U.S. DEPARTMENT OF COMMERCE

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Note: This bulletin will be revised at frequent intervals. New material will be available in the form of loose-leaf sheets to be added to or substituted for the present pages. New sheets will be mailed only to those requesting that their names be included on the mailing list for this service.

I N D E X

Chapter I - GENERAL

Section 1. Purpose and Use of This Bulletin.

(A) This bulletin is supplementary to Aeronautics Bulletin No. 7-A, Airworthiness Requirements for Aircraft, in which the structural and performance requirements for aircraft are outlined. The material contained herein is intended to explain and clarify the various requirements specified in Aeronautics Bulletin No. 7-A and to simplify and expedite the presentation of technical data required in connection with an application for an approved type certificate.

(B) The arrangement of chapters in this bulletin corresponds to that in Aeronautics Bulletin No. 7-A. Subjects are taken up in the approximate order in which they usually appear in the complete stress analysis. An effort has been made to reduce the work of preparing a stress analysis to a minimum, tables and charts being used wherever convenient.

(C) The use of the stress analysis methods outlined herein will substantially reduce the amount of time required for checking and should therefore expedite the final approval of the technical data. However, these methods are not required to be followed in detail and are presented only for the convenience of the airplane manufacturer. Any rational method of stress analysis which yields equivalent results will be accepted by the Department.

Section 2. Standard Symbols.

A - area, sq. ft. (Wing area unless otherwise noted).

A_D - equivalent drag area, sq. ft. (see Section 4).

A_E - equivalent flat plate area, sq. ft.

C - chord, feet; coefficient; constant; subscript, "chord".

C.P. - center of pressure, fraction of chord.

D - subscript "drag".

F - force, pounds.

HP - horsepower.

K - a general factor.

L - subscript "lift" or "level".

M - moment, ft. pounds; subscript, "moment".

M.A.C. - mean aerodynamic chord.

N - subscript, "normal force".

P - load, pounds.

a - position of aerodynamic center, fraction of chord; subscript "actual".

a.c. - aerodynamic center.

b - distance between spars, fraction of chord; span of wing.

c.g. - center of gravity.

d - drag loading, lbs/sq. ft.

e - unit wing weight, lbs/sq. ft.

f - unit stress, lbs/sq. in.; front spar location, fraction of chord; subscript, "fuselage".

g - acceleration of gravity (= 32.2 ft./sec.^{2}); subscript "gliding".

h - distance measured perpendicular to M.A.C., in terms of M.A.C.

i - subscript "induced".

j - position of wing c.g., fraction of chord.

k - factor of safety.

m - slope of lift curve, ΔC_L/radian; moment divided by W; subscript, "maximum vertical".

n - applied load in terms of W. (net value equals acceleration factor)*.

o - subscript, "zero lift"; "initial"; "standard sea level".

p - power loading, lbs./HP.

* Without subscript, n refers to an applied load normal to the basic wing reference chord.

Sec. 2

R - resultant force or reaction, lbs.; aspect ratio; subscript, "resultant".

T - tail load, pounds.

U - gust velocity, ft./sec.

V - airplane speed, ft./sec.

W - gross weight, lbs.

q - dynamic pressure, lbs./sq.ft.

r - rear spar location, fraction of chord.

s - wing loading, lbs./sq.ft.; subscript, "stall".

t - subscript "tail".

v - airplane speed, miles per hr.

w - unit pressure, lbs./sq.ft.; subscript, "wing".

$\bar{w}$ - average unit pressure, lbs./sq.ft.

x - distance measured parallel to M.A.C., in terms of M.A.C.; subscript. *

α - (alpha) - angle of attack, radians or degrees.

β - (beta) - flight path angle, degrees.

Δ - (delta) - increment.

η - (eta) - propeller efficiency.

ρ - (rho) - mass density of air.

* With subscript "x", n refers to an applied load parallel to the basic wing reference chord. (See Figure 13).

Section 3. Standard Values and Formulas.

(A) The following standard values and formulas are useful in converting flight conditions into structural loading conditions.

Air Density:

1. ρ_0 = .002378 slugs (lbs./32.2) per cu. ft. (standard sea level value).

Dynamic Pressure:

2. $q = \frac{1}{2} \rho_0 V_i^2$

$= .00119\ V_i^2$ (where V_i is "indicated" speed, ft. per sec.)

$= .00256\ v_i^2$ (where v is "indicated" speed, m.p.h.)

Basic Airplane Parameters:

3. $s = W/A_W$

4. $p = W/HP$

5. $d = W/A_D$ (when V_L is known, d may be computed from eq. 16).

Aerodynamic Coefficients:

6. $C_R = (C_L^2 + C_D^2)^{\frac{1}{2}}$

7. $C_N = C_L \cos \alpha + C_D \sin \alpha$

8. $C_C = - C_L \sin \alpha + C_D \cos \alpha$ (positive rearward)

9. $C_{M_x} = C_N (x - C.P.)$ (Where x is the distance, from the leading edge, of the point on the chord about which the moment is computed, expressed as a fraction of the chord).

Forces, Unit Loadings, and Couples:

10. $F_x = C_x A q$ (Where x may be R, L, D, N, C, or M)

11. $F_D = A_D q$

12. $M = F_M C$ (torque or couple)

$= C_M A q C$ (torque or couple)

13. $\bar{w} = C_N q$

14. $n = F/W$

15. $F_{pr} = 550\,\eta\,HP_a/V_a$ (propeller thrust, pounds)

Speeds:

16. $V_{L_a} = 77.3\ (\eta\ d/p_a)^{\frac{1}{3}}\ (\rho_o/\rho_a)^{\frac{1}{3}}$ (ft. per sec.) = actual air speed at air density ρ_a

17. $V_s = 29\ (s/C_{L\ max})^{\frac{1}{2}}$ (ft. per sec.) = indicated stalling speed.

18. $V_m = 29\ (d)^{\frac{1}{2}}$ (ft. per sec.) = indicated theoretical maximum vertical speed.

(NOTE: The value of "d" should be the same as that used in, or determined from, Eq. 16.

19. $V_i = V_a(\rho_a/\rho_o)^{\frac{1}{2}}$ where V_i = indicated air speed.
V_a = actual air speed.
ρ_o = standard density of air at sea level.
ρ_a = density of air in which V_a is attained.

20. $\Delta C_L = m\ (U/V)$ = change in C_L due to gust.

21. $\Delta n = \Delta C_L\ (q/s)$ = change in applied load factor due to gust.

Section 4. General Aerodynamic Information.

(A) Definitions. (See also Aeronautics Bulletin No. 7-A, Secs. 11 and 12).

(1) Aerodynamic Center - The point on the wing chord, expressed as a fraction of the chord, about which the moment coefficient is substantially constant for all angles of attack. The theoretical location is at 25 percent of the chord. The actual location may differ from the theoretical location and may be determined from the slope of the moment coefficient curve as outlined in Sec. 7 (B)(3).

(2) Drag Area - The area of a hypothetical surface having an absolute drag coefficient of 1.0.

(3) Equivalent Drag Area - The drag area which, at a given value of dynamic pressure, will produce the same aerodynamic drag as the body or combination of bodies under consideration. (Note: $A_D = 1.28\ A_E$, where A_E is the equivalent flat plate area).

(B) Coefficients.

