June 15, 1942

GROUP NO. 1 REVISION SHEETS
CIVIL AERONAUTICS MANUAL
05-GLIDER AIRWORTHINESS
(Edition of June 1, 1940)

1. Attached hereto are revision sheets numbers 1-54, inclusive, which cover all technical amendments that have been made to date to the June 1, 1940 Edition (proposed) of Civil Aeronautics Manual 05, Glider Airworthiness. These sheets should be entered in your copy of CAM 05 opposite the page affected.

2. For your information, there is listed on the back of this sheet the subjects of all currently effective revision sheets to CAM 05 (proposed).

3. After the revision sheets have been inserted, a record of entry should be made as indicated below, and this sheet then entered preceding page (b) of CAM 05.

Entered by ______________

Date ______________

5-23020

CIVIL AERONAUTICS MANUAL
05-GLIDER AIRWORTHINESS
(Edition of June 1, 1940)

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F-33142

Revision to Page (b)

05.060 CHANGE, REPAIR OR ALTERATION OF CERTIFICATED GLIDERS.

This matter is covered by CAM 18.

Revise 05.123-C, a to read as follows:

a. The normal force coefficient C_N can be determined from Eq. 5, CAM 05.1C. The steps involved are shown as items 1 through 8 of Table .1-I.

05.211 DESIGN AIRSPEEDS

1. The values of the design airspeeds specified in CAR Table 05-2 are minimum values. In certain cases it may be desirable to use larger values; e.g., utility type gliders designed under Class III (see CAM 05.01) or high performance type gliders designed under Class II. (See CAM 05.70j regarding the possibility of accidentally exceeding the placard speed.) In order to provide for a high auto-winch tow placard speed, it may be advantageous to use a higher design gliding speed.

2. The k values specified in CAR Table 05-2 have been determined on the basis of studies of the "cleanness" of current gliders. The values of k for Class I gliders have been set approximately 11% higher than the values of k for Class II gliders since blind flying in Class I gliders is permitted and higher speeds are apt to be encountered in recovery from inadvertent attitudes (which may be obtained in blind flying) than in normal gliding operations under good conditions. Since these constants have been established on a simplified basis it is possible that they may lead to irrational values of V_g when applied to particular cases. In any case it will be unnecessary to design Class I gliders to a V_g greater than .4 times the terminal velocity or to design Class II gliders to a V_g greater than .36 times the terminal velocity. In cases when the value of V_g is based on terminal velocity, in accordance with the above, calculations substantiating the value of terminal velocity should be submitted.

05.2130 BASIC FLIGHT ENVELOPE

1. In so far as CAR 05.2130 is concerned, the basic flight envelope or V-n diagram is a locus of points representing the limit wing load factors and the corresponding velocities for the design criteria specified in CAR 05.2131 through 05.2133.

2. A sample basic flight envelope is shown in Figure .2-2. This envelope is contructed for a Class I sailplane of very clean design, having a full cantilever wing. The basic design features are as follows:

$W/S = s = 3.5$ lbs. per sq. foot.
R (Aspect ratio) = 12
m (corrected to R of 12) = 4.8 C_L per radian
e (weight of wing) = 1.5 lbs. per sq. foot.

In accordance with CAR 05 Table 05-2, a value of k = 61 should be used so that the minimum design gliding speed which could be used would be $V_{g\,min.} = 61\sqrt{s} = 114$ mph. The corresponding placard "Never exceed" speed would be .9 x 114 = 103 mph. In this particular case, however, it is assumed by the designer that a somewhat higher placard "never exceed" speed is desirable so a design V_g of 125 mph is selected (corresponding placard "never exceed" speed is .9 x 125 = 112 mph). The construction of the basic flight envelope for this sailplane will be outlined in the following sections.

05.2131

a. Plot the following equation to obtain line 1 of the positive portion of the V-n diagram. (See Figure .2-2)

$$n = \frac{V^2}{196s}$$ where n = maximum possible positive limit wing load factor at the speed, V (mph). This is based on a C_L max. (dynamic) of 2.0.

b. Draw a vertical line through the velocity corresponding to V_g (line 2 of Figure .2-2).

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05.2131 c. Plot the following equation to obtain line 3 of the negative portion of the V-n diagram (See Figure .2-2)

$$n = -\frac{V^2}{391s}$$ where n = maximum possible negative unit wing load factor at the speed V (mph). This is based on a C_L max. (dynamic) of 1.0 (negative).

05.2132 When applying the requirements specified in CAR 05.2132, the following procedure should be followed:

Draw a horizontal line (Line 4 of Figure .2-2) through the greatest value of n specified in Items 5, 6, and 7 of CAR Table 05-2. This will intersect line 1, drawn in accordance with CAM 05.2131 a, at point C and will intersect line 2, drawn in accordance with CAM 05.2131 b, at point E.

Example:

Item 5 The specified maneuver load factor = 5.33

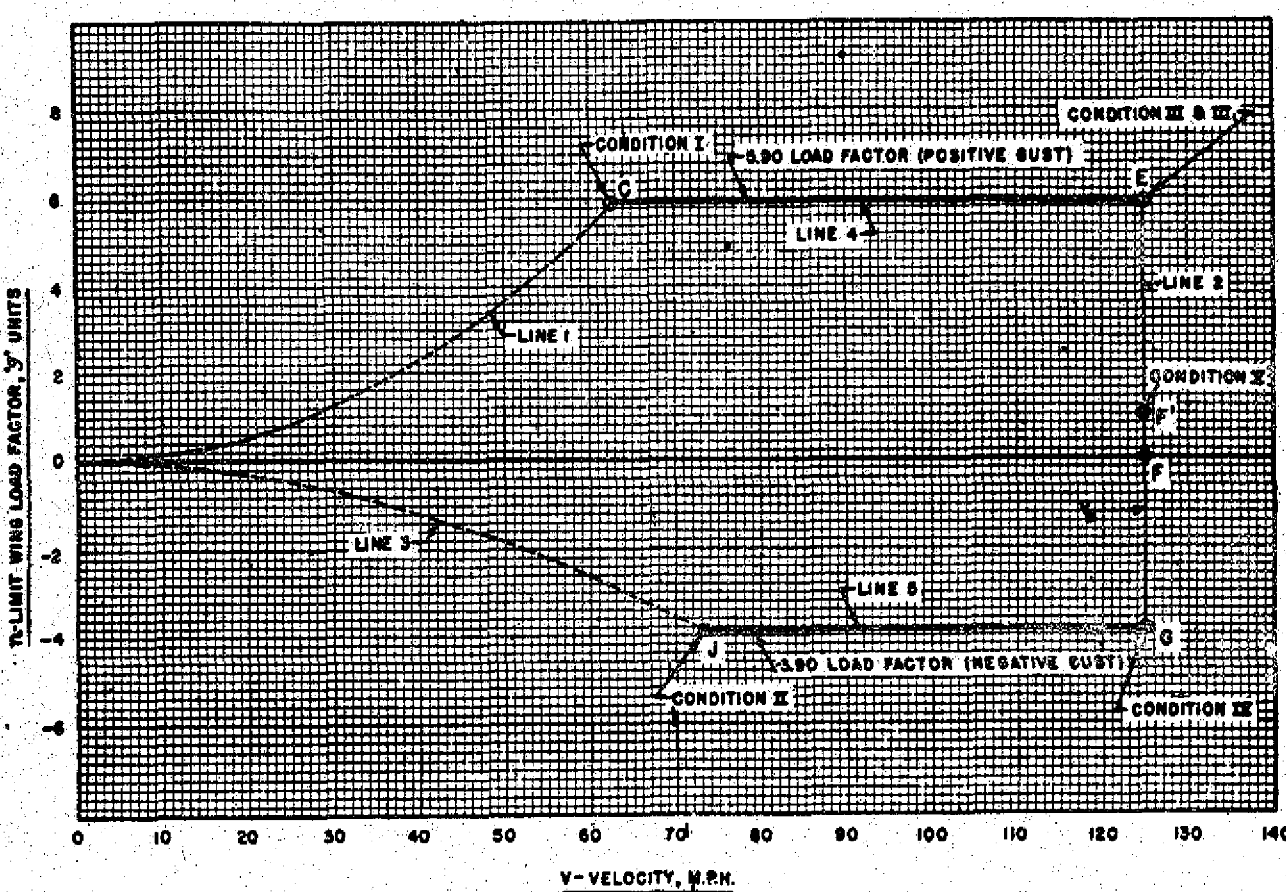
Item 6 The gust relieving factor from CAM Figure .2-1 is .685 for s = 3.5 so for this example,

$$n = 1 + \frac{.685 \times 24 \times 125 \times 4.8}{575 \times 3.5} = 1 + 4.90 = \underline{5.90}$$

Item 7 $V_{taw} = 35\sqrt{3.5} = 65.5$ mph.

$$n_{taw} = \frac{1}{(3.5-1.5)}\left[\left(\frac{65.5^2}{391}\right) - 1.5\right] = \underline{4.75}$$

The greatest positive load factor of the three values above is 5.90 and it therefore determines line 4 of the basic flight envelope of Figure .2-2. It should be noted that the positive portion of the basic flight envelope is represented by the figure OCEF of Figure .2-2.



SAMPLE BASIC FLIGHT ENVELOPE SHOWING CRITICAL BASIC FLIGHT CONDITIONS

(REF. CAAM 05.2130)

FIG. 2-2

REVISION TO PAGE 2-4

F-2009

05.2133 When applying the requirements specified in CAR 05.2133, the following procedure should be followed:

Draw a horizontal straight line (Line 5 of Figure .2-2) through the greatest negative value of n specified in items 8 and 9 of CAR Table 05-2. This will intersect line 3, drawn in accordance with CAM 05.2131c, at point J and will intersect line 2 at point G.

Example:

Item 8 The specified maneuver load factor = - 2.67.
Item 9 The negative load factor due to a down gust is as follows:

$$n = 1 - \frac{.685 x 24 x 125 x 4.8}{575 x 3.5} = 1 - 4.90 = -3.90$$

The greater negative load factor of the above is -3.90 so it determines line 5 of the basic flight envelope of Figure .2-2. It should be noted that the negative portion of the basic flight envelope is represented by the figure FGJO of Figure .2-2.

05.2134 In general, an investigation of the following specific basic flight conditions, which correspond to points on the basic flight envelope (See Figure .2-2), will insure satisfactory coverage of the critical loading conditions.

a. Condition I (Positive High Angle of Attack). This condition corresponds to point C on the basic flight envelope. The aerodynamic characteristics C_N, CP (or C_M), and C_c to be used in the investigation should be determined as follows:

(Continued on revision Sheet No. 8)

Figure .2-3

Delete this figure.

(1) $C_{N_I} = \frac{n_I s}{q_I}$ (usually equals approximately 2.0)

where n_I = wing load factor corresponding to point C

q_I = dynamic pressure corresponding to the velocity V_I, which in turn corresponds to point C

(2) C_c = value corresponding to C_{N_I}, as obtained from the airfoil characteristics curves (See Figure .1-5)

(3) C.P. or C_M = value corresponding to C_{N_I}, as determined from the airfoil characteristics curves.

b. Condition II (Negative High Angle of Attack). This condition corresponds to point J on the basic flight envelope. The aerodynamic characteristics to be used in the investigation should be determined as follows:

(1) $C_{N_{II}} = \frac{n_{II} s}{q_{II}}$ (usually equals approximately -1.0)

(2) C_c = value corresponding to $C_{N_{II}}$ (may be assumed equal to zero if positive)

(3) CP or C_M = value corresponding to $C_{N_{II}}$

c. Condition III (Positive Low Angle of Attack). This condition corresponds to point E on the basic flight envelope. The aerodynamic characteristics should be determined as follows:

(1) $C_{N_{III}} = \frac{n_{III} s}{q_{III}}$ $(q_{III} = q_g)$

(2) C_c = value corresponding to $C_{N_{III}}$ (may be assumed equal to zero if positive)

(3) C.P. or C_M = value corresponding to $C_{N_{III}}$

d. Condition III_I (Modified Positive Low Angle of Attack). In order to cover the effects of limited use of the ailerons at V_g on the wings and wing bracing, such structure should be investigated for the following:

(1) $C_{N_{III_I}} = C_{N_{III}}$

(2) C_c = value corresponding to $C_{N_{III_I}}$

(3) C_M' = value obtained from Figure .2-4, where C_M is the value corresponding to $C_{N_{III}}$ C_M' need only be applied to that portion of the span incorporating ailerons, using the basic value of C_M determined in Condition III over the remainder of the span.

e. Condition IV (Negative Low Angle of Attack). This condition corresponds to point G on the basic flight envelope. The aerodynamic characteristics should be determined as follows:

(1) $C_{N_{IV}} = \frac{n_{IV}S}{q_{IV}}$

(2) C_c = value corresponding to $C_{N_{IV}}$ (may be assumed equal to zero if positive).

(3) C.P. or C_M = value corresponding to $C_{N_{IV}}$

f. Condition V (Gliding). This condition corresponds to point F' on the basic flight envelope and represents the flight condition where the maximum rearward acting chord load occurs. This condition will only be critical for wing and wing bracing. The aerodynamic characteristics to be used in the investigation should be determined as follows:

(1) C_{N_V} = value corresponding to C_c max. (positive)

(2) C_c' = C_c max. (positive) + 0.01.

(3) C.P. or C_M = value corresponding to C_{N_V}

REVISION SHEET NO. 9

On Figure .2-4 delete:

(Ref. CAAM 05.2134e)

and add:

(Ref. CAM 05.2134d)

05.2140-A, 1.

1. For internally braced wings, the effects of trailing edge flaps on the wing structure as a whole can, in general, be accounted for satisfactorily by modifying, when necessary, the basic flight conditions in the following manner:

   a. The average value of C_M' used in design conditions III and IV (CAM 05.2134 c and e) should equal or exceed the quantity

   $$C_{M_f} \times \left(\frac{V_f}{V_g}\right)^2$$

   where: C_{M_f} is the average moment coefficient about the aerodynamic center (or at zero lift) for the airfoil section with flap completely extended. (The average moment coefficient refers to a weighted average over the span when C_M is variable. The wing area affected should be used in the weighting).

   V_f is the design speed with flaps extended, as specified in CAR 05.110.

   V_g is the design gliding speed used in conditions III and IV.

   b. The average value of C_c' used in design condition V (CAM 05.2134f) should equal or exceed the quantity

   $$C_{c_f} \times \left(\frac{V_f}{V_g}\right)^2$$

   where: C_{c_f} is the maximum positive chord force coefficient (average) for the airfoil section with flap completely extended. (The average chord force coefficient refers to a weighted average over the span).

When the above provisions are made, no balancing computations for the extended flap conditions need be submitted; hence, these conditions can also be eliminated from the design of the horizontal tail surfaces.

05.2150 In special cases where an investigation for the effects of unsymmetrical flight loads is required, the following assumptions should be made:

a. Modify Conditions I and III (see CAM 05.2134-a and -c) and the most critical negative condition by assuming 100 per cent of the air load to be acting on one side of the glider and 40 per cent on the other.

b. Assume the moment of inertia of the entire glider is effective.

