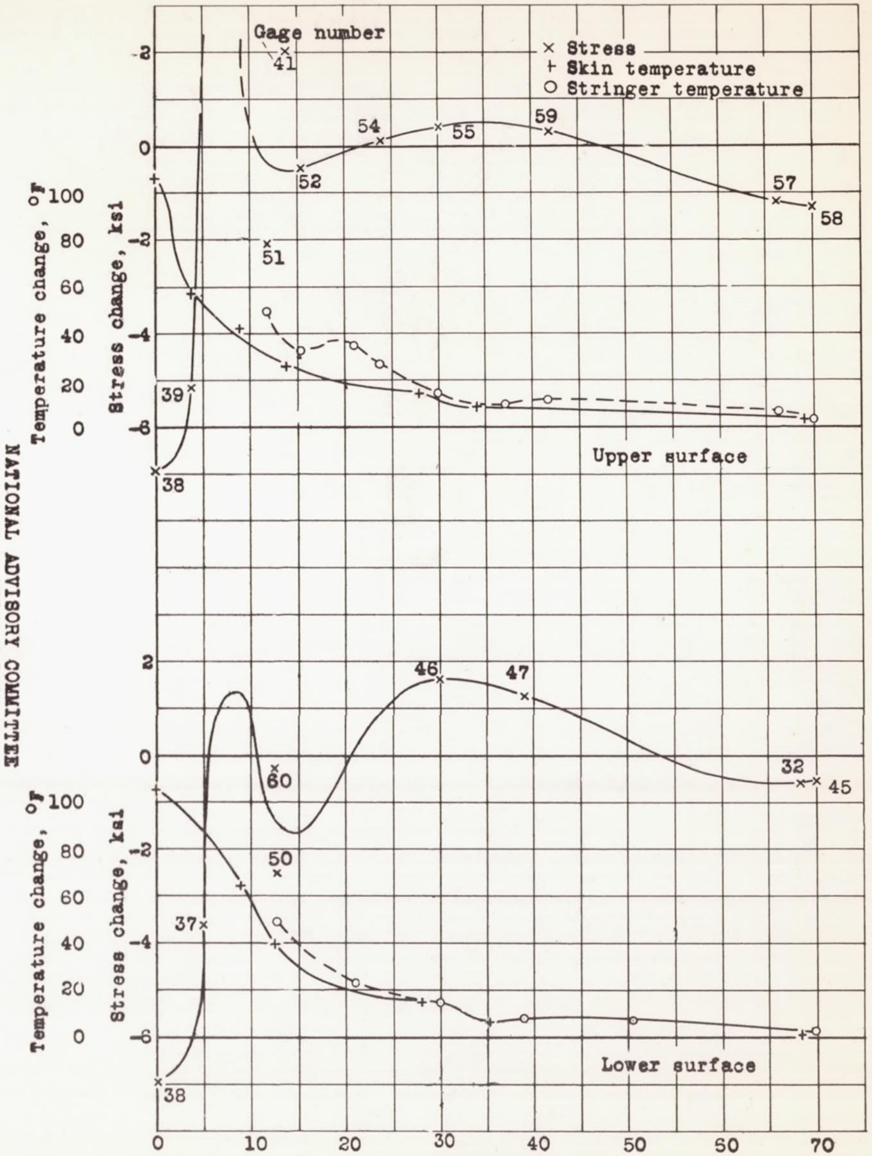


(b) Station 47, level flight, 135 mph

Figure 12 (a to h).- Chordwise distribution of stress change in C-46 wing resulting from operation of the thermal ice-prevention system.