(1) The coefficients used in Aeronautics Bulletin No. 7-A are absolute (non-dimensional) coefficients. When applied to an airfoil surface of given area they represent the ratio between an actual average unit pressure referred to the projected area of the airfoil and the dynamic pressure corresponding to the flight condition being considered. The subscripts denote the direction along which the force is measured, but do not change the basic reference area.

(2) The subscripts "L" and "D" refer to directions normal to and parallel to the relative wind, while the subscripts "N" and "C" refer to directions respectively normal to and parallel to the basic wing chord. Subscript "R" refers to the direction of the resultant force. These vectors are illustrated in Figures 1 and 2 . When the planes of the drag truss and lift trusses do not coincide respectively with the planes of the basic chord and the plane of the normal forces, a correction is necessary before the coefficients can be used directly in the wing analysis method outlined in Chapter III. The corrected coefficients are obtained by resolving the resultant force coefficient into components in the planes of the lift truss and drag truss, as shown in Figure 3 . The effect on the chord coefficient may be considerable, but the correction for C_N will usually be negligible.

(3) The moment coefficient may be considered to be of the same nature as the force coefficients if the force to which it corresponds is applied as a couple at the leading and trailing edges of the wing chord, as shown in Figures 1 , 2 and 3. A positive moment coefficient requires an upward force at the leading edge, as shown. The conversion of center of pressure position into a moment coefficient about any given point can be easily accomplished by means of equation 9 in Section 3 . It should be noted that the center of pressure and the moment coefficient are alternative in nature and can not both be used at the same time.

(4) The standard airfoil characteristics are obtainable for conventional airfoils in the form of N.A.C.A. Reports and Technical Notes. For stress analysis purposes certain corrections are usually required. The detailed procedure for making such corrections is outlined in Section 7.

Section 5. General Presentation of Data for Airplane Approval.

(A) Application for Approved Type Certificate. The following information is often incorrectly or incompletely supplied in making application for an approved type certificate. Careful attention to these details will prevent delays from this source:

(1) The model number of both propeller and hub should be supplied. If the propeller has an approved type certificate the number should be inserted on the application blank. If the propeller (or hub) is not approved, the airplane manufacturer should request the propeller manufacturer to submit the necessary data to the Department for approval and should inform the Department that this has been done.

(2) Approved wheels and tires, or floats, are required and such components should be given their proper model number or size in the application blank. If not approved, or if approved for a lower load than that for which they are to be used, steps should be taken by the airplane manufacturer as outlined in Paragraph (1).

(3) The shock absorber should be designated by an identifying model number. Shock absorbers are approved for each individual installation, involving the type and geometry of landing gear used, size of tire, and the weight of the airplane. Unless <u>all three of these factors</u> are substantially the same in two different cases, approval of a shock absorber for one case does not extend to the second case. When the shock absorber is to be supplied by a different manufacturer the responsibility for the approval of the shock absorber should be definitely determined and the Department should be informed of the arrangement agreed upon.

(4) The proper model number of the engines used should be furnished. When the engines to be used have not been approved by the Department, the engine manufacturer should be requested to submit the necessary data to the Department.

(5) One copy of the application for approved type certificate should be notarized. Each copy should be completely filled out, listing all the data required under each item.

(B) <u>Technical Data</u> - Attention to the following list of frequently omitted items will be of assistance in expediting the work of the Department:

(1) Drawing lists.

(2) Lists of standard equipment.

(a) Items such as safety belts, position lights, and special throttle controls, which require individual approval, should be specified by model number and the name of the manufacturer.

(3) Complete dimensions and references to standard components on three-view drawings.

(4) Signature of engineer responsible for the stress analyses.

(5) Adequate material specifications on all shop drawings.

(6) Location and details of control system pulleys and brackets; location and nature of control surface stops.

(7) Drawings to show provision for expansion space in oil tanks.

(B) Adequate drawings and descriptions of the operation of unconventional mechanisms such as flap, tab, and retractable landing gear control devices.

(C) Revisions. In submitting data for approval of revisions to an approved type airplane the pertinent corrected pages of the drawing lists should be attached. Failure to state the particular models to which the revisions apply may result in undesirable delays. The serial numbers of the airplanes to which the revisions are to apply should be given in all cases where some existing airplanes are not to be changed. Alternate installations should be so designated and properly indicated on the drawing lists.

Checking of revised drawings of relatively large size will be expedited if the change letters are also printed in two perpendicular margins opposite the letter on the drawing.

Chapter II - BASIC STRESS ANALYSIS DATA.

Section 6. General Design Data.

(A) For reference purposes, it is desirable to specify the following general information at the beginning of the stress analysis:

(1) W = maximum gross weight, pounds.

(2) A = effective wing area, square feet. This should agree with the requirements of Section 12 (A), Aeronautics Bulletin No. 7-A and can be determined graphically as a part of the procedure outlined in Section 9 of this bulletin.

(3) HP = total rated horsepower. This should be based on the approved output for the engines used. If altitude engines are employed, the altitude at which the rating applies should be stated.

(4) A_D = estimated total drag area at high speed, sq. ft. See Section 4 for definition. When the value of V_L is known or has been estimated, A_D can be determined by solving equation 16 for d. When it is desired to estimate A_D first in order to compute the value of V_L, the equation $A_D = A_{Df} + C_D A_W$ can be used. Typical values for A_{Df} (drag area of airplane less wing) are given on Figure 4. C_D can usually be assumed to be the minimum drag coefficient, which is approximately 0.01 for conventional airfoils.

(5) s = wing loading, lbs./sq.ft. (Equation 3) (Based on maximum gross weight).

(6) p = power loading, lbs./HP. (Equation 4) (Based on total rated power).

(7) d = drag loading, lbs./sq.ft. (Equation 5) (Based on maximum gross weight).

(8) V_L = estimated or measured indicated high speed in level flight, ft./sec. When the maximum actual value of V_L occurs at a certain altitude, the corresponding indicated value is obtained from equation 19, Section 3. The actual air speed at any altitude may be estimated by the use of equation 16, in which the value of P_a should be based on the power available at that altitude. See also Aeronautics Bulletin No. 7-A, Section 11 (H) and Section 14.

(9) V_S = calculated stalling speed, ft./sec. This speed can be determined from equation 17, Section 3. When wing flaps or similar high-lift devices are used, the stalling speed should be calculated for the two extreme flap positions used.

(10) V_m = maximum vertical velocity (with zero propeller thrust). See equation 18, Section 3, and Aeronautics Bulletin No. 7-A, Section 14 (C) (1).

(11) V_g = design indicated gliding speed, ft./sec. To be determined by the manufacturer in accordance with the type and purpose of the airplane. See Aeronautics Bulletin No. 7-A, Section 14-(C) for minimum value. The equation for the minimum value of V_g is designed to provide for the following features:

(a) Probability of exceeding the high speed in level flight. (V_g can never be less than V_L).

(b) Effect of cleanness and weight on the gliding speed which can be attained at a given gliding angle. Both these quantities are included in the term V_m. Propeller drag at terminal speed is not allowed for as the formula will not give values of V_g high enough to cause the propeller thrust to reverse in direction.

(c) Influence of airplane size on the maximum speed likely to be used. The factor K_g is an empirical factor based on the weight of the airplane. Its purpose is to provide higher design gliding speeds for small, highly-maneuverable airplanes.