It will usually be convenient to separate the effects of the loads due to linear accelerations from the loads due to torque T. It may be assumed that the stresses due to unsymmetrical loads can be obtained by adding algebraically the stresses due to 70 per cent of the normal (unmodified) loading to those determined by considering 30 per cent of the normal total load to be acting upward on one wing panel and 30 per cent to be acting downward on the other. The unbalanced moment or torque T is equal to 60 per cent of the normal total load on one wing panel times the distance from the longitudinal axis to the centroid of the load normally acting on the panel. This is illustrated in Fig. .2-5.

The angular acceleration α resulting from the torque T may be obtained from the following formula:

$$\alpha = \frac{T}{I_x} \quad (\text{rad./sec.}^2)$$

where I_x = the moment of inertia of the glider (and its contents) about the X, or longitudinal axis.
(I_x may be computed in accordance with the procedure contained in NACA Technical Note No. 575.) - I_x in mass units.

The torque T_n resisted by any portion of the glider may be obtained from the following expression, assuming the angular acceleration to be constant for all parts of the glider:

$$T_n = I_n \alpha \quad (\text{ft. lbs.})$$

when $I_n = M_n d_n^2$

where M_n = the mass of part n.
d_n = the distance from the longitudinal axis to the C.G. of part n in feet

05.217A-1 For wings having mean taper ratios (see c below) equal to or greater than .33, the span distribution should be determined as follows:

a. If the wing does not have aerodynamic twist (i.e., if the zero lift lines of all sections are parallel), the span distribution of normal force coefficient (C_N) should be assumed to vary in accordance with CAM 05 Figs. .2-6 and .2-7, which are assumed to represent two extreme cases of tip loading. Each case should be investigated, unless it is demonstrated that only one is critical. As an alternative method, it will be acceptable to investigate each design condition for only one span distribution using a rational distribution, except in the case of the high-angle-of-attack condition which gives the maximum forward chord loads (Condition I). For this condition, the analysis should be made for both the rational distribution and that given in Fig. .2-6.

b. If the wing has aerodynamic twist, the span distribution should be determined by the alternative method given in (a) above.

c. For these purposes, the mean taper ratio is defined as the ratio of the tip chord (obtained by extending the leading and trailing edges to the extreme wing tip) to the root chord (obtained by extending the leading and trailing edges to the plane of symmetry).

d. Acceptable methods of determining a rational span distribution are given in Army-Navy-Civil publication ANC1(1), "Spanwise Air Load Distribution" (obtainable from the Superintendent of Documents, Washington, D. C., at the nominal sum of 60 cents), in N.A.C.A. Technical Report No. 572, in N.A.C.A. Technical Report No. 585, in N.A.C.A. Technical Note No. 606, and in Appendix I, herein.

05.217A-2 For all wings having mean taper ratios less than .33, the span distribution should be determined by rational methods, unless it is shown that a more severe distribution has been used.

05.2160 GUST AT REDUCED WEIGHT

(This paragraph deleted)

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The note below the figure on this page should read as follows:

"NOTE: Above figures apply only to wings which have mean taper ratios greater than .33 and without aerodynamic twist."

Revision to Page .2-15

05.217-B 1

1. The approximate method of chord loadings outlined in CAM 05.34310 for the testing of wing ribs is suitable for conventional two spar construction if the rib forms a complete truss between the leading and trailing edges. An investigation of the actual chord loading should be made in the case of stressed-skin wings if the longitudinal stiffeners are used to support direct air loads. In some cases it is necessary to determine the actual distribution, not only for total load but for each surface of the wing. If wind tunnel data are not available, the methods outlined in N.A.C.A. Reports Nos. 383, 411, 456, 631 and 634 are suitable for this purpose. These methods consist in determining the "basic" pressure distribution curve at the "ideal" angle of attack and the "additional" pressure distribution curve for the additional angle of attack. These curves can be coordinated with certain values of C_L, so that the final pressure distribution curve can be obtained immediately for any C_L. Curves of this nature for several widely-used airfoils can be obtained directly from the N.A.C.A.

g. Fig. .2-9(b) is now plotted, using as ordinates the values of R_b C_x obtained from item 7 of Table .2-I. The area under curve .2-9(b) divided by the area under curve .2-8(d) gives the distance of the mean aerodynamic center from the base line 0-E in Fig. .2-8(a). This distance is indicated as $\bar{X}$ on that figure.

Revision to Page .2-19

05.218 B - 3

3. A tabular form will simplify the computations required to obtain the balancing loads for various flight conditions. A typical form for this purpose is shown in Table .2-II. In using Fig. .2-II and Table .2-II the following assumptions and conventions should be employed:

a. Careful attention should be paid to those foot-notes of Fig. .2-11 which pertain to the sign convention that has been adopted. If known forces are opposite in direction from those shown in Fig. .2-11, a negative sign should be prefixed before inserting in the computations. In particular, it should be noted that the vertical distance, h_2, is negative when the wing aerodynamic center is above the c.g., and positive when the wing aerodynamic center is below the c.g.. The direction of unknown forces will be indicated by the sign of the value obtained from the equations. A negative value of n_3 will usually be determined from the balancing process, indicating a down load on the tail. For conditions of positive acceleration the solution should give a negative value for n_2, as the inertia load will be acting downward. The convention for m_1 corresponds to that used for moment coefficients; that is, when the value of C_M is negative m_1 should also be negative, indicating a diving moment.

b. All distances should be divided by the MAC before being used in the computations.

c. The chord load acting at the tail surfaces may be neglected.

05.244 HEAD-ON LANDING

It should be noted that the load factor specified in CAR 05.244 (4.0) is an ultimate load factor and not a limit load factor.

Revision to Page .2-35

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REVISION SHEET NO. 18

The note below sketch (d) of figure .2-16 should read as follows:

"Note: P = 900 lbs. or 2.0 times the gross weight, whichever is greater (limit).

Figure .3-6 should be as shown below.

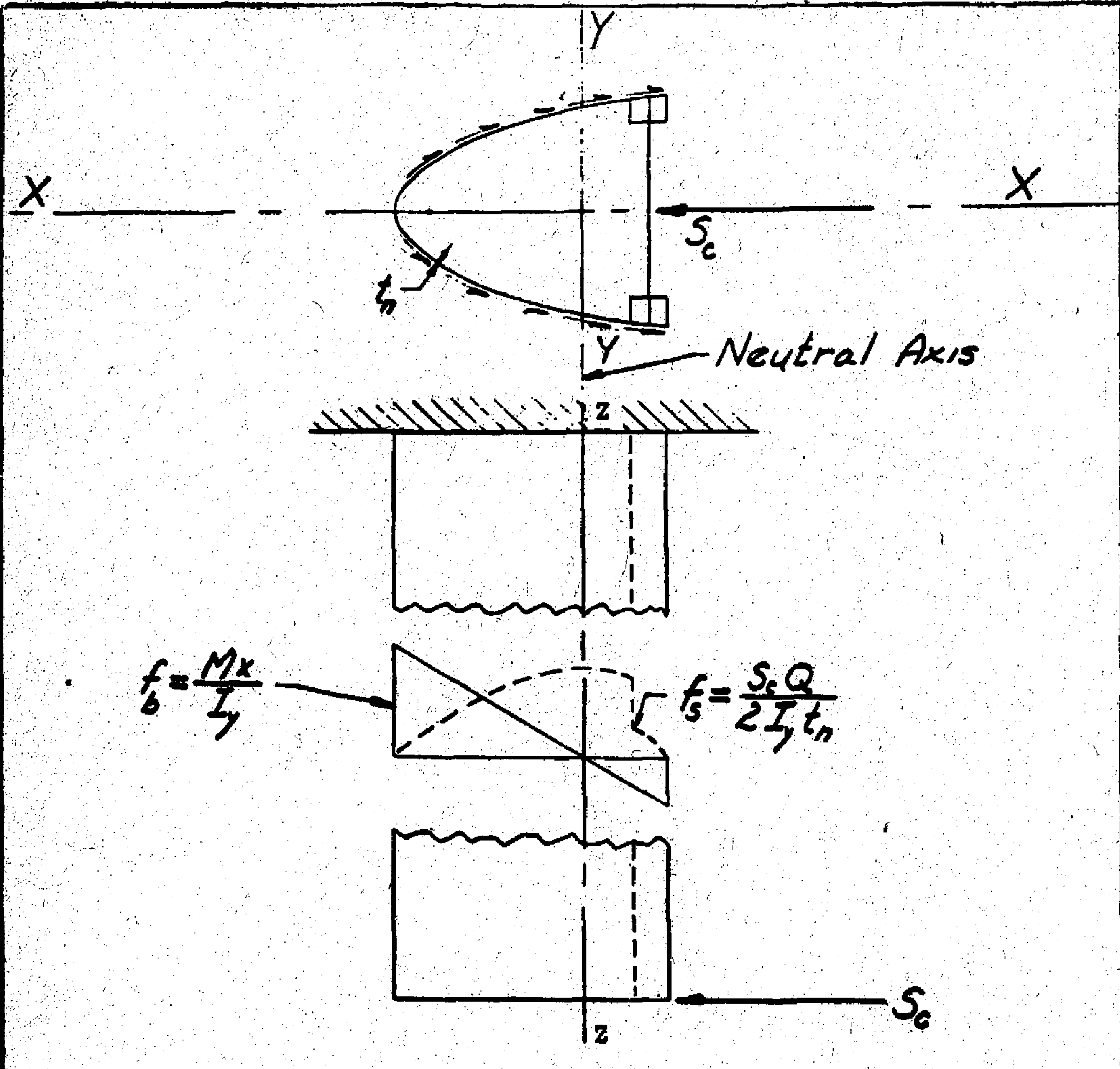

FIG. 3-6

Revision to Page .3-16

05.3213-B DETERMINATION OF ALLOWABLE STRESSES AND MARGINS OF SAFETY

1. Allowable Stresses.

a. In order to determine the allowable stresses for the leading edge skin, a representative section of the D-nose spar should be static tested for the following loadings (the length of the test specimen should not be less than four times the width (or chord) of the spar):

(1) Pure torsion (T) to establish allowables for torsion and chord shear in the leading edge skin.

(2) Chord bending (M_y) to establish the allowable bending stress for the leading edge skin. If chord shear (S_c) is also included in this test, it should be kept low as compared with the chord bending.

b. In order to determine the allowable stresses for the flanges and the web, if necessary because the web is unconventional, a specimen representing the web and flanges should be static tested for pure beam loads (M_x and S).

c. In order to determine the general behavior of the entire structure, the complete spar should be proof tested for the critical conditions(s).

d. The formulas covered in Table .3-III should be used to compute the allowable stresses. Of course, the "effective" areas used in such computations should be consistent with those used in the final stress analysis.

05.3241 **WHEELS AND TIRES**

In wheel type landing gears, the approved wheel rating shall be equal to or exceed the gross weight if one main landing wheel is used, or half the gross weight if two main landing wheels are used. When unrated wheels are employed, their ultimate strengths should not be less than the ultimate loads to which they are subjected. Any standard tire adaptable to the wheel will be considered acceptable.

Revision to Page .3-23

05.3253 - A, 1, b. **Behavior of covering in compression and as a shear web, including the effects of wrinkling.**

insert:-

05.342 OPERATION TEST OF TOWING RELEASE MECHANISMS

As stated in CAR 05.342 (d), operation tests are required to demonstrate that the towing or launching release mechanism will function properly when so loaded as to correspond to one-half the limit forces specified in CAR 05.251. (See also CAM 05.3434 (2d)). These operation tests should be conducted with the fuselage supported and restrained in a manner which will avoid excessive "springing" of the fuselage nose sections when the test load is suddenly released. Inasmuch as the purpose of the test is to demonstrate the releasing characteristics of the mechanism under load, the fuselage resisting loads may be applied at any convenient points on the fuselage near the release mechanism location.

05.34302-2. Loading schedule. A loading schedule should be prepared, showing the load distribution to be used and giving the values of the loads to be applied at each stage of the loading process. When the load is to be applied by means of bags of shot or by weights, it is desirable to weigh each increment of loading in advance and to assign it to a marked space on the floor, so that no confusion will result. The loads can be divided into suitable increments of about one sixth (16.7%) of the required "ultimate" load. In the usual case, such increments will be one quarter (25%) of the required "limit" load, so that the "proof" test load will have been reached at the fourth increment. The "ultimate" load will then be reached at the sixth increment. After reaching the "ultimate" load, the size of the increments should be reduced so that the second additional increment will produce 115% of the "ultimate" load. However, if the structure should show signs of failing at any time the loading increments should be accordingly reduced.

05.34310 - 2c(1)

(1) For the high angle of attack flight condition, ribs of chord length greater than 60 inches should be subjected to 16 equal loads at the load points given in Tables .3-IV or .3-V. In order to determine which set of load points is applicable to the particular airfoil used, it is first necessary to determine the following airfoil characteristics:

a. PD (Pressure Distribution) classification - this is expressed by a capital letter followed by a two digit number such as C 10, B 11, D 12, etc. For the present purpose, only the number portion of the classification need be considered.
b. $C_{m_{a.c.}}$ - moment coefficient about the aerodynamic center.
c. Camber - in percent chord. (This is necessary only in the case of airfoils having a "12" pressure distribution classification.)

These characteristics are readily obtainable for most airfoils from N.A.C.A. Technical Reports Nos. 610 and 628. For airfoils in the 10 or 11 classification, the load points should be taken from Table .3-IV, using the line corresponding to the $C_{m_{a.c.}}$ value of the airfoil. (Table .3-IV should also be used for rib loading points in cases where the P.D. classification is not available, or in cases where the designer does not wish to determine it.) For airfoils in the 12 classification, the load points should be taken from Table .3-V, using the line corresponding to the $C_{m_{a.c.}}$ and the camber of the airfoil. In cases where the actual position of load number 1 is less than 1/2 inch from the leading edge, loads 1 and 2 may be combined into a single load (of twice the unit value) and applied at their centroid. For ribs having a chord of less than 60 inches, 8 equal loads may be used, their arrangement being such as to produce shears and moments of the same magnitude as would be produced by the application of 16 equal loads at the locations specified above.

05.34310-2c(2)

(2) For the medium angle of attack condition 16 equal loads should be used on ribs of chord of 60 inches or greater, 8 equal loads for chords less than 60 inches. In either case the total load should be computed as specified in CAM 05.34310-2a. When 16 loads are used, they should be applied at 8.34, 15.22, 19.74, 23.36, 26.60, 29.86, 33.28, 36.90, 40.72, 44.76, 49.22, 54.08, 59.50, 65.80, 73.54, and 85.70 per cent of the chord. When 8 loads are used they shall be so arranged as to give comparable results.