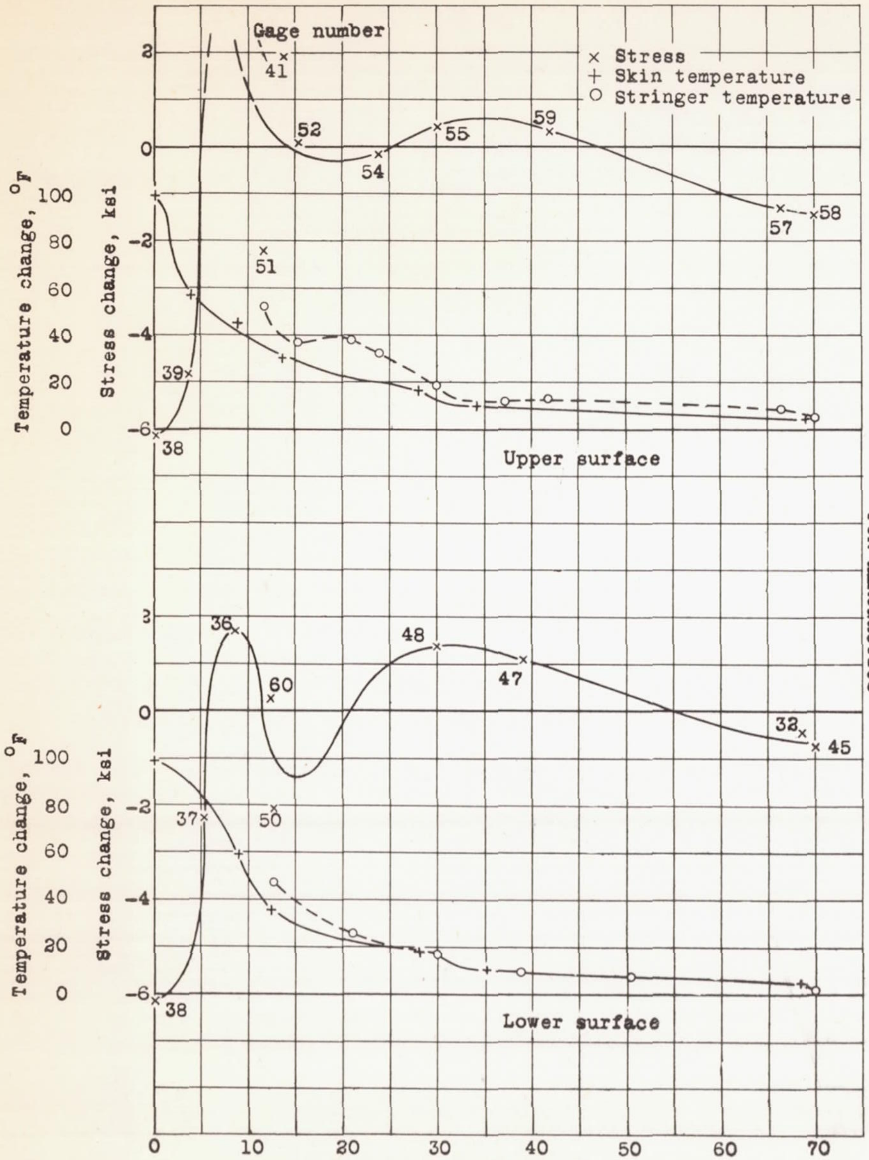


(c) Station 47, 2g bank, 155 to 165 mph