Section 7. Determination of Corrected Airfoil Characteristics.

The basic airfoil characteristics as presented in the standard form must usually be corrected and several additional coefficients should be plotted for use in the stress analysis. Simplified equations are outlined below for this purpose and Table II has been compiled to facilitate the numerical work. The results should be replotted in a convenient form such as that shown in Figure 5, where C_L is used as the basic coefficient, instead of angle of attack.

(A) Effects of Aspect Ratio. The methods of correcting for aspect ratio are well defined and are outlined in various text books and reports. The following equations may be used in this connection:

(1) $R = (kb)^2 / A$

Where R = aspect ratio,
k = Munk's span factor for biplanes, (for monoplanes $k = 1.0$)
b = span of longest wing,
A = total wing area, including any portion replaced or blanketed by the fuselage.

(2) $K = \frac{1}{R} - \frac{1}{R_6} = \frac{1}{R} - 0.1667$

Where K = correction factor

(3) $\alpha = \alpha_6 + 18.24\ KC_L$ (Items 2 to 4, Table II)

Where α_6 = angle of attack (degrees) for a given C_L when aspect ratio is 6,

α = angle of attack (degrees) for same C_L when aspect ratio is R.

(4) $C_D = C_{D6} + 0.318\ K\ C_L^2$ (Items 5 to 7, Table II)

Where C_{D6} = C_D for given C_L when aspect ratio is 6.
C_D = " " " " " " " " R.

(5) $m = m_6 \left(\frac{4}{3 + 6/R}\right)$

Where m_6 = slope of lift curve when aspect ratio is 6.
m = " " " " " " " " R.

(B) <u>Computation of Additional Characteristics.</u> As indicated in Table II, certain additional characteristics are desirable and they may be determined as follows:

(1) The normal force coefficient, C_N, can be determined from Equation 7, Section 3. The steps involved are shown as items 8 to 12 of Table II. It will be found that C_N is almost identically equal to C_L for small values of the latter. This may not be true, however, for unconventional or modified airfoils, such as those equipped with flaps.

(2) The chord force coefficient, C_C, is determined from Equation 8, Section 3. The steps are outlined as items 13 to 15 of Table II.

(3) The moment coefficient about the aerodynamic center, C_{M_a}, is not usually given in airfoil reports. In some cases the moment coefficient about the quarter chord point $C_{M_{C/4}}$ is plotted against C_L. In such cases a straight line can be drawn to fit the $C_{M_{C/4}}$ curve as closely as possible. See Figure 5. The average value of C_{M_a} is then obtained from the straight line where $C_L = 0$. The position of the aerodynamic center can then be obtained by the following equation:

$$a = .25 - (C_{M_1} - C_{M_a})$$

Where C_{M_1} is the value given by the straight line for $C_{M_{C/4}}$ where $C_N = 1.0$

(4) The values of a and C_{M_a} can also be obtained directly from C.P. curves as outlined in steps 16 and 17 of Table II, in which the values of $C_{M_{C/4}}$ are determined. These values can be plotted against C_L and the process for determining a and C_{M_a} can then be carried out as outlined in Paragraph (3) above. In any case, the operations should be confined to the values of C_L which lie on the substantially straight portion of the lift coefficient curve.

(5) The value of C_{M_a} can be separately determined for any given value of C_L by means of the equation:

$$C_{M_a} = C_{M_{C/4}} + (a - .25)\ C_N.$$

It may be advisable to plot these values for unconventional airfoils which do not have a well-defined aerodynamic center. Provision is made under item 18 of Table II for determining local values of C_{M_a}.

(C) Extension of Characteristic Curves. In the accelerated flight conditions it is possible to closely approach or exceed the maximum value of C_L shown on the basic airfoil characteristic curves without the breakdown of the flow characterized by the change in slope of the lift curve. The curves to be used for stress analysis purposes can be extended to represent the effect of a sudden change in angle of attack by the following approximations:

(1) Referring to Figure 5, extend the curve of angle of attack, α , to higher values of C_L by means of a straight line coinciding with the substantially straight portion of the original curve. The values of α so obtained should be entered in Table II under item 4. (The dotted lines in Figure 5 indicate extended values).

(2) Determine the induced drag coefficient as outlined in item 19 of Table II. R and K are defined in Paragraph (A)(1) and (2).

(3) Determine the profile drag coefficient C_{D_o}: item 20 of Table II. Plot these values for the original straight portion of the C_L curve and extend the curve so obtained along the same general path followed at the lower values of C_L, as shown in Figure 5. Enter the values of C_{D_o} thus obtained under item 20.

(4) Extend the C_D curve by determining the values for item 7 of Table II, as indicated.

(5) The C_{M_a} curve can be extended as a horizontal straight line.

(6) The extended values of C_N and C_C are determined as indicated under items 8 to 15 of Table II, using the extended values of C_D.

(7) The C.P. values should be extended by means of the equation

$$C.P. = a - C_{M_a}/C_N$$

using the extended values of C_N.

(D) Biplane Effects. The effects of biplane interference can be conveniently accounted for by a suitable modification of the corrected airfoil characteristic curves illustrated in Figure 5. The modification of the various characteristics for each wing can be carried out as follows, referring to Table II:

(1) Lift Coefficients. The individual lift coefficient for each wing should be determined for the useful range of average lift coefficient, C_L, (Item 1 of Table II). N.A.C.A. Report No. 458, "Relative Loading on Biplane Wings", can be used for this purpose. This method derives increments which are added to and subtracted from the average lift coefficient. Items 21 to 24 are provided in Table II for this purpose. When this method is applied to wing arrangements incorporating considerable overhang, further corrections may be required. A special ruling should be requested in such cases, pending the publication of a suitable report on this subject.

(2) Normal Force Coefficients. The corrected normal force coefficients for each biplane wing are plotted on Figure 5. These values can be determined from the original curve of average normal force coefficient by using the corrected values of C_L under items 23 and 24, Table II.

(3) General Characteristics. It is not necessary to plot the remaining characteristics for each biplane wing as they can be readily determined by the following method. Given a design value of the average C_N, the corresponding points on the C_N curves for each wing are determined. The individual values of biplane C_L corresponding to the biplane C_N are determined by horizontal lines intersecting the average C_N curve. The various coefficients for each wing are then determined for these values of C_L, as indicated by the vertical dotted lines on Figure 5.

Section 8. Determination of Center of Gravity Positions.

(A) An outline drawing should be made, to a suitable scale, showing the actual location of the center of gravity of each item or group of items. It is desirable to designate each item by a number on the drawing. The horizontal and vertical reference axes used in computing the moment arms of the items should be shown on the drawing. This drawing should also indicate the location of the mean aerodynamic chord and the mean aerodynamic center (see Sec. 9). These should be referred, by suitable dimensions, to convenient reference points which can be easily located on the completed airplane. For instance, the centerline of a main fitting bolt or a similar definite point can be used for this purpose. This will facilitate final checking of the center of gravity locations from the flight test report.

(B) A balance table should be prepared, specifying the weight and moment arm of each item with reference to the horizontal and vertical reference axes.

(C) The balance table should also include the summation of the products of the weights and distances, from which the following center of gravity positions should be determined.

(1) Airplane fully loaded.
(2) Airplane less landing gear (or floats).
(3) Most forward c.g. location for which approval is desired.
(4) Most rearward c.g. location " " " " "

Section 9, Determination of Resultant Air Forces.