TABLE .3-IV - RIB LOAD POINTS FOR HIGH ANGLE OF ATTACK

FD Classification	(2) Camber	$C_{m_{a.c.}}$	Load Points in Percent Chord															
			1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
(1) 10 & 11	Any Value	(3) 0 to -.019	.5	1.9	3.4	5.2	7.2	9.6	12.4	15.5	19.0	23.4	28.2	35.2	40.3	48.8	72.0	90.0
		-.020 to -.039	.5	2.0	3.5	5.6	8.0	10.5	13.4	16.6	20.6	25.2	29.6	36.3	42.1	50.2	72.0	90.0
		-.040 to -.059	.7	2.0	4.0	6.3	8.6	11.4	14.6	18.5	22.8	27.2	32.7	38.0	44.6	52.8	72.0	90.0
		-.060 to -.079	.8	2.6	4.5	6.7	9.5	12.7	16.2	20.0	24.2	28.8	34.0	40.0	46.7	54.4	72.0	90.0
		-.080 to -.099	.8	2.8	5.0	7.5	10.5	13.7	17.4	21.2	25.7	30.3	35.6	41.5	47.8	56.4	72.0	90.0
		-.10 or Greater	.8	3.0	5.8	8.2	11.4	14.8	18.6	22.7	27.3	32.2	37.5	42.9	49.6	57.5	72.0	90.0

(1) Shown as C 10, B 11, etc., in data tables of N.A.C.A. Reports 610 and 628.
(2) Expressed as % chord.
(3) Airfoils with + values of $C_{m_{a.c.}}$ are classified with those having a $C_{m_{a.c.}} = 0$

TABLE .3-V - RIB LOAD POINTS FOR HIGH ANGLE OF ATTACK

FD Classification	Camber	$C_{m_{a.c.}}$	Load Points in Percent Chord															
			1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
12	0.0 to 2.9	0.00 to -.0199	.7	2.3	3.9	5.6	7.9	10.4	13.1	16.3	20.1	24.4	28.9	34.3	41.0	48.0	72.0	90.0
		-.02 to -.0399	.8	2.5	4.5	6.4	8.7	11.3	14.1	17.6	21.8	25.5	30.6	36.2	45.2	51.1	72.0	90.0
	3.0 or Greater	0.00 to -.0199	.8	2.5	4.6	6.5	8.7	11.0	13.5	16.4	19.7	23.6	28.0	33.3	39.7	47.7	72.0	90.0
		-.02 to -.0399	.9	2.8	5.0	7.2	9.6	12.0	14.7	17.9	21.5	25.6	30.1	35.3	41.8	49.7	72.0	90.0

05.34311 WING TORSION TEST

(Delete this section. (This subject is now covered by § 05.350. Figure .3-8 (page .3-39) is still applicable.)

05.3434-2, d

d. Towing and launching loads. CAM 05.251 states that it will be unnecessary to investigate launching and towing loads aft of the front spar. However, in testing for these conditions, loads must be applied at points aft of the rear spar to resist the test load on the towing hook. Care should be taken in testing for these conditions to guard against overloading such portions of the fuselage. For instance, if the side load (CAR 05.251-d) is resisted by the front strut fitting and the tail post, and nothing else, the loading in the rear part of the fuselage might be higher than the design load and failure would occur. The solution in this case would be to apply a moment at the wing root fittings and at the strut points as well as at the tail, each of which would be less than the maximum loads for which the fuselage is designed.

When deciding the magnitude and location of the fuselage loads which will resist the towing loads for test purposes, it should be borne in mind that, in actual flight, the loads on the tow line, especially side loads, are resisted by inertia loads as well as by air loads. For example, much of the side load will be resisted by the inertia of the wing, through the wing root fittings, while the vertical components of the towing loads will be resisted mainly by the inertia of the various items of mass in the fuselage, with the loads being applied through their points of attachment to the fuselage.

Reference is made to CAR 05.245 which affects the strength of the wing root fittings and surrounding structure against unsymmetrical loads.

The comments of the preceding section regarding testing for towing loads apply also to landing gear static tests. Care should be exercised so that no part of the fuselage is overloaded locally at points where high loads would not normally be expected.

insert:

05.35 FLUTTER PREVENTION TESTS

05.350 WING TORSION TESTS AND DETERMINATION OF COEFFICIENT OF TORSIONAL RIGIDITY C_{TR}

1. In order to determine the coefficient of torsional rigidity C_{TR}, it is necessary to apply a pure torsional couple near the wing tip and to measure the resulting angular deflection of the wing at selected intervals along the semi-span.

2. Set-up. The wing should be mounted on the fuselage or a suitable jig, either of which should be anchored solidly to the floor or wall to prevent movement or displacement of the wing. The landing gear should be blocked on the airplane. The torque load may be applied to the wing tip through several beams clamped to the wing as near to the tip as is practical, such as the outermost drag truss compression rib location. The platform cables should be attached to the torque beams an equal distance forward and aft of the elastic axis of the wing. This axis may be located experimentally by rocking the torque beam and noting the modal point on the wing chord as viewed from the tip. Typical set-ups are shown in Figure .3-8. Care should be taken to see that the strength of the local wing structure at the points of application of the torque loads from the beams is adequate. For conventional two spar wood wings, it is advisable to apply the load directly to the spars through wood blocks rather than attempt to carry the load through a rib to the spars. Wings which are to be fabric covered should be tested uncovered.

Scales for reading the deflections should be suspended from the leading and trailing edges of the wing (excluding the aileron T.E.) at intervals of approximately 10% of the wing semi-span, and should preferably be graduated in the decimal system with graduations sufficiently fine to obtain readings to a hundredth of an inch. The deflection readings can be readily obtained by the use of a "Wye" level or transit set up at some point that will permit sighting on all scales. Several additional scales should be attached to the fuselage and opposite wing (or jig) to determine if there is any relative movement of the entire airplane. The level should be checked against a bench mark on the wall before and after each group of readings.

3. Loading. The following procedure may be used:

a. Obtain zero torque reading on all scales, i.e., the two platforms should be supported so that there will be no torque couple acting.

b. Add a sufficient amount of weight to each platform until readable deflections are obtained. In general, it will be found desirable to make this first torque moment (inch-lbs.) equal numerically to approximately 140% of the gross weight of the glider. Care should be taken to include the tare weights of the platforms in the torque computations.

c. Take readings of all scales.

d. Add sufficient load to increase the torque by 50% and take scale readings.

e. Add sufficient load again to increase the original torque by 100% and take scale readings. This last torque should result in a twist of the wing of from 1.5 to 2.25° at the wing tip, which is desired in order to obtain satisfactory data for computing C_{TR}.

f. The data to be recorded are: the load applied; its lever arm; the deflection readings at selected points; and the exact location of these points both along the span and along the chord of the wing. It would be desirable to use a table such as shown on page .3-52 which would include all computations necessary for determining C_{TR}.

4. Interpretation of results. Having obtained the leading and trailing edge deflections (F and R in table .3-VI or a corresponding set of data, the angle of twist at each section of the wing for a given torque, or platform load, is calculated and plotted against the wing semi-span measured from the wing tip.

θ = Angle of twist in degrees at any section of the wing

$$\theta = \tan^{-1} \frac{(\text{Leading edge defl. (F)} + \text{trailing edge defl. (R)})}{(\text{(c) Chord distances between scales})}$$

or $$\theta = 57.3 \frac{(F + R)}{(C)} \text{ degrees}$$

Plotting the deflection (F and R) and angle of twist (θ) against wing semi-span (L) will reveal any inaccuracies in the data and will facilitate checking the results.

The coefficient of torsional rigidity may now be computed, using the following expression:

TABLE .3-VI

WING TORSION TEST OF ____________ MODEL ____________ SERIAL NO. ____________

DATE ____________ TORQUE ARM ____________ inches

BY ____________ LOCATED ____________ inches from wing tip

MOMENTS (M)
1. W_1 x ARM =
2. W_2 x ARM =
3. W_3 x ARM =

DEFLECTION READINGS OF ____________ WING (in.)

Platform load (incl. Platform wt.) (lbs)	section A-B		C-D		E-F		G-H		I-J		K-L		ETC
	Front	Rear	F	R	F	R	F	R	F	R	F	R	-
0	Fo	Ro											
1. W_1	F_1	R1											
2. W_2	F2	R2											
3. W_3													
0													

DEFLECTIONS OF ____________ WING (in.)

1. W_1	F_1-F_o	R_1-R_o											
2. W_2													
3. W_3													

TOTAL DEFLECTION OF ____________ WING (in.) = F + R

1. W_1							
2. W_2							
3. W_3							

(C) CHORD DISTANCE BETWEEN DEFLECTION POINTS, F and R (in.)

ANGLE OF TWIST OF ____________ WING (degrees) = $57.3 \left(\frac{\text{total defl}}{c}\right)$

1. W_1							
2. W_2							
3. W_3							

(L) SEMI-SPAN DISTANCE FROM WING TIP (in.)

$dL/d\theta$ = 1/TANGENT TO "θ" VS "L" CURVE AT SECTIONS

1. W1						
2. W2						
3. W3						

$C_{TR} \times 10^{-6} = M \frac{dL}{d\theta} \times 10^{-6}$

1. W1						
2. W_2						
3. W_3						

Remarks:

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$$C_{TR} = M \frac{dL}{d\theta} = \frac{M}{\frac{d\theta}{dL}}$$

where

C_{TR} = Coefficient if torsional rigidity (lb.in.^{2}). It is equal to the reciprocal of the torsional deflection per unit length per unit torque and is usually expressed in values to the 10^{-6}.

$d\theta$ = Angle of twist in degrees, in length dL (in inches), caused by a torque of M inch pounds. Referring to the curve of angle of twist (θ) vs. semi-span (L) shown in Figure .3-9, it will be seen that $\frac{d\theta}{dL}$ = slope of the tangent drawn to the curve at any given point. Hence, it is only necessary to draw the required tangent to the curve at the value of L at which the C_{TR} is desired and obtain $\frac{d\theta}{dL}$ to use in the above formula for C_{TR}.

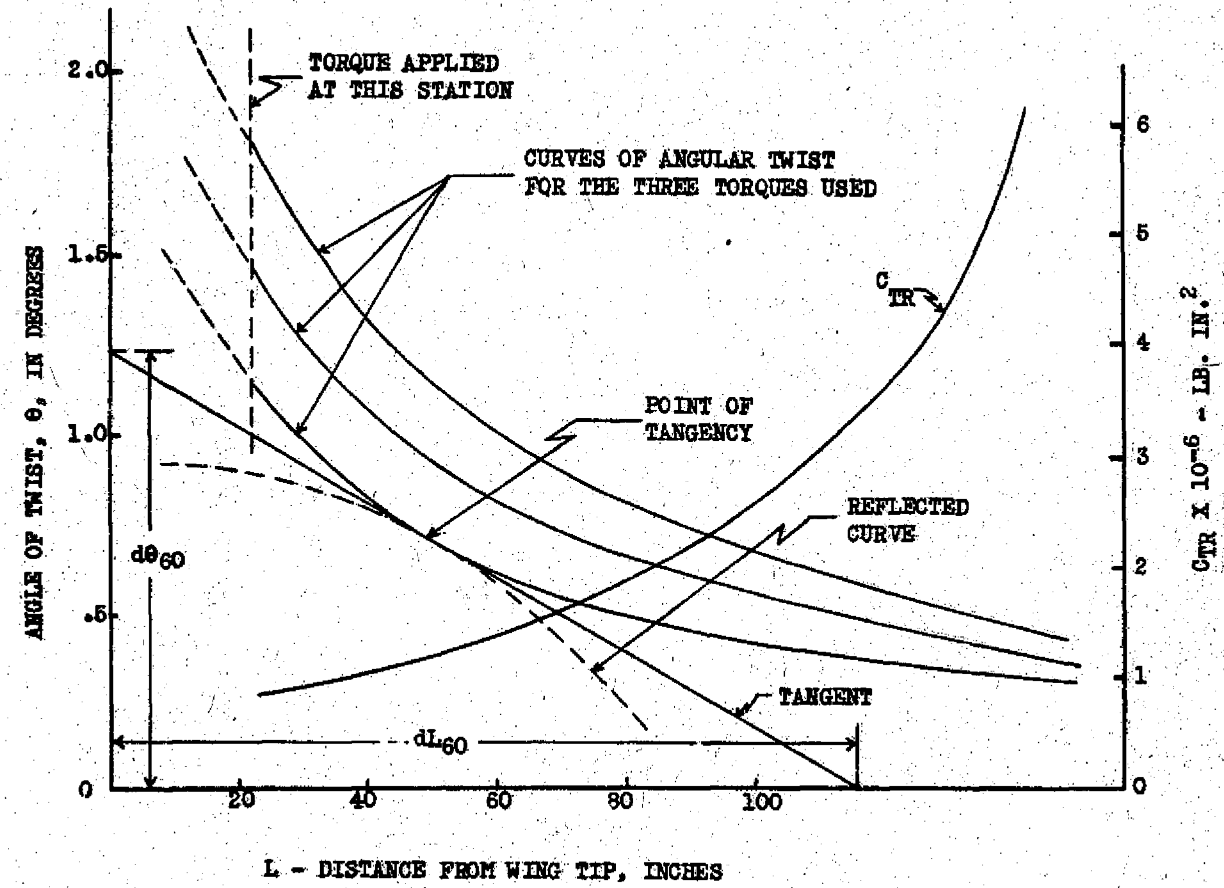
It is very important that the tangent line be drawn accurately. This can best be done by first drawing the reflected curve to the point of tangency (original curve may be drawn on transparent paper and used reversed or the tangent spotted in directly by use of a small mirror), and then by bisecting the resulting angle, as shown in Figure .3-9 for a wing section 60 inches from the wing tip. The tangent line should be extended to both axes so that the slope of the line may be computed accurately which in this example is equal to $d\theta_{60}/dL_{60}$.

C_{TR} should be computed for each of the three torque conditions used at a number of points along the wing semi-span and plotted against the distance from the wing tip (L). This curve will show the variation of torsional rigidity throughout the semi-span and may be used for purposes of comparison with wings similarly tested. (See CAM 05.4100).

.351 VIBRATION TESTS

1. The required vibration tests may be made by shaking the various units of the glider by means of an unbalanced rotating weight driven through a flexible shaft at speeds which can be controlled and measured, or by other acceptable methods. These tests should be made on a complete glider. The frequencies obtained for the various units should be entered in Form ACA-719-Flutter Control Data. (A reproduction of this form to approximately ½ scale is shown as Table .3-VII on page .3-55.) Copies of this form may be obtained from the offices mentioned in paragraph 2 below.

2. Vibration equipment is available at the Civil Aeronautics Administration offices at LaGuardia Field, Long Island; Kansas City, Missouri, and Santa Monica, California. Loan of this equipment may be obtained by contacting the Regional Manager. Civil Aeronautics Administration representatives will assist the manufacturers' personnel in operating this equipment.