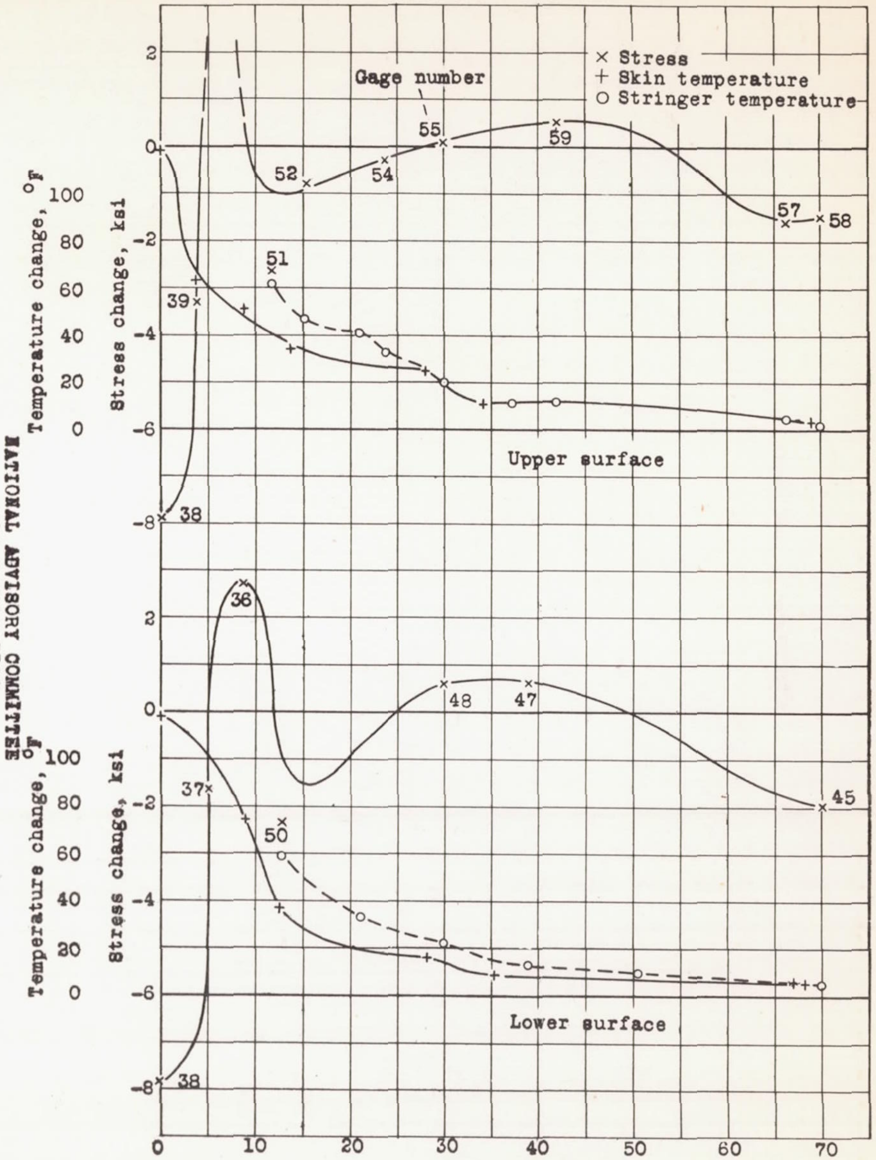


(d) Station 137, level flight, 110 mph

Figure 12.- (Continued)