(A) A general method will be outlined for determining the mean effective value of the normal force coefficient, the average moment coefficient, location of the mean aerodynamic center and value of the mean aerodynamic chord. These factors are needed in order to determine the balancing loads for various flight conditions. The most general case will be considered, so that certain steps can be omitted when simpler wing forms or span load distribution curves are involved.

(B) In general, the summation of all the forces acting upon a wing can be expressed as a single resultant force acting at a certain point, and a couple or moment of air forces about this point. If the point is chosen such that, at constant dynamic pressure, the moment of the air forces does not appreciably change with a change in the angle of attack of the airfoil, the point can be considered as the mean aerodynamic center of the wing. The resultant force can be resolved into the normal and chord components and represented by the average coefficients C_N and C_C, while the moment is represented by the average moment coefficient, C_M, times a distance which can be considered to be the mean aerodynamic chord. The values of the above quantities and the location of the mean aerodynamic center will depend on the plan form of the wing and the type of span distribution curve assumed.

(C) For convenience and clarification, Table III has been developed and the various curves obtained as a part of this method are illustrated in Figures 6, 7, and 8. It should be particularly noted that when the area under a curve is referred to, the area should be expressed in the items to which the curve is drawn, not in the actual units of measurement. The procedure is as follows:

(1) Figure 6 (a) illustrates the actual wing plan form, plotted to a suitable scale. This should agree with the definition of effective area outlined in Section 12 (A) of Aeronautics Bulletin No. 7-A.

(2) Figure 6 (b) shows the variation of wing chord, C, with span. The values of C are entered in Table III as item (2). The area of the figure should be accurately determined and converted to the proper units. It should be one-half the value of effective wing area specified in Section 6 (A), item (2).

(3) Figure 6 (c) represents an assumed span distribution curve. The factor R_b represents the ratio of the actual C_N at any point to the value of C_{N_o} at the root of the wing. Values of R_b from this curve are entered in Table III under item (3). The span distribution curve used in the analysis should correspond to the requirements of Aeronautics Bulletin No. 7-A, Section 20 (B).

(4) Figure 6 (d) is obtained by plotting R_bC (item 4), Table III) against span. The ordinates of this curve are proportional to the actual force distribution over the span. The area under curve (d) should be accurately determined and expressed in the proper units. K_b, the ratio of the mean effective C_N to the value of C_{N_o} (at the root) is obtained by dividing the area under curve (d) by the area under curve (b), using the same units of measurement for each area. This value of K_b is indicated by the dotted line on curve (c).

(5) To determine the location of the mean aerodynamic center along the span, Figure 7 (a) is drawn. The ordinates are obtained by multiplying the ordinates of curve 6 (d) by their distance along the span, as shown in item 5, of Table III. The area under curve 7 (a), divided by the area under curve 6 (d), gives the distance from the wing root to the chord on which the mean aerodynamic center of the wing panel is located. This distance is indicated on Figure 6 (a) by the dimension $\bar{b}$.

(6) The locus of the aerodynamic centers of each individual wing chord is plotted on Figure 6 (a) as the dotted line A-B. In Table III, the distances x from the base line O-E to the line A-B are entered under item (6).

(7) Figure 7 (b) is now plotted, using as ordinates the values of R_b Cx obtained from item 7 of Table III. The area under curve 7 (b) divided by the area under curve 6 (d) gives the distance of the mean aerodynamic center from the base line O-E, of Figure 6 (a). This distance is indicated as $\bar{x}$ on that figure.

(8) If it is assumed that the moment coefficient about the aerodynamic center of each individual chord is constant over the span, the magnitude of the mean aerodynamic chord is determined by means of Figure 7 (c). The ordinates for this curve are determined from item 8 of Table III. The area under 7 (c) divided by the area under curve 6 (c) gives the value of the mean aerodynamic chord. By way of illustration, it is drawn on Figure 6 (a) so that its own aerodynamic center coincides with the location of the mean aerodynamic center of the wing panel.

(9) In cases in which wing flaps or similar devices are used over a portion of the span it is desirable to obtain the mean effective moment coefficient. This is the coefficient to be used for balancing purposes in connection with the mean aerodynamic chord previously determined under the assumption of a uniform moment coefficient distribution. In Table III under item 9 the local values of the moment coefficient about the aerodynamic center are entered. These are also plotted as Figure 8 (a) to illustrate a type of distribution which might exist.

(10) Figure 8 (b) is plotted from the values indicated under item 10 of Table III. The area under this curve divided by the area under curve 7 (c) gives the mean effective value of the moment coefficient for the entire wing panel.

(11) It should be noted that the above method for determining the position of the mean aerodynamic center is based on the assumption that the wing is not twisted. For a twisted wing, a different span distribution would exist for each angle of attack. Under such conditions there is no single point about which the moment of the air forces will remain constant at all angles of attack. The location of the resultant forces can, however, be determined in the above manner for any known span distribution.

Section 10. Resultant Forces on Biplanes.

(A) The mean aerodynamic center location and the value of the mean aerodynamic chord for each wing panel can be found as outlined in Section 9. When wing flaps are used the mean effective moment coefficient for each wing panel should also be obtained. For a given flight condition, the values of C_N for each wing can be determined from Figure 5. The location of the mean aerodynamic center of the biplane and the determination of the resultant forces and moments can be accomplished as follows, referring to Figure 9:

(1) The mean aerodynamic center of the biplane cellule lies on a straight line connecting the mean aerodynamic centers of the two wing panels. The location on the line is determined from equation (a), Figure 9.

(2) Assuming that the mean effective moment coefficient is the same for each wing panel, the value of the mean aerodynamic chord for the biplane is determined from equation (b), Figure 9.

(3) If the mean effective moment coefficients for the two wing panels are different in value, the effective moment coefficient for the biplane can be determined from equation (c), Figure 9.

(B) The mean aerodynamic center of a biplane, as determined in paragraph (A), is based on the relative values of the normal forces acting on each wing. When the average normal force coefficient for the entire biplane is near zero, the relative loading on the wings varies over a wide range and the mean aerodynamic center, if determined as outlined above, would in some cases lie entirely outside of the wing cellule. For the same conditions, however, the chord force coefficients for the wings would be nearly equal, so that the resultant chord force would not act at the same point as the resultant normal force. As the location of the mean aerodynamic center is of interest mainly in balancing and stability computations, the following approximations and assumptions are permissible:

(1) A single nominal location can be assumed for the mean aerodynamic center for all the balancing conditions.

(2) When two different span distributions are required, the more nearly constant span distribution may be used in determining the nominal mean aerodynamic center and M.A.C. (In the requirements now outlined in Aeronautics Bulletin No. 7-A a constant span distribution may be assumed for this purpose).

(3) The computations may be made for an average value of $C_N = 0.5$, unless the biplane has an unusual amount of stagger, decalage, or is otherwise unconventional.

(4) When the use of a single location for the aerodynamic center is not sufficiently accurate, the computation of the mean aerodynamic center for the entire biplane should be omitted and in balancing the airplane each wing should be treated as a separate unit.

Section 11. Design Flying Conditions.