RECOMMENDED SCALES: L:1" = 20"
θ:1" = .1° TO .2°

FIG. .3-9 PLOT OF RESULTS OF TORSION TESTS

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TABLE .3-VII

Form ACA-779

Date of Test: ______ Recorder: ______

Test Recorder: ______

FLUTTER CONTROL DATA (k)

Model: ______

Serial No.: ______

Surface			Actual wt. of surface and balance lbs.	Semi-span, length or height ft.	Weight at 3/4 semi-span lbs/ft.	Area sq. ft. (a)	Balance area (b)	[illegible]	[illegible]	Elastic axis % chord	C.G. ... (i)	Static unbalance (c)	I Static axis (h) ⊥		I/I ⊥ Static axis				Vibration test data — Frequencies — Structure: 1 Bending c.p.m.	2 Torsion c.p.m.	Control system: 3 Symmetrical c.p.m. (j)	4 [illegible] c.p.m.		Size & weight of exciter	Sketch no. (g)	Remarks
A Aileron	Left																									
	Right																									
B Wing	Left																									
	Right		(e)								(i)								(e)	(f)						
C Rudder																										
D Fin																										
E Fuselage																			V S							
F Elevator	Left																									
	Right																									
G Stabilizer																										
H Tab-Aileron	Trim Servo																									
I Tab-Rudder	Trim Servo												(d)													
J Tab-Elevator	Trim Servo												(d)													
K Flap																										

Frequency (l) Ratios
Rudder / Fuselage torsion = [illegible] = { } =
Rudder / Fuselage side bending = [illegible] = { } =
Rudder / Fin bending = C3/[illegible] = { } =
* Rudder / Stab. bending = [illegible] = { } =
* Rudder / Stab. [illegible] = C4/[illegible] = { } =
* Rudder / Fin bending = C4/[illegible] = { } =
Elevator sym. / Fuselage vertical bend. = F3/EV = { } =
Elevator sym. / Stabilizer bending = F3/G1 = { } =
Elevator unsym. / Fuselage torsion = F4/E2 = { } =
Ailerons / Wing bending = A3/B1 = { } =
Ailerons / Wing torsion = A4/B2 = { } =
Wing bending / Wing torsion = B1/B2 = { } =
Stabilizer bending / Stabilizer torsion = G1/G2 = { } =

*Only for rudders not in the plane of symmetry.

NOTES:

(a) Area of: (1) Aileron, (1) Rudder, (1) Flap, (1/2) Total Wing (including Ailerons and Flaps), (1/2) Total Stabilizer, (1/2) Total Elevator.
(b) All area forward of hinge line.
(c) Estimated unit weight at 3/4 semi-span location.
(d) Not required if irreversible mechanism is used and tab frequency is greater than 1500 c.p.m.
(e) Include secondary mode for large aircraft, also unsymmetrical modes.
(f) For bi-planes include cellule torsion frequency.
(g) Show nodal lines, amplitudes, and vibration exciter location.
(h) Use area given in CAM 04, or base upon vibration tests.
(i) C.G. location at 3/4 semi-span location. In percent of chord.

(j) Symmetrical - Both Ailerons down (or up).
- Rudder - Twin
Rudder Trailing Edges both in (or both out)
- Elevators - Both halves down (or up).
(k) See CAM 04 for details.

(l) In above fractions shown in Frequency Ratio column, letters refer to SURFACE line, numbers refer to FREQUENCIES column, thus C3 is Rudder symmetrical (control) system.
EV Fuselage torsion

V = Vertical bending
S = Side bending
R = Stab. rocking about attachment

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3. Attitude of Aircraft. All of the vibration tests, with the exception of the fuselage vertical bending and possibly the fuselage side bending tests, can be conducted with the glider tail wheel (or skid) resting on the ground, providing that the natural frequencies of the various units may be correctly recognized with the glider in this position. It has been found desirable, in some cases, to deflate the landing gear tires (and tail wheel tire, if used) approximately 25%, in order to lower the natural frequency of the tires below the frequency range expected for the structure. If difficulty is experienced in recognizing the significant frequencies with the tail wheel (or skid) on the ground, it should be raised just free from the ground, either by a sling around the fuselage located as far forward as is practical, or by blocking up in the region of the wings. The latter procedure may be preferable for the fuselage vertical and side bending modes.

4. Location of Vibrator on the Structure.

a. The proper location of the vibrator on the structure is important in obtaining satisfactory results. Suggested vibrator locations for exciting various modes of vibration are given in Fig. .3-10.

b. The effect of the vibrator weight on the frequency of the structure may be appreciable especially for the control surfaces. The lightest weight vibrator giving satisfactory results should be used. In general, vibrators weighing up to 10% of the weight of the surface to which they are attached may be used without correcting the observed frequencies, unless the vibrator distance from the hinge line is such as to create a much larger relative effect upon the moment of inertia of the surface. However, approximate frequency corrections can be made by adding several small increments of weight near the vibrator at the same arm from the hinge line as the vibrator, and plotting the resulting total increment weights (including vibrator weight) against the frequencies observed. Extrapolating this curve to zero weight should give the corrected frequency.

c. In general, the following points should be considered in the attachment of any type of vibrator to a structure.

(1) The location is of primary importance and should be at a point of large deflection. See Fig. .3-10.

(2) The vibrator should be so mounted that its line of force will be in the most advantageous direction to excite the vibration mode desired.

(3) It is desirable to attach the vibrator to a part of the structure that is fairly rigid such as the wing spar, control surface ribs, etc.

(4) The local structure to which the vibrator is attached should have adequate strength for the loads imposed by the vibrator.

VIBRATION MODES AND VIBRATOR LOCATION

(IN GENERAL ONLY A FRACTION OF THE MODES LISTED WILL BE APPLICABLE TO ANY ONE AIRPLANE)

ITEM	SURFACE	MODE	DESCRIPTION OF MODE AND SUGGESTED VIBRATOR LOCATION
1	Rudder (Single)	(As a Unit)	Rudder swinging back and forth, under the spring action of the control cables. (Vibrator aft of horn on rudder T.E.)
2	Rudder (Single)	Torsion	Torsional vibration of the rudder, under the spring action of the torque tube (Nodal line extends from trailing edge to torque tube.) (Vibrator near upper end on rudder T.E.)
3	Rudders (Twin Tail)	Sym.	Same as item 1 above, except both rudders swinging inward (or outward) at same time. (Vibrator aft of horn on rudder T.E.)
4	Rudders (Twin Tail)	Unsym.	Same as item 1 above, except both rudders moving to right (or left) together. (Vibrator aft of horn on rudder T.E.)
5	Rudders (Twin Tail)	Torsion	Same as item 2 above.
6	Elevator	Sym.	Both elevators swinging up or down together under the spring action of the control cables (or push pull tubes) (Vibrator at inner end of one elevator T.E. or aft of horn on single elevator.)
7	Elevator	Unsym.	One elevator (or 1/2 of single elevator) moving up, other down at same time under spring action of torque tube. (Nodal line in plane of surface, extending aft from point near center of elevator spar.) (Vibrator at center of semi-span near T.E.)
8	Aileron	Sym.	Each aileron as a unit swinging up and down together, under the spring action of the control cables (or push pull tubes). (Vibrator between center and inner end at aileron T.E.)
9	Aileron	Unsym.	One aileron as a unit moving up, other down at same time, under spring action of control cables. (Vibrator aft of horn at aileron T.E.)
10	Aileron	Torsion	Torsional vibration of aileron, under the spring action of the torque tube (Nodal line extends from trailing edge to torque tube.) (Vibrator near outer end at aileron T.E.)
11	Wing	Sym. Bending	Wing bending as a beam supported at its midpoint (fuselage), or as a braced beam supported at the brace points. (Vibrator near wing tip approximately on elastic axis of wing.)
12	Wing	Unsym. Bending	Same as item 11 above, except that right and left halves of the wing move in opposite directions at the same time. Usually only important in large multi-engine aircraft. (Vibrator near wing tip approximately on elastic axis of wing.)
13	Wing	Sym. Torsion	Torsional vibration of the wing about a spanwise axis. Right and left halves of the wing move in same direction about this axis at the same time. (Vibrator near wing tip, forward or aft of elastic axis of wing.)
14	Wing	Unsym. Torsion	Same as 13 above, except right and left halves of the wing move in opposite directions about a spanwise axis at the same time. (Vibrator near wing tip, forward or aft of elastic axis of wing.)
15	Wing (Biplane only)	Cellule Sym. Torsion	Same as item 13 above, except center of rotation may be located between upper and lower wings. (Vibrator acting in chordwise direction at upper wing outer rear interplane strut attachment.)
16	Wing (Biplane only)	Cellule Unsym. Torsion	Same as item 14 above, except center of rotation may be located between upper and lower wings. (Vibrator acting in chordwise direction at upper wing outer rear interplane strut attachment.)
17	Fuselage	Torsion	Whole tail unit and fuselage vibrating torsionally about longitudinal axis. (See item 23 also) (Vibrator on fin or stabilizer - on hinge line, at tip of surface.)
18	Fuselage	Side Bending	Whole tail unit vibrating in side direction about vertical axis through fuselage forward of tail unit. Usually important only on large airplanes. (Vibrator at tail end of fuselage.)
19	Fuselage	Vertical Bending	Same as item 18 above, except vibrating in vertical direction about horizontal axis through fuselage. Only important in very large aircraft having fuselage extending in top and bottom sides. (Vibrator at tail end of fuselage.)
20	Stabilizer	Sym. Bending	Stabilizer bending as a beam supported at its midpoint (fuselage). Usually only important in cantilever tail surface designs. If wire or strut braced with small tip overhang, the mode of vibration may not be such as to permit interaction with the elevator. (Vibrator near outboard end of stabilizer.)
21	Stabilizer	Sym. Torsion	Torsional vibration of the stabilizer. Similar to item 13, for the wing. Usually only important for cantilever and twin tail aircraft. (Vibrator near outer end of stabilizer at L.E.)
22	Stabilizer	Unsym. Torsion	Torsional vibration of stabilizer. Similar to item 14 for the wing. Nodal line at ℄ of stabilizer (fuselage). (Vibrator at leading edge of stabilizer or at trailing edge of elevator. Elevator clamped to stabilizer.)
23	Stabilizer	Rocking about its fuselage attachments	See item 17 also. Stabilizer as a unit rocking about its fuselage attachments. Usually only important for twin tail aircraft. (Vibrator on fin or outboard end of stabilizer on twin tail aircraft, or near outboard end of stabilizer - elevator hinge line on single tail aircraft.)
24	Fin (Single tail)	Bending	Fin bending as a beam fixed at one end. Usually only important in cantilever tail surface designs. (Vibrator near upper end of fin near rudder hinge line.)
25	Fin (Single tail)	Torsion	Torsional vibration of the fin. Similar to item 21 for the stabilizer. (Vibrator on fin L.E. near upper end.)
26	Fin (Twin Tail only)	Bending (Sym. with respect to attaching fin to stabilizer	Fin bending as a beam fixed at its attachment to stabilizer. (Vibrator near upper, or lower, end of fin near elastic axis of fin.) (Upper and lower portions may have different frequencies.)
27	Fin (Twin Tail only)	Bending (Unsym. with respect to attachment of fin to stabilizer)	Fin bending as a beam fixed at its attachment to stabilizer.
28	Flap	Torsion	Torsional vibration of outboard end of flap. Similar to item 10 above. Assuming irreversible control arm used. (Vibrator near outboard end of flap at T.E.) Only important when flap extends outboard on wing beyond the semi-span location.
29	Tabs (Rudder Aileron Elevator)		Note: Effect of vibrator weight on tab frequency should be investigated when vibrator is attached directly to tab.
30	Trim or Balance Tabs	Torsion	Torsional vibration of the tab under the spring action of the torque tube. Node on T.E. (Vibrator on tab near one end.)
31	Servo Tabs	Sym.	Tab as a unit swinging back and forth under the spring action of the control cables. (Vibrator on adjacent tab supporting structure, or on tab itself.)
32	Servo Tabs	Torsion	Same as item 10 above. (Vibrator on tab near one end.)
33	Balance Weights for movable surfaces - mounted on long supports	Bending of support	Support bending as a beam (fixed at one end) in various planes depending on the rigidity. Most important directions are vertical and sideways with relation to the airplane centerline. (Vibrator near balance weight support.)

FIG. .3-10

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5. Testing. A certain amount of experience is necessary in recognizing the various modes and resonant frequencies. In conducting the tests, the vibrator should be placed on the structure as suggested and then operated at increasing speeds until a response peak is reached (the amplitude of vibration of the structure is appreciably greater than at slightly higher or lower speeds, thus indicating a resonant condition).

6. During the vibration tests involving the control system, the controls should be restrained by an assistant to simulate the condition in flight. When the control system incorporates dampers or power boosters, their effect on the frequencies should be considered. It is important that cable control systems be rigged to their proper tension. In general it will be found that cable control systems will have a larger resonant frequency response range than a more rigid system, such as one incorporating push pull tubes with close fitting joints. In the former case, when an unusually large range is encountered, it is desirable to record the frequencies at both ends of the response range. In most cases it is satisfactory to note only the mean frequency value for the particular mode.

7. It should be noted that it may be possible to excite a certain mode in more than one way. For instance, the fuselage torsional frequency may be excited in the fin bending test and conversely the fin bending frequency may be excited in the fuselage torsion test. Cases of this type will serve as cross checks on each other.

The phase relationship of vibrating parts may be determined by the method shown in Fig. .3-11 as applied to the particular case of the elevators. The metal plates A and B, attached to the trailing edges of the elevators and interconnected with a wire, are necessary only in the case of fabric covered surfaces or surfaces which have a poor electrical interconnection. When the parts are vibrating the phase relationship may be determined by manually holding the leads C and D close to the surfaces so that intermittent contact is made during each cycle. If the light flashes or clicks are heard in the headphones at regular intervals (with the contacts in the same side; i.e., upper or lower), the surfaces are vibrating in phase, whereas, if the light does not flash, or no click is heard in the headphones, the surfaces are out of phase. This should be verified by reversing one contact, for example, putting contact D on the upper side.

The location of the nodes of the various forms of vibration should be established by the tests. In many cases the location of the nodes is self-evident, or can be determined by visual observation or by "feel". Determination of the nodes by the foregoing methods is generally satisfactory for most modes of vibration. If the torsional axis of vibration of the fuselage (or the nodes for other modes of vibration) cannot be definitely established by the above methods, a more detailed procedure, involving measurements of the amplitudes of vibration at various points, should be employed.