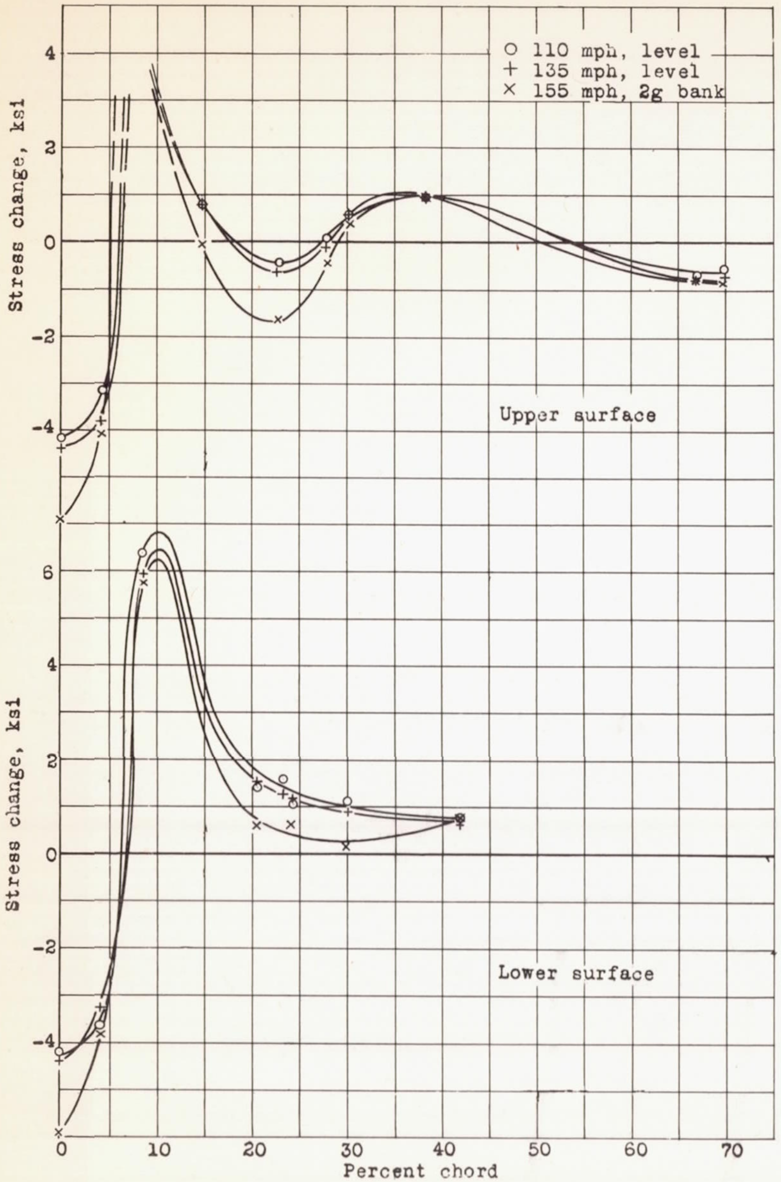


(e) Station 137, level flight, 135 mph

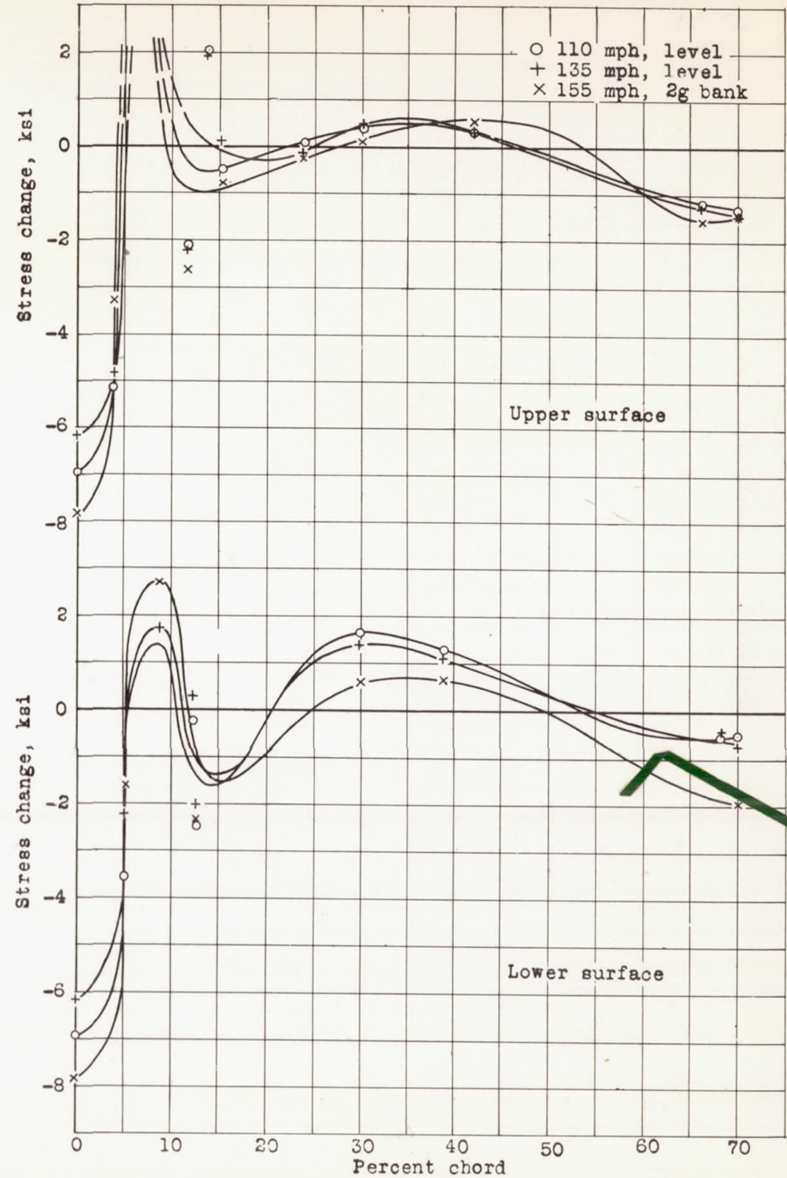


(f) Station 137, 2g bank, 155 to 165 mph

Figure 12.- (Continued)



(g) Comparison of three flight conditions, station 47



(h) Comparison of three flight conditions, station 137

Figure 12.- (Concluded)