(A) The basic design flying conditions are outlined in Aeronautics Bulletin No. 7-A, Section 15. These conditions are used as a basis for the determination of the external loads acting on the entire airplane. For convenience the flight conditions are graphically illustrated in Figure 10. This chart can be duplicated as part of the stress analysis, replacing the symbols and equations by the actual values used. Such a procedure will facilitate checking and the examination of revised data.

(B) Condition I. (Sec. 15(B) Aeronautics Bulletin No. 7-A). This condition is illustrated graphically in Figure 11. It is primarily designed to represent the conditions at which the highest positive acceleration or load factor is likely to be obtained. The values of acceleration specified in Section 15(B), Aeronautics Bulletin No. 7-A, are based on two separate possibilities. The first equation, (a), represents the approximate acceleration developed in encountering a sharp-edged gust of 30 feet per second while flying at the speed V_L. The second equation, (b), is semi-empirical and is based largely on past experience, as explained in Section 13 of Aeronautics Bulletin No. 7-A. It represents the highest applied acceleration which is to be expected during maneuvers.

(1) As it is possible to develop the applied load factor for Condition I in various flight attitudes, a definite range of values of C_L is included, as indicated in Figure 11. This corresponds to the assumption that the applied load factor will be developed at speeds somewhat below the high speed V_L, the lowest speed being that associated with the value of $C_{L_{max}}$. The modified flight conditions, which are explained in detail in succeeding sections, are intended to provide for the effects of this assumption and are so specified as to require a minimum amount of investigation.

(2) It will be noted that in Condition I a value for the C.P. is specified, instead of the moment coefficient. If it is desired to find the moment coefficient to be used in Condition I, the values of $C.P._I$ and C_{N_I} can be inserted in equation 9, Section 3. In the case of a biplane, the proper correction should first be made to the upper wing C.P.

(3) The arbitrary assumption of $C_C = -.20\ C_N$ is based on an average figure for C_C at $C_{L_{max}}$. This assumption is equivalent to assuming C_N to be that at $C_{L_{max}}$ and adjusting the design speed to give the applied load factor required. If the gust condition causes the value of C_L to exceed $C_{L_{max}}$, the negative design chord coefficient will usually be greater than the arbitrary value specified.

(C) Condition II. (Sec. 15(C) Aeronautics Bulletin No. 7-A). This condition represents the effects of encountering a downward gust of 30 feet per second while flying at the speed V_L. The coefficients to be used are graphically illustrated on Figure 11. The assumption of a zero chord coefficient in certain cases is not a requirement, but is permitted in order to simplify the analysis.

(D) Condition III. (Sec. 15(D) Aeronautics Bulletin No. 7-A). This condition represents an upward acceleration of the airplane at its design gliding speed V_g. The coefficients to be used are shown graphically on Figure 12. As in Condition I the applied load factor is considered to be produced either by a gust* or by a maneuver. As the speed V_g is the speed at which the airplane will probably be flown least, the gust load factor formula is based on a gust of 15 feet per second and the arbitrary value of the applied acceleration required is less than that for Condition I. This is further justified by the fact that for a conventional 2-spar wing, the value of the applied load factor affects the rear spar load much less than the values of speed and moment coefficient used and is therefore relatively unimportant. For other types of wings, the values of speed and moment coefficient are again usually the most important with respect to torsional loading, the maximum beam loading being obtained from Condition I.

(1) The use of the increment for the moment coefficient is explained in Aeronautics Bulletin No. 7-A. The nature of the moment coefficient is such that a slight wing rib distortion or aileron deflection will have the greatest effect on those airfoils which have the lowest moment coefficients. The use of a multiplying correction factor in this case would therefore be irrational, as the opposite effect would be obtained.

(E) Condition IV. (Section 15(E) Aeronautics Bulletin No. 7-A). This condition, which is graphically illustrated on Figure 12, represents the effects of encountering a "down" gust of 15 feet per second while flying at the design gliding speed, V_g. The considerations outlined for Condition III in the preceding paragraph apply also to this condition.

* "Upward" and normal to flight path.

Section 12. Balancing the Airplane.

(A) The basic design conditions must be converted into conditions representing the external loads applied to the airplane, before a complete stress analysis can be made. This process is commonly referred to as "balancing" the airplane and the final condition is referred to as a condition of "equilibrium". Actually, the airplane is in equilibrium only in steady unaccelerated flight; in accelerated conditions both linear and angular accelerations act to change the velocity and attitude of the airplane. It is customary to represent a dynamic condition, for stress analysis purposes, as a static condition by the expedient of assigning to each item of mass the increased force with which it resists acceleration. Thus if the total applied load acting on the airplane in a certain direction is "n" times the total weight of the airplane, each item of mass in the airplane is assumed to act on the airplane structure in an exactly opposite direction and with a force equal to "n" times its weight.

(B) If the net resultant moment of the air forces acting on the airplane is not zero, with respect to the center of gravity, an angular acceleration results. An exact analysis would require the computation of this angular acceleration and its application to each item of mass in the airplane. In general, such an analysis is not necessary, except in certain unsymmetrical conditions. The usual expedient in the case of the symmetrical flying conditions is to eliminate the effects of the unbalanced couple by applying a balancing load near the tail of the airplane in such a way that the moment of the total applied force about the center of gravity is reduced to zero. This method is particularly convenient, as the balancing tail load can then be thought of either as an aerodynamic force from the tail surfaces or as a part of a couple approximately representing the angular inertia forces of the masses of and in the airplane. Considering a gust condition, it is probable that angular inertia forces initially resist most of the unbalanced couple added by the gust, while in a more or less steady pull-up condition the tail load may consist entirely of a balancing air load from the tail surfaces.

(C) The following general assumptions are made in balancing the airplane:

(1) Full "power on" is assumed for conditions at V_L (Conditions I and II), but for conditions at V_g (Conditions III and IV) the propeller thrust is assumed to be zero.

(2) It is assumed that the applied load factors specified for the basic flying conditions are <u>wing</u> load factors. A solution is therefore made for the <u>net</u> load factor acting on the whole airplane. The value so determined can then be used in connection with each item of weight (or group of items) in analyzing the fuselage. For balancing purposes the net load factor can be assumed to act at the center of gravity of the airplane.

(3) Assuming that it is possible for a load to be acting in the opposite direction on the elevator, it is recommended that the center of pressure of the horizontal tail be placed at <u>20 per cent of the mean chord of the entire tail surface.</u> This arbitrary location may also be considered as the point of application of inertia forces resulting from angular acceleration, thus simplifying the balancing process.

(4) In Figure 13 the external forces are assumed to be acting at four points only. The assumption can generally be made that the fuselage drag acts at the center of gravity. When more accurate data are available, the resultant fuselage drag force can of course be computed and applied at the proper point. In cases where large independent items having considerable drag (such as nacelles) are present, it is advisable to extend the set-up shown in Figure 13 to include the additional external forces.

(D) As shown in Figure 13, a convenient reference axis is the basic chord line of the mean aerodynamic wing chord. (The <u>basic</u> chord line is the chord line to which the aerodynamic coefficients <u>are</u> referred and is usually specified along with the dimensions of the airfoil section). The determination of the size and location of the M.A.C. is outlined in Sections 9 and 10. If any other reference axis is used, the design aerodynamic coefficients cannot be used directly in the computations.