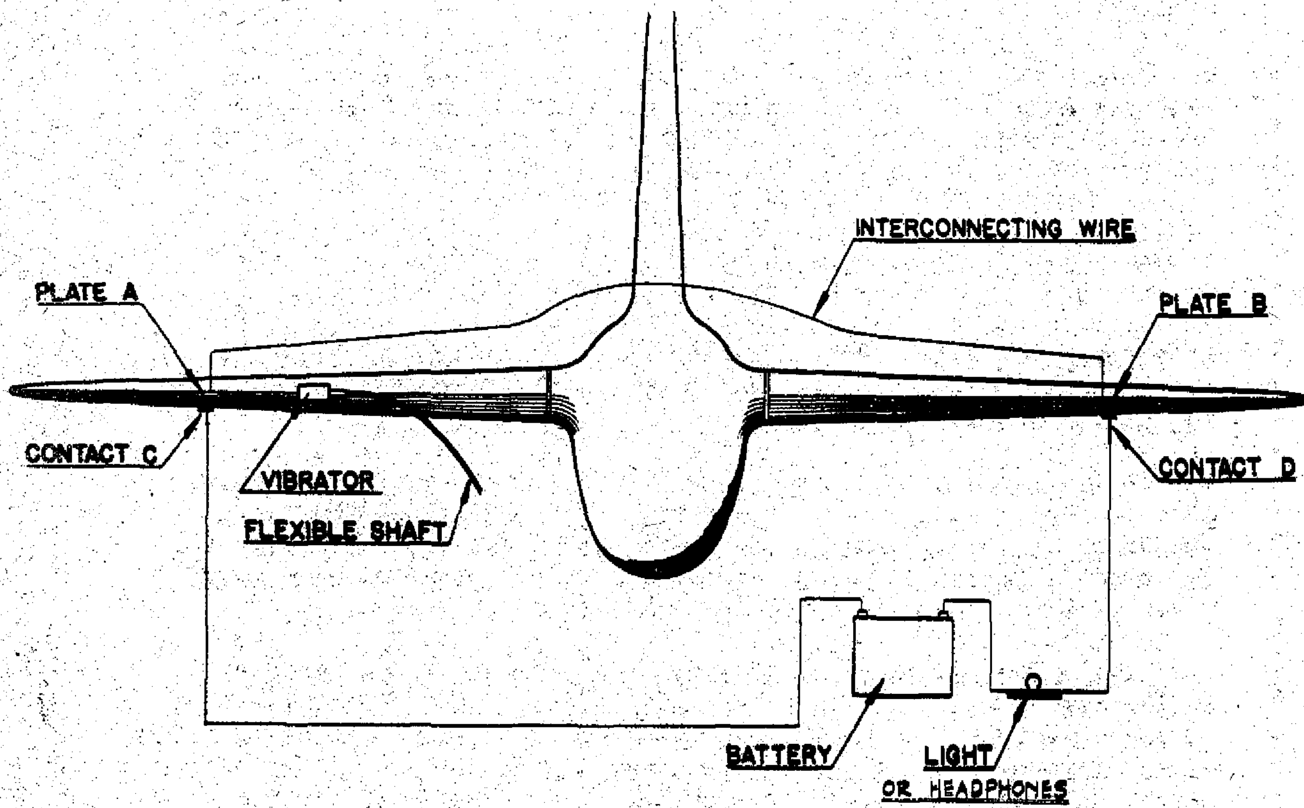


FIGURE .3-11

TEST SET-UP FOR
DETERMINATION OF PHASE RELATIONSHIP

8. Fig. .3-10 gives a detailed description of the possible modes that may be observed during the tests and includes suggested vibrator locations for each mode. In general, only a fraction of the modes listed will be applicable to any one glider.

05-4000-4. Cables and Wires.

Cable and wire where used should be of aircraft quality. Solid wire should be used only in drag systems. Sizes smaller than 16 gauge (.051" diameter) should not be used. The usual ferrule and thimble fastening is satisfactory if soldered with non-corrosive flux and well cleaned and varnished or primed. The wire should not be over heated in the soldering operation. Where used for tail or wing stays, or supports, 19 wire stranded cable with a tinned copper wire wrapped and soldered serving is satisfactory. The use of cables in control systems is discussed in CAM 05.434-1. See Figures .4-1, .4-2 and .4-3 for cable fittings and splices.

05.4000-5 Aluminum Alloys

The aluminum alloys have had some use in present day glider construction for fairing and gap covers as well as for primary structures. It should only be worked as a structural material by those trained in its use and then with the same care as applied to airplanes. It is not advisable to use it for fittings in wood or composite construction gliders unless properly protected from corrosion. (See CAM 05.4013).

05.4001- 2. Splices.

Splices in structural wood members when necessary should have a 12 to 1 slope or greater and surfaces fitted for perfect uniform contact before gluing. The surfaces should be formed with a planer, if possible. The dimensions and type of splice should be similar to those given in CAM 18. Care should be taken in clamping glued splices to use thick "cushion" blocks of the proper slope and size so as to produce clamping action perpendicular to the line of the splice, uniformly distributed, and not such that the pieces tend to slip past each other. A finished splice in wood or plywood should show no change in cross-section at the splice.

05.4001- 3. Glued joints.

High grade casein and synthetic resin glues are satisfactory for making glue joints in aircraft elements. The glue manufacturer's directions should be followed in the mixing and preparation of the adhesives. It should be noted that condition of the surface, moisture content of the wood, gluing pressure, assembly time, and protective coatings as well as other factors play an important part in the fabrication of acceptable joints. Casein glue is extensively used in glider and airplane wood construction and its application has been found satisfactory when used as follows:

All glue joints should be held for at least 8 hours under pressures of 100-150 psi for soft woods and 150-200 psi for hard woods, at temperatures of from 65° F to 90° F. No glue which has been mixed for more than 4 hours should be used for gluing structural parts. Enough glue should be used so that it squeezes out of a joint at the edges and this excess, while it may be removed by scraping, must not be wiped off.

05.4001-6 a. Steel sheet used for stressed fittings should be handled with special care to avoid scratching or marring the surface in any way. Lines should be drawn with a pencil and not a sharp instrument. All bends should be made around a block which has had the corner rounded off to a radius of at least twice the thickness of the metal (See CAM 18). During the bending operation, sheet should be held in a vice which has the jaws covered with copper, aluminum or brass so as to prevent marring. Hammering should be done through a hardwood block rather than on the bare metal. Holes may be punched provided they are reamed out at least .010" to get them up to size. When drilling stacked sheets, the burr should be removed from each sheet so that the hole is clean and surface smooth. Where cuts to form corners are made, a 1/8 inch diameter hole should be drilled in the corner and the cut made to the hole, not past it.

05.4001-6, b. High strength aluminum alloy sheet must be worked with caution to avoid any marring, scratching, or sharp bends. The strong aluminum alloys, such as 17 ST and 24 ST, should not be bent to a radius less than four times the thickness of the sheet (CAM 18). The bend should be preferably at right angles to the direction of the grain. Bends can be made over soft material such as hardwood.

05.4001-7, b. In glider construction, brazing will be found to be a satisfactory method of joining thin wall tubes. When properly done, there is no danger of burning or injuring thin wall tubes. The use of brazing will be permitted in the assembly of light control surface tubing, and in fuselage joints not subject to high loads or located at important connections.

05.4001-9, c. Wings. Long unsupported ribs may require bracing to prevent them from turning over under the fabric. This can be accomplished by cross-bracing with fabric tape cemented to the rib cap-strips. The ribs should be ribstitched with a good linen thread on about 5 inch centers (See CAM 18 for recommended sewing practices). It is also advisable to cement the fabric to the ribs.

5-23020

05.4020-3. Approved locking devices include cotter pins, safety wire, peening, and, with certain restrictions, self-locking nuts and Dardelet Threaded parts. For pins that are frequently removed, hard wire safety pins are satisfactory, provided it can be shown that there is no danger of their coming loose or interferring with the operation of the controls (See CAM 05.414 regarding the use of clevis pins).

05.4020-4. The use of self-locking nuts is limited to the restrictions noted on the pertinent manufacturer's recommended practice sheets listed on the CAA Product and Process Specification P. & P.4.

05.4100 TORSIONAL STIFFNESS

1. It is essential that the wing structure have adequate

stiffness in order to insure freedom from flutter and other undesirable characteristics. This is particularly important with reference to wing torsional stiffness. Fabric covered wings, in particular, may be critical in this respect. The test procedure for determination of the torsional rigidity of the wing is outlined in CAM 05.350.

2. Since the actual torsional deflection of the wing will depend on the moment coefficient of the airfoil employed, it is advisable to introduce the additional criterion that the maximum torsional deflection under the limit load critical for torsion not exceed 3°.

05.4111-1 BUILT-UP SPARS

1. Of the built-up types of spars, the box with smooth plywood faces is the most convenient for attaching ribs and has half as many flanges to make as the "I" type. In either type, blocking must be provided at all points where there are fittings attached, etc. Such blocking must be tapered off at the ends to avoid concentration of stress in the flanges. Intermediate verticals are provided in some cases to increase the allowable stress in the webs. These verticals need not be filleted at the ends unless they also carry a concentrated load to be distributed into the spar web. On box spars the attached rib verticals provide this stiffening effect on the webs. CAM 18 shows details of spar construction.

05.4111-4. Laminated spars may be spliced in either plane, and splices in the various laminations should be spaced well apart. Splices in solid spars, if any, should be in the vertical plane, as shown in CAM 18.

5-22020

05.4120-3. Steel struts have the necessary end fittings welded on so that the load is carried through the weld in shear, while with aluminum alloy the fittings are riveted or bolted into the end of the strut. A typical end-fitting for a streamline steel strut is shown in Figure .4-7. For a single strut, it is desirable to provide universal end fittings similar to Figure .4-7(b).

5-23020

05.434-1. Cables. Control cables should be of the 6 x 19 or 7 x 19 extra-flexible type, except that 6 x 7 or 7 x 7 flexible cable is acceptable in the 3/32 inch diameter size and smaller. For properties see Table 4-14 of ANC-5. The 7 x 7 construction is found satisfactory where slight cable bends around pulleys of only 30° or less are encountered. Cable end splices should be made by an approved tuck method such as that of the Army and Navy, except that standard served and soldered splices are acceptable for cable not over 3/32 inches in diameter. However, cables of other sizes with served and soldered fastenings will be satisfactory provided they are not stressed above 50% of their rated strength. Approved swaged type terminals are also acceptable. Dimensions for approved splices are given in Figure .4-1. It should be remembered that cable sizes are governed by deflection conditions as well as by strength requirements, particularly when a long cable is used. Some acceptable types of cable joints are given in Figure .4-3.

5-23020

05.4610-3. Pilot and passenger enclosures. Removable "scoops" around the pilot should be securely attached to carry the air loads encountered at the maximum gliding speed, but must be easy to release and push off in case the pilot has to bail out. They should be so designed that their removal in flight at high speeds will not injure or inconvenience the pilot or passengers, or block the exits. The nose may be built up around the pilot and only a local portion be removable. Sufficient room should be provided for exit, wearing the type parachute for which the seat is designed. A clear fore and aft opening of not less than 24 inches is desirable.

05.4630-4. Curved monocoque type. The curved monocoque style of construction necessitates laying the plywood on in smaller panels where there is double curvature. Longitudinal plywood seams should be supported by light internal stiffeners. These may run through the bulkheads and serve also as longerons, or be laid only in between as local (intercostal) stiffeners. The rear part of curved fuselages is often made straight conical with oval sections so that there is curvature in one direction only, and the plywood may be laid in long lengthwise panels.

5-23020

Insert:

05.48 TRAILER DESIGN AND RELATED CONSIDERATIONS

1. The usual method of transporting a glider from one location to another is to dismantle the craft to the extent necessary to place it on a trailer which may be towed to the desired location. This introduces the problem of properly mounting and securing the various components of the glider so that no damage will be sustained which would adversely affect the airworthiness of the glider.

2. The following precautions should be observed in an effort to minimize the effects of loads imposed on the glider structure during transit:

(a) The glider fittings normally designed for wing strut attachments, wing root attachments, or empennage assembly should not be used for mounting the glider components on the trailer unless special care is taken to eliminate the possibility of external loads being applied to the glider structure at such fittings.

(b) The points or areas on the glider structure at which the glider components are supported and secured should be locally reinforced and protected to avoid damage or distortion.

(c) The trailer supports should be designed so as to distribute the supporting loads over an appreciable area of the glider structure.

(d) Trailer frames which deflect or weave excessively while being towed over rough road should be avoided. The loads which might be imposed upon the glider structure as a result of excessive trailer frame deflection may cause incipient or actual structural failures.

5-23020

05.713 Airspeed Indicator Installation Calibration

In calibrating the airspeed indicator installation it is important to have available for computation purposes, the winds aloft, the pressure altitude and the free air temperature records. The above values will be employed in conjunction with any of the acceptable calibration methods described in the following:

1. A free flight over a measured speed course of at least one mile, suitable markers of which should be easily visible to enable observations to be made from the aircraft.

2. A flight over a measured speed course, as previously outlined, while being towed by a powered aircraft or an automobile.

3. A flight utilizing a trailing pitot-static tube bomb installation, as previously outlined, while being towed by a powered aircraft.

4. A free flight utilizing a trailing pitot-static tube bomb installation connected to a master airspeed indicator, the indications of which can be used to form a basis for comparison.

6-23020

APPENDIX I

A SIMPLE APPROXIMATE METHOD OF OBTAINING THE SPANWISE DISTRIBUTION OF LIFT ON WINGS

1. SUMMARY

This appendix presents and describes a simple and rapid approximate method for the determination of the spanwise distribution of section lift coefficient c_l* on wings, for use when a rational method is required. One completely worked example and three additional examples which compare the results obtained by this approximate method with available theoretical methods are included. Limitations of the method are given in Section 6. The method herein described incorporates a tabular form for use in making the necessary computations. In practice, it is necessary to enter only 7 basic columns in the table, and the remainder of the work is a simple routine procedure which can be carried out by personnel with no engineering knowledge of the principles involved.

2. BASIC CONSIDERATIONS

It is well known that the lift distribution for any wing can be found in terms of the wing lift coefficient C_L, the basic lift coefficient c_{lb}, and the additional lift coefficient c_{la_1}, as related by the formula:

$$c_l = C_L c_{la_1} + c_{lb} \qquad (1)$$

In order to determine the values of c_l along the span for any given design condition corresponding to a specific value of the wing lift coefficient C_L, it is, of course, necessary to know the values of c_{la_1} and c_{lb} along the span. (It might be noted here that, if the wing has no aerodynamic twist, $c_{lb} = 0$ and $c_l = C_L c_{la_1}$.) These may be determined by the following approximate formulas which were derived from the results given in Reference 1:

$$c_{la_1} = 1/2 \left[\frac{a_o}{\bar{a}_o} + \frac{4\bar{c}}{\pi c} \sqrt{1 - \left(\frac{y}{b/2}\right)^2} \right] \qquad (2)$$

$$c_{lb} = \frac{a_o}{2} (\alpha_{Ro} + \beta) \qquad (3)$$

* See Section 8 for nomenclature.

In using these basic formulas, the following values must be determined:

$$\bar{a}_o = \frac{\int_o^{b/2} a_o c dy}{b/2\ \bar{c}} \qquad \text{(Mean value of } a_o\text{)} \qquad (4)$$

$$\alpha_{Ro} = \frac{-\int_o^{b/2} a_o \beta c dy}{\int_o^{b/2} a_o c dy} \qquad \text{(Angle between wind direction and the reference axis, for zero wing lift)} \qquad (5)$$

$$= \frac{-\int_o^{b/2} c\beta dy}{b/2\ \bar{c}} \qquad \text{(This is a simplification of formula (5) for use when } a_o \text{ is constant along the span)} \qquad (5a)$$

The computation of $c_{l a_1}$ and $c_{l b}$ is conveniently adapted to a tabular form, the use of which is described in the following section:

3. USE OF TABULAR FORM

The tabular form for computing the values of $c_{l a_1}$ and $c_{l b}$ is shown as Table 1. Briefly, the use of this Table consists of entering the basic geometrical data required in columns 1, 2, 3, and 16; entering the "multiplier" in column 4; entering the basic aerodynamic data required in columns 6 and 17; and then proceeding with the routine computations as indicated in the body of the Table. The value of $c_{l a_1}$ is then obtained as column 15. The value of $c_{l b}$ is obtained as column 23, in case high lift devices are not used, and as column 26 if such devices are used. The procedure for using this Table will now be outlined:

Column 1

Before entering the values of $\frac{y}{b/2}$ in this column, it is necessary to divide the semi-span into a convenient number of sections, and then divide these sections into a convenient number of even parts. Examples of this are shown on Fig. 1. It is necessary to locate section divisions at the beginning of the tip region (See Fig. 1(1)), at the ends of high lift devices (See Fig. 1(2)), and at points where there is an abrupt change in plan form (See Fig. 1(3)). These section divisions are shown as heavy lines on Fig. 1. The sections thus obtained are then divided into an even number of parts as indicated in Fig. 1. (An even number of parts is necessary in order to insure accuracy in the numerical integration which is automatically provided for in the Table.) The values of $\frac{y}{b/2}$ may now be entered in column (1), taking care to enter the $\frac{y}{b/2}$ values at the main section division twice, as shown in the numerical example (Fig. 3).