(E) 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 IV. In using Figure 13 and Table IV the following assumptions and conventions should be employed:

(1) <u>If known distances or forces are opposite in direction from those shown in Figure 13, a negative sign should be prefixed before inserting in the computations.</u> For instance, in the case of a high-wing monoplane, h_2 will have a negative sign. Likewise n_{x4} will be either negative or zero in all cases. 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 positively accelerated conditions, 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.

(2) <u>All distances should be divided by the length of the M.A.C.</u> before being used in the computations.

(3) The propeller thrust may generally be assumed to act parallel to the basic reference line.

(4) The chord load acting at the tail surfaces may be neglected.

(F) Computation of Balancing Loads. In Table IV the computation of balancing loads is indicated for typical flight conditions. The equations are based on the fact that the use of the average force coefficients in connection with the effective wing area, mean aerodynamic chord, and mean aerodynamic center will give resultant forces and moments of the proper magnitude, direction, and location. Provision is made in the table for obtaining the balancing loads for different gross weights, if desired. The table may be expanded to include computations for several loading conditions, special flight conditions, or conditions involving the use of auxiliary devices. It should be noted that a change in the location of the c.g. will require a corresponding change in the values of x_2 and h_2 on Figure 13. The basic characteristics and load factors for the accelerated conditions are outlined in Chapter II of Aeronautics Bulletin No. 7-A. In general, the special requirements applying to wings and wing bracing need not be incorporated in the balancing computations, except that when wing flaps are used the required balancing tail load should be determined for the design conditions which apply with flaps deflected. (See Aeronautics Bulletin No. 7-A, Section 18(D)).

(G) The following explanatory notes refer by number to items appearing on Table IV:

(3) The wing loading, s, should be based on the effective wing area.

(5) n_1 = applied load factor required for the condition being investigated. (See Aeronautics Bulletin No. 7-A, Chapter II).

(9) Determine C_C as specified in Aeronautics Bulletin No. 7-A, Chapter II. See also Sec. 3, Equation 8, of this bulletin.

(10) Propeller thrust, F_{pr}, should be determined from Equation 16, Sec. 3, for conditions at V_1. For conditions at V_g assume $n_{x_4} = 0$.

(11) The value of C'_m is specified in Aeronautics Bulletin No. 7-A, Chapter II. For a biplane see Sec. 10 of this bulletin. See also Sec. 2([illegible]) in cases involving wing flaps.

(13) The net tail load factor, n_3 is found by a summation of moments about point (2) of Figure 13, from which the following equation is obtained:

$$n_3 = \frac{1}{(x_3 - x_2)} \left[m_1 - n_{x_1} h_2 + n_1 x_2 + n_{x_4} (h_4 - h_2) \right]$$

Note. The above comments apply only when the set-up shown in Figure 13 is used. If a different distribution of external loads or a different system of measuring distances is employed, the computations should be correspondingly modified.

Chapter III - WINGS AND WING BRACING

Section 13. General.

(A) The wings and wing bracing are designed primarily by the basic flying conditions. However, in order to provide for probable deviations from the basic conditions, certain modifications and supplementary conditions are required in Chapter III of Aeronautics Bulletin No. 7-A. Some of these modified conditions apply only to certain types of wing structures. In any case, the investigation required will not usually have to be carried beyond the members immediately effected.

(B) The modified and supplementary design conditions are included on the chart shown as Figure 10. These conditions are further explained in the following sections

Section 14. Modifications to Basic Flying Conditions.

(A) Condition I_1 - (Sec. 18 (A) Aeronautics Bulletin No. 7-A). In Condition I, the value of C_N required to produce the specified applied load factor at the high speed of the airplane will usually be considerably less than that corresponding to C_{Lmax}. Condition I is designed to be critical for the front spar in bending and compression. For this reason arbitrary values of C_C and C.P. are assigned, which ordinarily represent a pull-up to the applied load factor at a speed lower than V_L. In certain cases, however, the actual accelerated condition at V_L may be critical, in which case it should be checked. The characteristics used for Condition I_1 are illustrated on Figure 15 . This condition applies to the following cases:

(1) Front Spar. When the tension flange or chord member of a front spar is designed for low margins of safety, the smaller forward chord component which occurs in Condition I_1 may permit the net tension load to become greater than that computed for Condition I and thereby result in negative margins of safety.

(2) Rear Spar and Rear Lift Truss. When a wing section having a low or a positive moment coefficient is employed, it is possible for the rear spar to receive its greatest beam loading when the applied load factor for Condition I is developed at the speed V_L. Condition I_1 should be investigated in such cases.

(B) Condition III_1 - (Sec. 18 (F) Aeronautics Bulletin No. 7-A). This condition is included to provide for the use of ailerons during a pull-up or gust loading at the design gliding speed. It should be noted that a relatively small downward aileron deflection is sufficient to change the moment coefficient from a very small or positive value to the arbitrary value of -0.05 specified. The effect of the displaced ailerons on high-moment airfoils is small in proportion and for that reason no corrections are required for such airfoils. In general, only the rear spar and rear lift truss need to be investigated for this condition. This requirement is not applied to Condition IV, as the down load on the front spar is not as sensitive to changes in aileron position.

Section 15. Supplementary Wing Design Conditions.

(A) The basic design flying conditions may not cover certain possible combinations of aerodynamic coefficients which cause critical loads for a particular portion or component of the airplane structure. The supplementary conditions are designed for this purpose and are purposely made as simple as possible to reduce the extra amount of investigation required.

(B) Condition V - (Sec. 19 (A) Aeronautics Bulletin No. 7-A). An airfoil which has a negative moment coefficient always tends to produce an up load on the rear spar. It will usually be found, therefore, that none of the basic flying conditions produces any considerable down load on the rear spar (or any considerable "stalling" moment about the elastic axis of a wing). At large negative angles of attack, however, the moment coefficient about the aerodynamic center approaches zero and may even reverse in sign. This means that the C.P. approaches or lies behind the aerodynamic center. Condition V therefore represents such a condition, which is likely to be developed only in inverted flight. The applied load factors represent either a gust load factor, which may be produced while flying inverted at a speed less than V_L, or a pull-up load factor based on the corresponding value for Condition I. For simplicity the value of C_C is assumed to be zero, as the condition is not well-defined and the speed is assumed to be so low that chord loads would be negligible. See Figure (14) for illustration of this condition.

(1) It should be noted that the maximum rearward position of the C.P. for large negative angles of attack (above the negative stalling angle) approaches 40 percent of the chord as a practical limit. For highly maneuverable airplanes, it would therefore be advisable to use this location of the C.P. in the inverted flight condition, in order to obtain adequate strength in the rear lift truss system.

(2) In general, Condition V will not be critical for portions of the structure other than the rear spar, rear lift truss, and fuselage carry-through members. When a single-lift truss is used, a preliminary check should be made for this condition.

(C) Condition VI - (Sec. 19 (B) Aeronautics Bulletin No. 7-A).

(1) For a conventional 2-spar airplane in which the wing moment is assumed to be resisted entirely by the lift trusses, Condition VI_1 is sufficient to check the strength of the drag trusses for rearward chord loads. This condition, as indicated on Figure 14, is equivalent to the assumption that while flying at the speed V_g, a small negative gust changes the value of C_L to that at which the rearward chord load is a maximum. Since it is assumed that the drag trusses are not affected by normal or moment forces, the actual values of C_N and C_M need not be determined.

(2) When Condition VI is applied to certain types of wing construction, it is desirable to include also the effects of normal forces and wing moment, as specified in Sec. 19 (B)(2) of Aeronautics Bulletin No. 7-A. The condition is then called VI_2, but it is to be considered as replacing VI_1, not as an additional condition. Figure (14) indicates the coefficients which apply in this case.

(a) When this condition is applied to biplanes having a single lift truss, it will usually be found that only the lower wing is critical with respect to rearward chord loads.

(D) Unsymmetrical Flying Conditions - (Sec. 19 (C) Aeronautics Bulletin No. 7-A). On account of the large resistance to angular acceleration which is offered by the wings of an airplane, it is probable that the unsymmetrical flying conditions do not impose severe unbalanced loads on the fuselage structure. The arbitrary assumptions which have previously been used in connection with unsymmetrical flying conditions have, however, been associated with the assumption that the fuselage resists all the unbalanced rolling moment from the wings. Furthermore, it is probable that the side landing conditions which have been used for land aircraft have not been severe enough with respect to the "whipping" effect of the wings produced during the righting process following a side landing. In view of these facts, it is required in Aeronautics Bulletin 7-A that the helpful effects of the angular inertia of the wings shall be neglected. This is to be considered as a temporary requirement, pending the development of more rational unsymmetrical conditions in both flying and landing. Provision is made, however, for the use of a more rational unsymmetrical flying condition by the insertion of a special side landing condition in Chapter VI of Aeronautics Bulletin No. 7-A.

(1) The analysis of conditions involving angular acceleration is explained in Chapter VIII, as such conditions are usually critical for members of the fuselage structure.

(2) The unsymmetrical flying conditions apply particularly to cabane bracing, which should be considered as part of the lift truss.

(3) In applying the unsymmetrical flying conditions as specified in Aeronautics Bulletin No. 7-A, Sec. 19 (C), the approximate method of applying adjustments directly to the wing reactions may be used if desired. This method obviates the necessity for an additional determination of the beam loads.* Actually, however, the change in loading on one side of the wing does not imply a proportional difference in the loading for each component of the structure but affects mainly the values of C_N. The gain in accuracy which would result from a more rational analysis is not believed to be worth the additional labor involved, in view of the arbitrary nature of the unsymmetrical flying conditions.

* However, if the wing moment coefficient and aileron disposition are such as to have made necessary an analysis of Condition III_1, then 100 percent of the loads from that condition are to be applied on one side and 70 percent of the loads from Condition III on the other side.

Section 16. Determination of Spar Loading-Conventional Wing.

(A) The following method of determining the running load on the spars of a two-spar, fabric-covered wing has been developed to simplify the calculations required and to provide for certain features which cannot be accounted for in a less general method. It will usually be found that certain items are constant over the span, in which case the computations are considerably simplified.

(B) The net running load on each spar, in pounds per inch run, can be obtained from the following equations:

$$y_f = \left[\left\{C_N\ (r-a) + C_{M_a}\right\} q + n_2\ e\ (r-j)\right] \frac{C'}{144\ b}$$

$$y_r = \left[\left\{C_N\ (a-f) - C_{M_a}\right\} q + n_2\ e\ (j-f)\right] \frac{C'}{144\ b}$$

Where y_f = net running load on front spar, lbs./inch.
y_r = net running load on rear spar, lbs./inch.
a, b, f, j, and r are shown on Figure 16 and are all expressed as fractions of the chord at the station in question.

(Note: the value of "a" must agree with the value on which C_{M_a} is based.)

q = dynamic pressure for the condition being investigated.

C_N and C_{M_a} are the airfoil coefficients at the section in question.

C' is the wing chord, in inches.

e is the average unit weight of the wing, in pounds per square foot, over the chord at the station in question. It should be computed or estimated for each area included between the wing stations investigated, unless the unit wing weight is substantially constant, in which case a constant value may be assumed. By properly correlating the values of e and j, the effects of local weights such as fuel tanks and nacelles, can be directly accounted for.

n_2 is the net applied load factor representing the inertia effect of the whole airplane acting at the c.g. The inertia load always acts in a direction opposite to the net air load. For positively accelerated conditions n_2 will always be negative, and vice versa. Its value and sign are obtained in the balancing process outlined in Section 12.

(1) The computations required in using the above method are outlined on Tables V and VI, in a form which is convenient for making calculations and for checking. The following modifications and notes apply to these tables:

(a) When the curvature of the wing tip prevents the spars from extending to the extreme tip of the wing, the effect of the tip loads on the spar can easily be accounted for by extending the spars to the extreme span as hypothetical members. In such cases the dimension (f) will become negative, as the leading edge will lie behind the hypothetical front spar.

(b) The local values of C_N, item 14 , are determined from the design value of C_N in accordance with the proper span distribution curves. Figure 6 (a) is used for this purpose, together with the value of A_p obtained for this figure, as outlined in Section 9 (C)3.

(c) Item 15 provides for a variation in the local value of C_M. For Condition I, the value of C_M should be determined from the design value of C.P. by the following equation, using item numbers from Tables V and VI:

$$C'_{M_a} = (14) \times ((3) - C.P.')$$

When conditions with deflected flaps are investigated, the value of C_{M_a} over the flap portion should be properly modified. For most other conditions C_{M_a} will have a constant value over the span.

(d) It will be noted that the gross running loads on the wing structure can be obtained by assuming e to be zero, in which case items 19 , 25 , and 30 become zero, y_F becomes (18) x (13) , y_R becomes (24) x (13) , and y_c becomes (29) x (2).

Section 17. Determination of Running Chord Load.

(A) The net chord loading, in pounds per inch run, can be determined from the following equation:

$$y_c = \left[C_C \, q + n_{x2} \, e \right] \, c/144$$

Where y_c = running chord load, lbs./inch

C_C = design chord coefficient at each station. The proper sign should be retained throughout the computations.

q = dynamic pressure for the condition being investigated.

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n_{x2} = net applied chord load factor approximately representing the inertia effect of the whole airplane in the chord direction. The value and sign are obtained in the balancing process outlined in Sec. 17. Note that when C_C is negative, n_{x2} will be positive.

e and C' are the same as in Sec. 16 (B).

(1) The computations for obtaining the chord load are outlined in Table VI, items 28 to 32. The following points should be noted:

(a) The value of C_C, item 28 , can usually be assumed to be constant over the span. The only variation required is in the case of partial-span wing flaps or similar devices.

(b) The relative location of the wing spars and drag truss will affect the drag truss loading produced by the chord and normal air forces. This can be easily accounted for by correcting the value of C_C as indicated in Sec.4(B)and Figure 3.

(2) It is often necessary to consider the local loads produced by the propeller thrust and the drag of items attached to the wing. The general rules concerning these items are outlined in Aeronautics Bulletin No. 7-A, Sec. 21(B). The drag of nacelles built into the wing is usually so small that it can be safely neglected. The drag of independent nacelles and that of wing-tip floats can be computed by using a rational drag coefficient or drag area in conjunction with the design speed. The beam or torsional loads applied to the wing through the attachment members should also be considered in the analysis. In general, the effects of nacelles or floats can be separately computed and added to the loads obtained in the design conditions.

Section 18. Determination of Running Load and Torsion at Elastic Axis.

(A) The following method can be used in cases where it is desired to compute the running load along any given axis, together with the unit value of the torsion acting about that axis.