Column (2)

Enter the chord, c, in inches corresponding to the $\frac{y}{b/2}$ value on the same line.

Column (3)

Enter here the actual width in inches of the small divisions of the semi-span within the section (See Fig. 3).

Column (4)

Enter here a multiplier which, within a section, is a series of the following form (Simpson's rule for approximate integration):

It will be noted that the first and last terms of this series are .333, and the intermediate terms are a repetition of 1.333 and .667. Examples of multiplier values follow:

Two divisions: .333, 1.333, .333
Four divisions: .333, 1.333, .667, 1.333, .333
Six divisions: .333, 1.333, .667, 1.333, .667, 1.333, .333
Eight divisions: .333, 1.333, .667, 1.333, .667, 1.333, .667, 1.333, .333

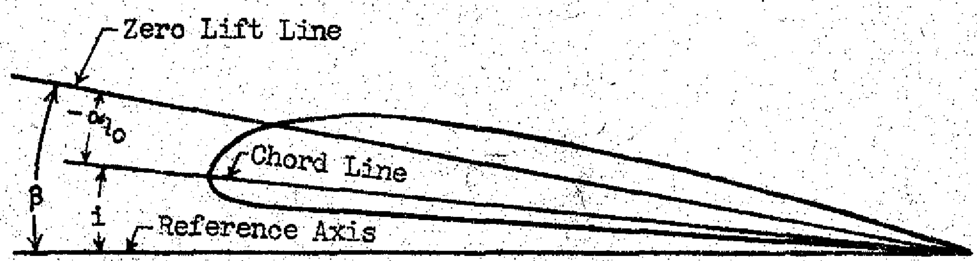
(See also Fig. 3 for an example of this procedure.)

Column (6)

Enter here the slope of the lift curve, a_o, for infinite aspect ratio in C_L per degree for the pertinent airfoil section (or airfoil-flap combination). Data for this purpose can be obtained from standard NACA reports.

Column (16)

Enter here the angle of incidence, i. This is the angle between the chord line (the line used as datum for airfoil ordinates and angles) and the reference axis. The reference axis can be chosen as any convenient axis in the plane of symmetry, such as, the fuselage axis or the chord line of the root chord. Care should be taken in using the proper sign for i, positive values being as measured in the sketch below. (The sign of α_{l_o} is shown as negative in the sketch to agree with NACA airfoil data where the reference line for angles of attack is always the chord line. Therefore, considering only the geometry of the particular sketch, β is obviously $i + \alpha_{l_o}$ or using the sign and expression for β given on the sketch, $\beta = i - (-\alpha_{l_o}) = i + \alpha_{l_o}$).



$\beta = i - \alpha_{l_o}$

Column 17

Enter here the angle of attack for zero lift, α_{l_o} for the pertinent airfoil section (or airfoil-flap combination), taking care to use the proper sign. Data for this purpose can be obtained from standard NACA reports. Computations can now proceed in accordance with the instructions on Table 1. In cases where high lift devices are not used, the final values of $c_{l_{a_1}}$ and c_{l_b} for design purposes are given in Columns 15 and 23, respectively.

When such devices are used, the $c_{l_{a_1}}$ values of Column 15 still apply, but it is necessary to fill out Columns 24, 25, and 26 in order to obtain the final c_{l_b} (Column 26) values for design purposes. Instructions for filling out these columns are given in the following section.

4. PROCEDURE FOR OBTAINING c_{l_b} WHEN HIGH LIFT DEVICES ARE USED

When high lift devices are employed, it will be found that the c_{l_b} values in Column 23 have a sharp discontinuity at the end (s) of the flap, as shown in the example problem, Fig. 3. It is, therefore, necessary to properly adjust these values in order to obtain better agreement with actual measured span distributions.

This adjustment process is performed by computing Column 24 to obtain $c_{l_{bc}}$ and plotting the values thus obtained against the semi-span. Examples of this are shown by the dashed lines on Figs. 2 and

6. These curves are then faired as shown by the solid lines on Fig. 2 and 6, taking particular care to fair in such a manner that the total area under the faired curve is equal to zero. The values of $c_{lb}c$ from the faired curve are then entered in Column (25) and the final c_{lb} for design purposes is obtained in Column (26)

5. COMPARISON EXAMPLES

These examples are included to show a comparison between the results obtained by the approximate method outlined herein and more exact theoretical methods which have previously been shown to give satisfactory agreement with experimental results. (Reference 1 includes a large number of comparison examples which are of interest.)

Example #1

The wing planform of this example is shown in Fig. 2. The wing has no aerodynamic twist, except that induced by the flap, which is deflected 30°. This example is taken from NACA Technical Report 585, page 3 (Reference 2). A table showing the computation of c_{la_1} and final c_{lb} is shown in Fig. 3, fairing of $c_{lb}c$ is shown on Fig. 2, a table giving the computation of c_l for a wing lift coefficient $C_L = 1.72$ is shown on Fig. 4, and a comparison of final values of c_l, c_{la} and c_{lb} with those obtained theoretically by reference 2 is shown in Fig. 5. It will be noted that the agreement of the c_l values is very satisfactory for design purposes.

Example #2

The wing planform for this example and the comparison curves of c_l are shown on Fig. 7. This wing has no aerodynamic twist. It will be noted that the agreement of the approximate method with the theoretical results is satisfactory for design purposes. (The C_L value of 4.52 for this example is a theoretical value corresponding to an angle of attack beyond the stall. However, the c_l values for this wing at angles of attack below the stall would be directly proportioned to those shown on the Figure.)

Example #3

The wing planform and comparison curves for this example are shown on Fig. 8. This wing has a straight center-section, a root to tip chord ratio of 4, and an aerodynamic washout of 2 degrees. The lift coefficient of the wing, C_L, is 0.687. In this case, a comparison is made between the $\frac{c_l c}{b/2}$ values given by the approximate method and the theoretical curve from NACA Technical Note 732 (Reference 3). (It will be noted that the value $\frac{c_l c}{b/2}$ is directly proportional to the load per foot of span acting on the wing.)

Example #4

This comparison example is shown on Fig. 9. The flap deflection is 60 degrees, the ailerons are in neutral position, and the wing C_L is 1.716.

The theoretical curves for c_l, c_{la}, and c_{lb} are the same as those shown in Fig. 7-4, page 7-44, of ANC 1 (1). It will be noted that the agreement of the approximate method with the ANC 1 (1) theoretical method is entirely satisfactory for structural design purposes. Fig. 9 also shows a comparison of the $c_l c$ values of the approximate and theoretical methods. (The term $c_l c$ is directly proportional to the load per foot of span acting on the wing.)

6. CONCLUSIONS

On the basis of the comparison examples contained herein and many other comparison examples which have been completed, it is concluded that the approximate method given herein of computing spanwise distribution of c_l is satisfactory for structural design purposes, provided that:

1. The aspect ratio is within the normal range of values (say from 5 to 12), and,
2. The wing has reasonably rounded tips, if the taper ratio is greater than 0.5. (This restriction as to rounded tips does not apply for taper ratios less than .5.)

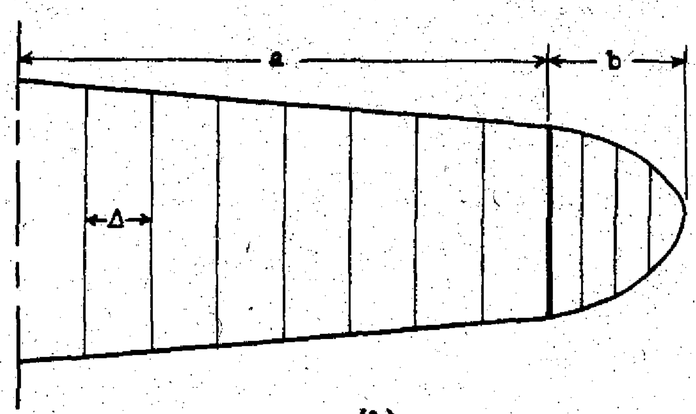
In cases where the wing does not have reasonably rounded tips and, at the same time, the taper ratio is greater than .5, the approximate method can also be used provided that an empirical tip correction such as is outlined in paragraph 1.32 of ANC 1 (1), "Spanwise Air Load Distribution" is employed. This method is considered satisfactory for any amount of aerodynamic twist that may be encountered in conventional design practice.

7. REFERENCES

1. Schrenk, O.: A Simple Approximation Method for Obtaining the Spanwise Lift Distribution. T. M. 948, N. A. C. A., 1940.
2. Pearson, H.A.: Span Load Distribution for Tapered Wings with Partial-Span Flaps. T. R. 585, N. A. C. A., 1937
3. Sherman, Albert: A Simple Method of Obtaining Span Load Distribution. T. N. 732, N. A. C. A., 1939.
4. ANC 1 (1): Spanwise Air-Load Distribution. 1938.

8. NOMENCLATURE

S	Wing area, square inches
b	Span, inches
c	Chord, inches
$\bar{c}$	Average chord, inches ($= S/b$)
y	Distance of a particular station from centerline of wing, inches
C_L	Wing lift coefficient
c_{la_1}	Additional lift coefficient for a section when wing $C_L = 1.0$
c_{la}	Additional lift coefficient for a section ($= C_L c_{la_1}$)
c_{lb}	Basic lift coefficient for a section due to aerodynamic twist, when wing is operating at zero lift
c_l	Section lift coefficient ($= c_{la} + c_{lb}$)
α_{l_o}	Angle of attack of a section for zero lift, degrees
i	Angle between the chord line and the reference axis, degrees (see sketch on page 6)
β	Angle between the zero lift line and the reference axis, degrees (see sketch on page 6; note that $\beta = i - \alpha_{l_o}$)
α_R	Angle between reference axis and the wind direction, degrees (positive when the reference axis is so inclined to the wind direction as to produce positive lift, assuming (for this purpose only) that the reference axis acts as a zero lift chord line on airfoil section)
α_{R_o}	Angle between the wind direction and the reference axis when the wing is operating at zero lift, degrees
α_{a_s}	Angle between the zero lift line of a section and the wind direction, degrees ($\alpha_{a_s} = \alpha_R + \beta$)
a	Lift curve slope, C_L/degree
a_o	Section lift curve slipe, c_l/degree (slope of graph of c_l vs. α)

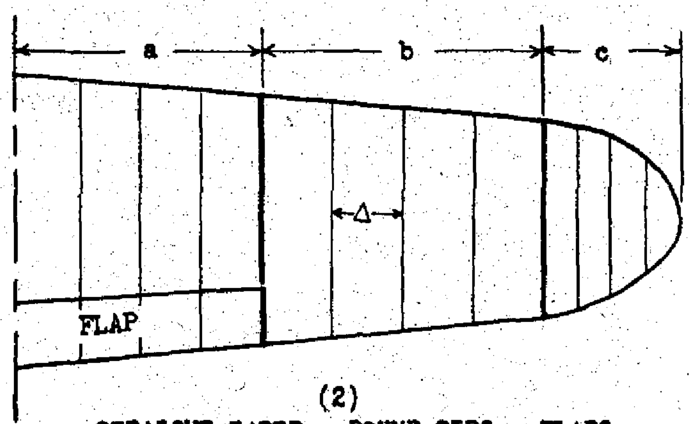


SUGGESTED DIVISIONS

a = 8 even parts

b = 4 even parts

(1)
STRAIGHT TAPER - ROUND TIPS

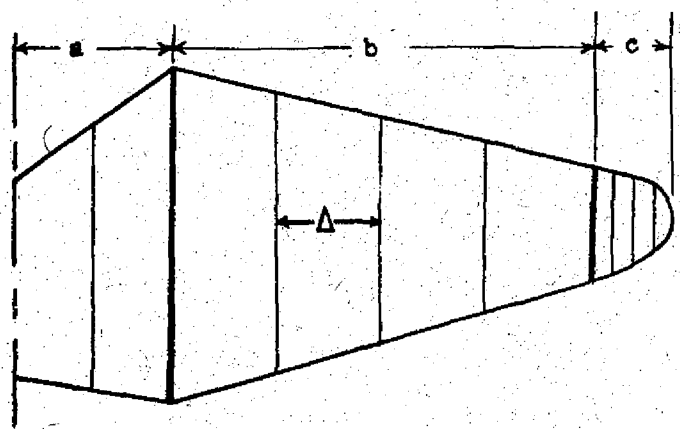


a = 4 even parts

b = 4 even parts

c = 4 even parts

(2)
STRAIGHT TAPER - ROUND TIPS - FLAPS



a = 2 even parts

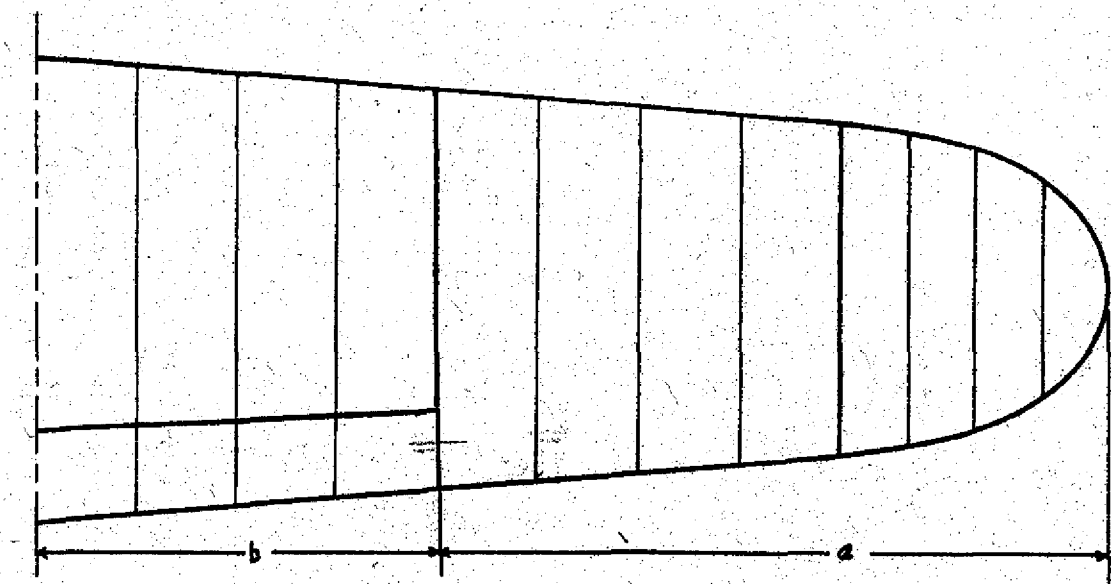
b = 4 even parts

c = 4 even parts

(3)
REVERSE TAPER - ROUND TIPS

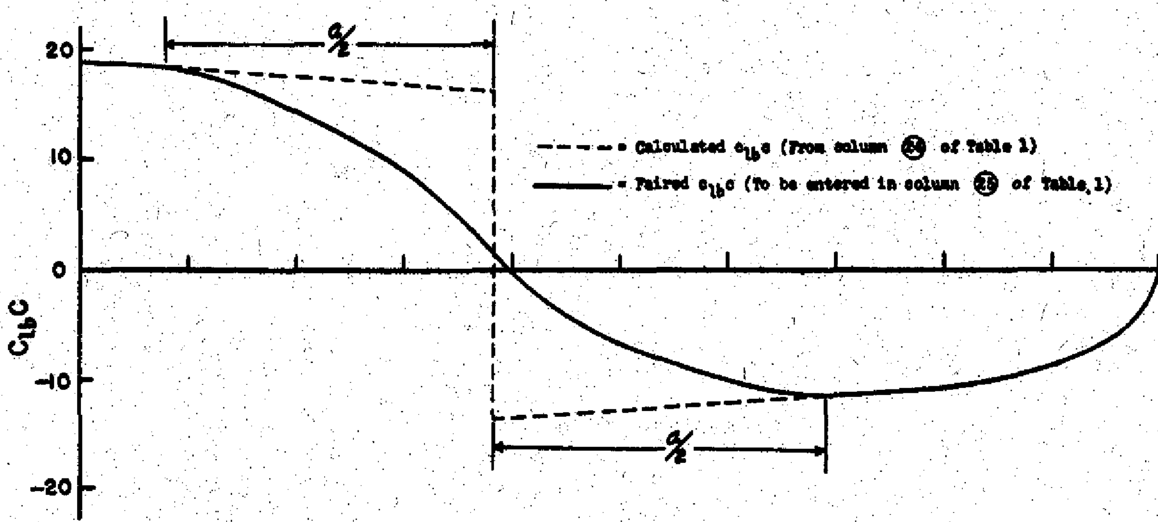
EXAMPLES OF WING PLANFORM DIVISION

FIG. I



TAPER RATIO = .625
ASPECT RATIO = 6

PLANFORM



FAIRING THE CURVE OF $C_{l_b}C$ VS. $\frac{y}{b/2}$

COMPARISON EXAMPLE FROM N.A.C.A. T.R. 585

FIG. 2

SPANWISE AIR-LOAD DISTRIBUTION

														Final c_{la}								Computed c_{lb}	Computed $c_{lb}c$	Faired $c_{lb}c$	Final c_{lb}
	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26
	$y/(b/2)$	c - ins.	a - ins.	Multiplier	② x ③ x ④	a_0 C_L per degree	*(1) ⑤ x ⑥	*(2) ⑥/a_0	$\frac{4\bar{c}}{\pi}$/②	①²	1 - ⑩	$\sqrt{⑪}$	⑨ x ⑫	⑧ + ⑬	⑭/2	*(3) ε	α_{lo}	⑯ - ⑰	*(4) ⑤ x ⑱	*(5) ⑦ x ⑱	⑥/2	⑱ + α_{R0}	*(6) ㉑ x ㉒	② x ㉓	*(7) $c_{lb}c$	㉕/②
Section From ℄ of Airplane to End of Flap	.000	102.00	23.04	.333	783.36	.100		1.000	.996	.000	1.000	1.000	.996	1.997	.998	.000	-8.00	8.00	4266.80		.050	3.70	.185	18.87	18.87	.185
	.0960	98.70	23.04	1.333	3032.16	.100		1.000	1.030	.009	.991	.995	1.025	2.025	1.012	.000	-8.00	8.00	24256.98		.050	3.70	.185	18.26	18.00	.182
	.1920	94.69	23.04	.667	1454.40	.100		1.000	1.074	.037	.963	.981	1.054	2.054	1.027	.000	-8.00	8.00	11435.58		.050	3.70	.185	17.41	14.70	.155
	.2880	91.02	23.04	1.333	2796.13	.100		1.000	1.117	.083	.917	.957	1.069	2.069	1.034	.000	-8.00	8.00	22,369.89		.050	3.70	.185	16.83	9.70	.107
	.3840	87.80	23.04	.333	674.31	.100		1.000	1.158	.147	.853	.924	1.070	2.070	1.035	.000	-8.00	8.00	5394.88		.050	3.70	.185	16.24	1.25	.014
Section From End of Flap to Beginning of Tip	.3840	87.80	21.96	.333	642.76	.100		1.000	1.158	.147	.853	.924	1.070	2.070	1.035	.000	-1.20	1.20	771.29		.050	-.310	-.155	-13.60	1.25	.014
	.4756	83.95	21.96	1.333	2458.08	.100		1.000	1.211	.226	.779	.880	1.066	2.066	1.033	.000	-1.20	1.20	2949.67		.050	-.310	-.155	-13.01	-5.75*	-.068
	.5670	80.40	21.96	.667	1177.36	.100		1.000	1.265	.321	.679	.824	1.042	2.042	1.021	.000	-1.20	1.20	1412.47		.050	-.310	-.155	-12.46	-9.25	-.115
	.6585	76.87	21.96	1.333	2250.75	.100		1.000	1.323	.434	.566	.752	.995	1.995	.997	.000	-1.20	1.20	2700.90		.050	-.310	-.155	-11.91	11.40	-.148
	.7500	73.38	21.96	.333	537.14	.100		1.000	1.386	.563	.437	.661	.916	1.916	.958	.000	-1.20	1.20	644.57		.050	-.310	-.155	-11.37	-11.37	-.155
Section From Beginning of Tip to Extreme Tip of Wing	.7500	73.38	15.00	.333	366.90	.100		1.000	1.386	.563	.437	.661	.916	1.916	.958	.000	-1.20	1.20	440.28		.050	-.310	-.155	-11.37	-11.37	-.155
	.8125	69.18	15.00	1.333	1383.60	.100		1.000	1.470	.660	.340	.583	.857	1.857	.928	.000	-1.20	1.20	1660.32		.050	-.310	-.155	-10.72	-10.72	-.155
	.8750	62.27	15.00	.667	622.70	.100		1.000	1.633	.766	.234	.484	.790	1.790	.895	.000	-1.20	1.20	747.24		.050	-.310	-.155	-9.65	-9.65	-.155
	.9375	49.51	15.00	1.333	990.20	.100		1.000	2.054	.879	.121	.348	.715	1.715	.857	.000	-1.20	1.20	1188.24		.050	-.310	-.155	-7.67	-7.67	-.155
	1.000	.000	15.00	.333	.000	.100		1.000	—	1.000	.000	.000	—	—	—	.000	-1.20	1.20	.000		.050	-.310	-.155	.000	.000	.000

Σ⑤ = 19,169.41 Σ⑦ =

Σ⑲ = 82,437.33 Σ⑳ =

*NOTES:

(1) Column ⑦ is calculated only when (a = variable)

(2) When (a = constant) the values in column ⑧ are 1

(3) See sketch page . The signs of the angles should be carefully observed.

(4) α_{R_0} of column ⑲ is calculated only when (a = constant)

(5) α_{R_0} of column ⑳ is calculated only when (a = variable)

(6) Values taken from Column ⑮ are final values of c_{lb} when no high-lift device is operating.

(7) Values of $c_{lb}c$ for Column ㉕ are taken from the faired curve of column ㉔.

CALCULATIONS FOR CONSTANTS

$b/2$ = 240 ins.

S (total wing area) = 2Σ⑤ = 2 X 19,169.41 = 38,338.82 sq. ins.

$\bar{c}$ (mean chord) = $\frac{\Sigma ⑤}{b/2} = \frac{19,169.41}{240}$ = 79.87 ins.

$\frac{4\bar{c}}{\pi} = \frac{4 X 79.87}{\pi}$ = 101.69 ins.

$\bar{a}_0 = \frac{\Sigma ⑦}{S\ b/2}$ = C_L per degree

$\alpha_{R_0} = \frac{-\Sigma ⑲}{S\ b/2} = \frac{82,437.33}{19,169} = -4.3$ degrees [This value of α_{R_0} used only when (a = constant)]

$\alpha_{R_0} = \frac{-\Sigma ⑳}{\Sigma ⑦}$ = degrees [This value of α_{R_0} used only when (a = variable)]

TABULATION FOR WING SHOWN IN FIG. 2

FIG. 3

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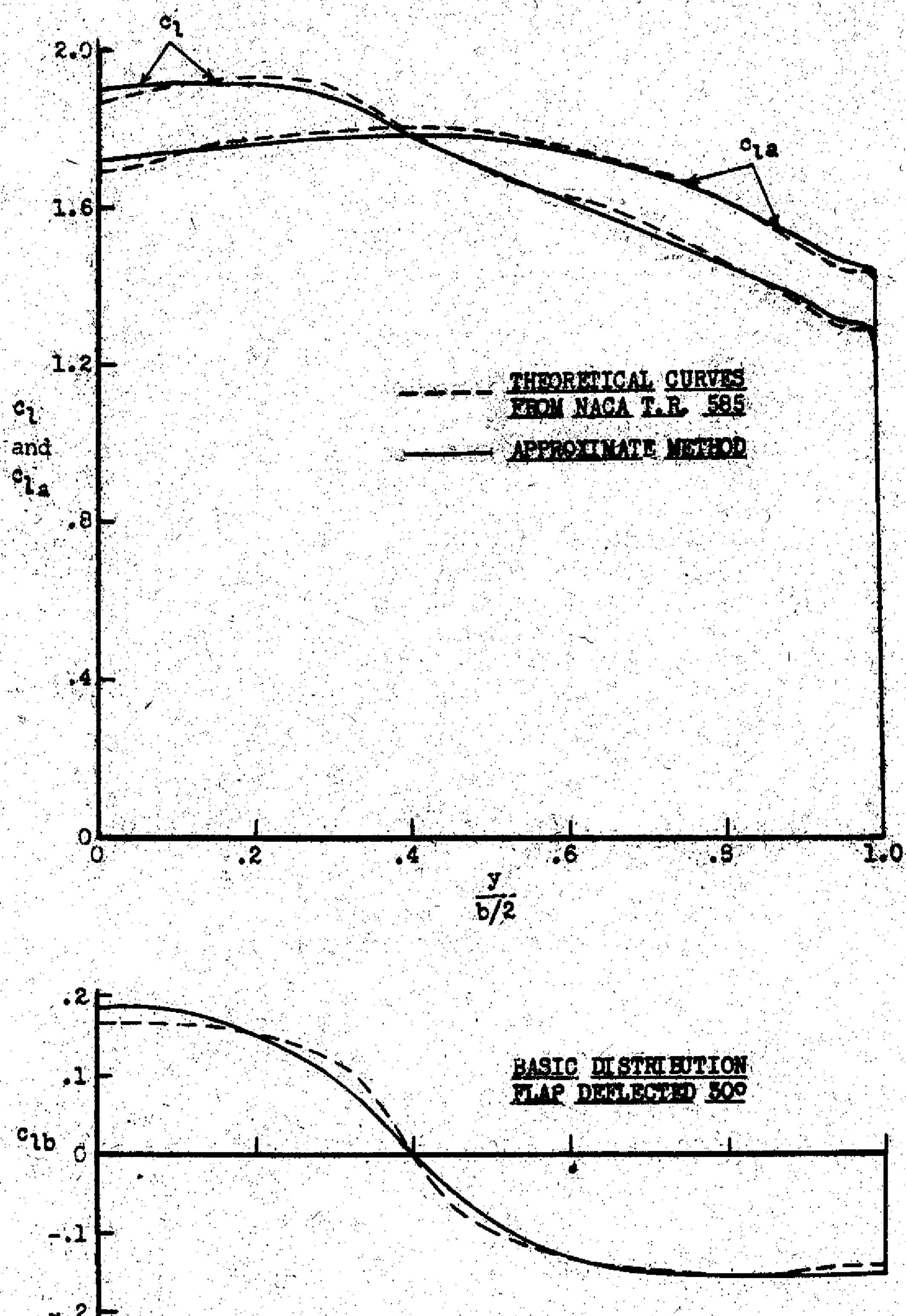
		Final c_{l_a}	Final c_{l_b}	Final c_l
1	2	3	4	5
$\frac{y}{b/2}$	c_{l_a}	$C_L \times$ 2	c_{l_b}	3 + 4
0	.998	1.72	.185	1.905
.0960	1.012	1.74	.182	1.922
.1920	1.027	1.77	.155	1.925
.2880	1.034	1.78	.107	1.887
.3840	1.035	1.78	.014	1.794
.4755	1.033	1.78	-.068	1.712
.5670	1.021	1.76	-.115	1.645
.6585	.997	1.71	-.148	1.562
.7500	.958	1.65	-.155	1.495
.8125	.928	1.60	-.155	1.445
.8750	.895	1.54	-.155	1.385
.9375	.857	1.47	-.155	1.315
1.000	.000	.000	.000	.000

EXAMPLE FROM N.A.C.A. T.R. 585
FINAL c_l DISTRIBUTION FOR A WING C_L of 1.72

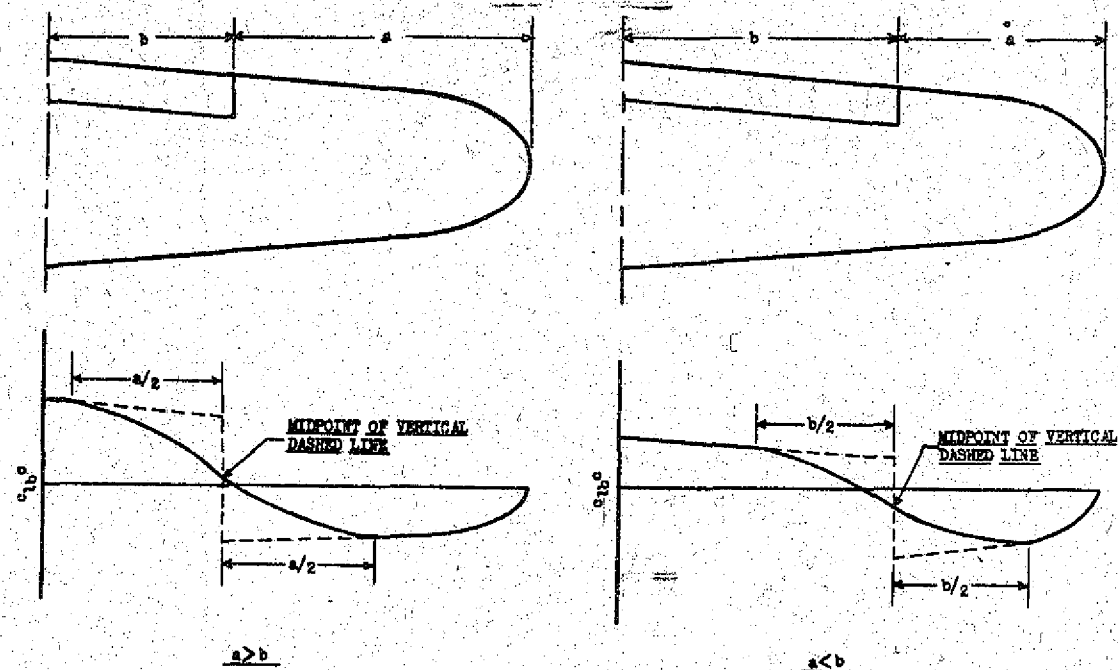
8-23020

FIG. 4

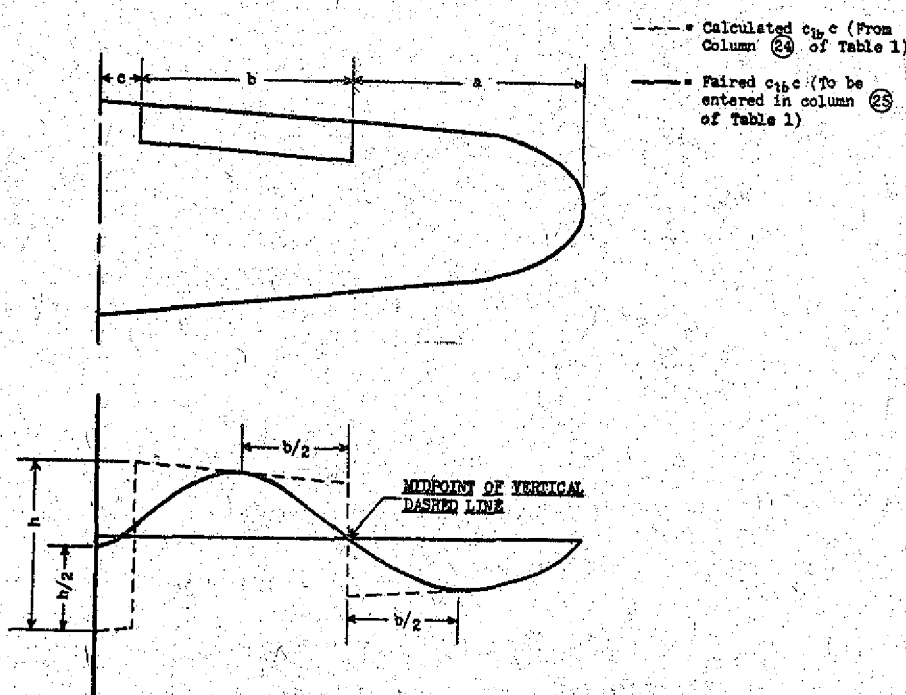


COMPARISON EXAMPLE FROM N.A.C.A. T.R.585

FIG. 5



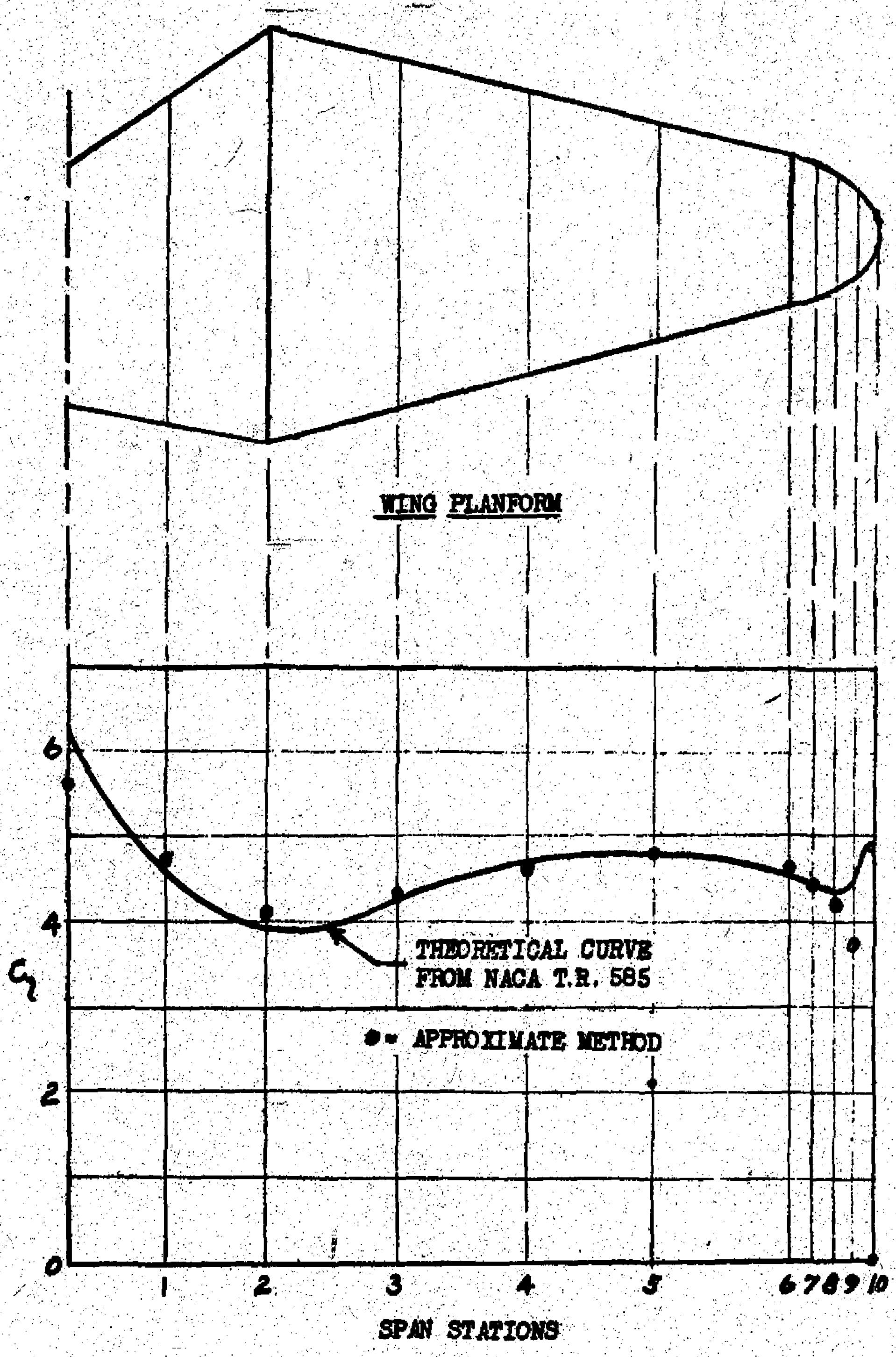
FLAP EXTENDING OUT FROM ℄



FLAP EXTENDING OUT FROM SIDE OF FUSELAGE

EXAMPLES OF FAIRING THE CURVE OF $C_{l_b}C$ VS $\frac{Y}{b/2}$

FIG. 6



WING WITH REVERSE TAPER
COMPARISON EXAMPLE FROM T.R. 585

FIG. 7

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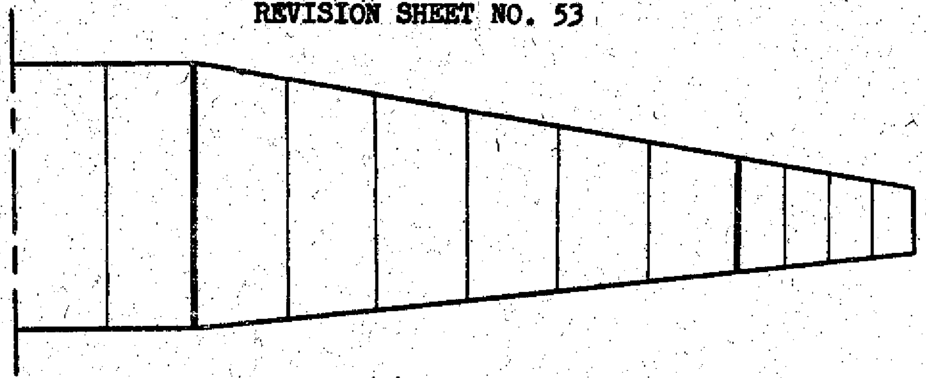
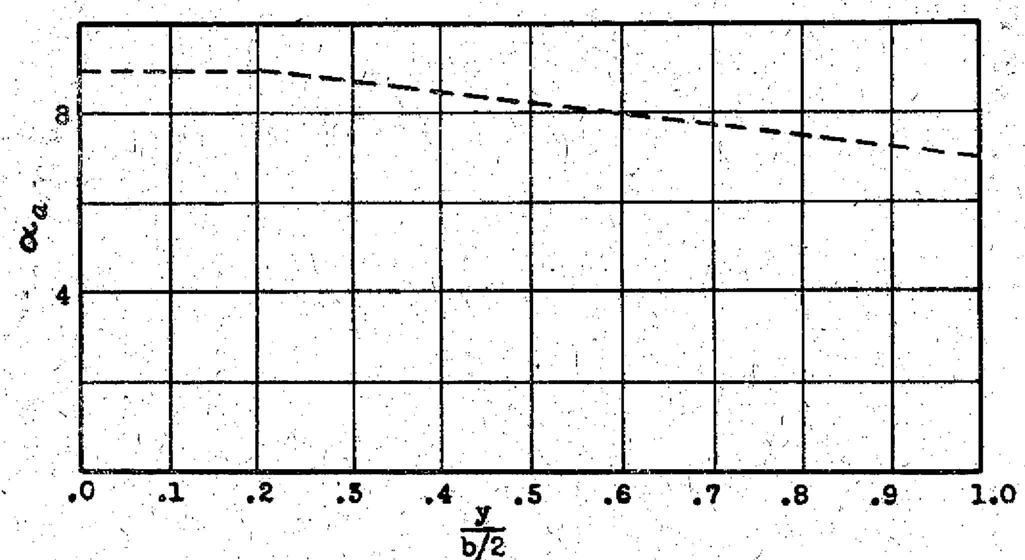
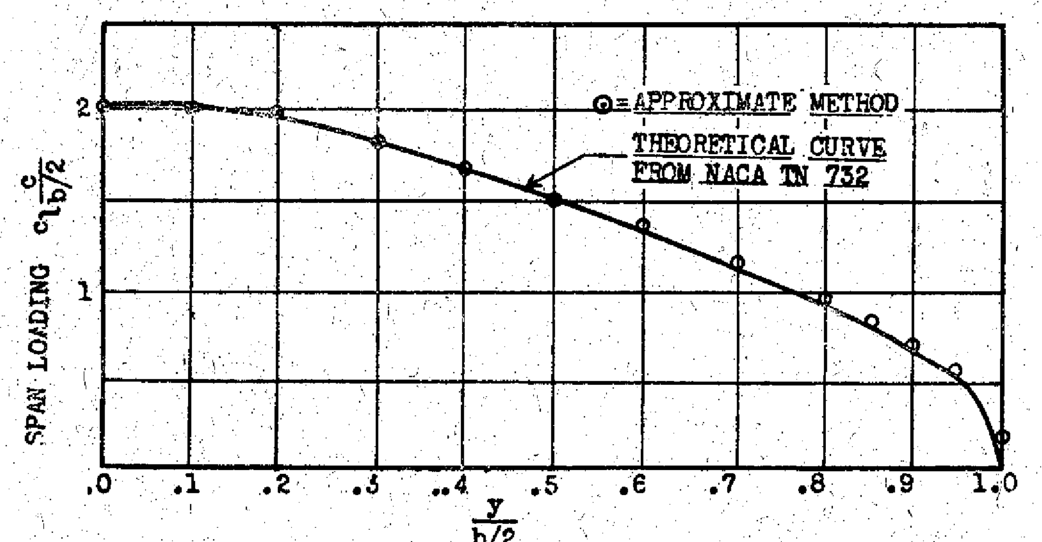


(a) PLANFORM



(b) GEOMETRIC ANGLE OF ATTACK ALONG SPAN
(FROM ZERO LIFT)



COMPARISON EXAMPLE FROM N.A.C.A.-T.N. 732

FIG. 8

CIVIL AERONAUTICS MANUAL
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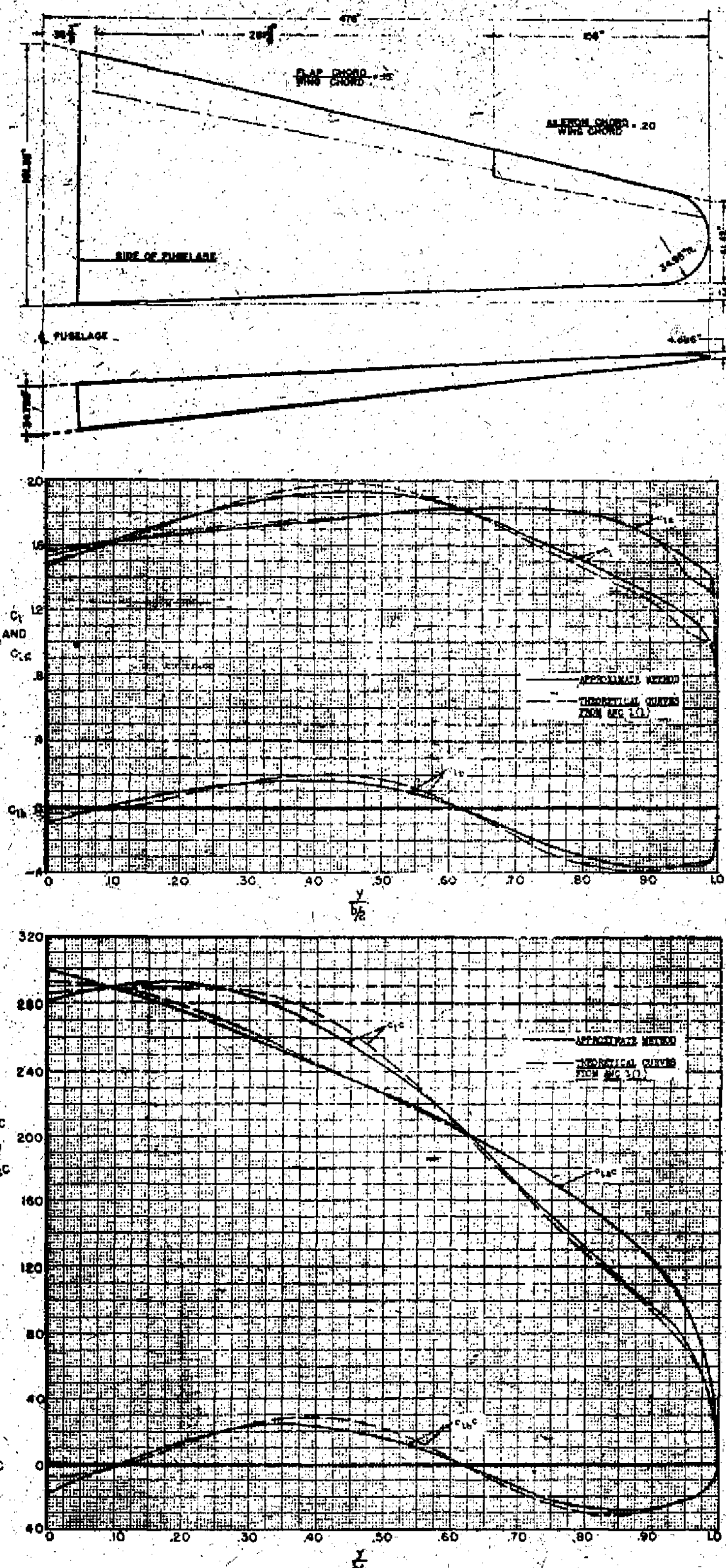

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COMPARISON EXAMPLE FROM ANC-1(1)

FIG. 9