(B) As shown in Figure 17, x denotes the location of the reference axis, expressed as a fraction of the chord. The net running load along the locus of the points x and the net running torsion about these points are found from the following equations:

$$y_x = (C_N q + n_2 e) \frac{C'}{144}$$

$$m_x = \left[\left\{ C_N (x-a) + C_{Ma} \right\} q + n_2 e (x-j) \right] \frac{(C')^2}{144}$$

Where y_x is in pounds per inch run
m_x is in inch pounds per inch run.
x is expressed as a fraction of the chord.
C' is the wing chord, in inches.

The remaining symbols are explained in Sec. 16(B). (As noted previously, n_2 will always be negative for positively accelerated conditions).

(1) The computations required for this form of analysis can be conveniently carried out through the use of tables similar to Tables V and VI. The items appearing in each table would be changed to correspond to the equations given in Paragraph (B) of this section. The computation of the running chord load can be made in the manner outlined in Sec. 17.

Section 19. Analysis of Wood Wing Spars.

(A) In the design of wing spars and other members subjected to combined axial and transverse loading the effects of secondary bending can be accounted for by the "precise" equations based on the equation of the elastic axis. In order to maintain the required factor of safety, it is necessary to base such computations on the <u>design</u> loads, rather than the <u>applied</u> loads.

(B) The allowable total unit stress in spruce members subjected to combined bending and compression may be computed from Figure 18 . On the right-hand side of this figure are two families of curves, the upper one for the determination of the modulus of rupture and the lower one for the elastic limit in bending. Each of these quantities is dependent on the ratio of compression flange thickness to total depth of beam and the ratio of web thickness to total width of beam. On the left-hand side of the figure are two additional families of curves. The "horizontal" family indicates the elastic limit under combined bending and compression and the "vertical" family shows the effect of various slenderness ratios on the former quantity. The allowable total stress under combined load, F_t, is found as follows:

(1) For the cross section of the given beam find the elastic limit in bending and the modulus of rupture from the ratios of compression flange thickness to total depth, and web thickness to total width, locating points such as A and B.

(2) Project points A and B to the central line, obtaining points such as C and D.

(3) Locate a point such as E, indicating the elastic limit of the given section under combined bending and compression. This point will be at the intersection of the curve of the "horizontal" family through C and the curve of slenderness ratio corresponding to the distance between points of inflection.

(4) Draw E D.

(5) Locate F on E D, with an abscissa equal to the computed ratio of bending to total stress. The ordinate of F represents the desired value of F_t.

(C) The following rules should be observed in the use of Figure 18:

(1) In computing the margin of safety of any point of the spar, L should be taken as twice the distance from this point to the nearest point of inflection or point of support. If the dimensions of the spars are such that this rule results in a distance greater than the unsupported span, L should be taken as equal to the latter distance.

In computing ρ for the purpose of applying the curves of Figure 18, filler blocks may be neglected and, in case of a tapered spar, the average value should be used.

(2) In computing the modulus of rupture and the elastic limit in bending, the properties of the section being investigated should be used. Filler blocks may be included in the section for this purpose and also in computing the imposed stresses f_b, f_c, and f_t.

(3) The bending moment from which f_b is computed should include an allowance for secondary bending. When possible, this should be done by using a precise method for computing the bending moments. When this cannot be done, a conservative allowance should be made for its effect.

(D) The maximum intensity of stress in longitudinal shear in the webs of wood spars may be determined by means of the following formula:

$$F = SQ/bI$$

Where
- S = vertical shear at the section.
- Q = the statical moment about the neutral axis of the area above the section under consideration.
- b = the entire web thickness.
- I = moment of inertia of the cross section.

(E) For conventional wings, the strength of the beams against lateral buckling may be determined by considering the sum of the axial loads in both spars to be resisted by the spars acting together. The total allowable column strength of both spars is the sum of the column strengths of each spar acting as a pin-ended column the length of a drag bay. Fabric wing covering may be assumed to increase the total allowable column strength, as above determined, by 50 percent. When further stiffened by plywood or metal leading edge covering extending over both surfaces forward of the front spar a total increase in allowable column strength of 200 percent may be assumed.

Section 20. Metal Spars - General.

(A) The bending moments and shears should be computed by precise formulas which allow for the effects of the axial loads. Formulas for shear can be developed by differentiating the formulas for bending moments. The values of EI used in the computations should preferably be determined from a test on a section of beam subjected to loads in the plane of the beam and normal to its axis. In such tests it is recommended that the beam be simply supported at the lift truss fittings and subjected to equal concentrated loads at or near the third points of the span of such magnitude that the maximum shear and bending moment on the test specimen shall be in the same ratio as the maximum primary shears and bending moments on the corresponding spans of the beam in the airplane. If this is not practicable, the shear on the test beam should be relatively larger than in the airplane. The deflections in the test should be read to the degree of precision necessary to obtain a precision in the computed value of EI of not less than $\pm$ 5 percent. When such a test cannot be made, the value of EI may be computed from the geometrical properties of the section and the elastic properties of the material used, but before being used in the formulas for computing deflections, shears, or secondary bending moments, this value should be multiplied by a correction factor to allow for shear deformation, play in joints, and lack of precision in computing the geometric properties of irregular sections. The correction factors recommended are 0.95 for beams having continuous webs that are integral with the chords, extruded I, and similar beams; 0.85 for built-up plate girders having continuous webs connected to the chords by riveting; 0.75 for beams with webs having lightening holes of such shape that the beam cannot be analyzed as a truss.

(B) Thin-Web Metal Spars.

Thin-web metal spars may be analyzed in accordance with the theory of flat plate metal girders, under the assumption that diagonal tension fields will be produced by the shear forces. For detailed information on this subject see N.A.C.A. Technical Note No. 469. The analysis should include the attachment of the web to the flanges.

(C) Truss-Type Metal Spars.

(1) Metal truss spars, in which the axial load is so small that L/j (or equivalent symbol as used in the formulas for computing the stresses in beems subjected to combined loadings) is less than unity, may be analyzed as pin-jointed structures if the axes of the members meeting at each joint intersect at a point. When the axes of the members meeting at any joint do not intersect at a single point, the figure formed with the axes of the members as its sides may be called the "eccentricity pattern" of the joint. In these cases the axial loads in the actual truss members may be assumed to be the same as those in the members of an equivalent truss with the joints located anywhere on that side of the eccentricity pattern formed by the axis of the chord member. Where there is an eccentricity pattern at the end of any truss member, the load on that member applied through that joint may be assumed to be composed of an axial load P, computed as described above, and a bending moment equal to Pe, where e is the normal distance from the axis of the member to the most distant corner of the eccentricity pattern. A more rational analysis can be made by dividing the total eccentric moment (about the true intersection of the web members) between the members intersecting at the joint in proportion to their relative resistance to rotation of the joint.

(2) In metal truss spars, for which L/j is greater than unity, the bending moments and shears on the spar should be obtained by the use of the precise formulas. The values of EI to be used in these formulas should be obtained whenever possible from deflection tests of the type described above in Paragraph (A). When tests are not practicable, the deflections used for determining EI may be obtained by the use of any of the standard methods of computing the deflections of a truss, the assumed loading being that which would be used in a test. In computing these deflections it should be assumed that there is from 0.005 to 0.010 inch slip in the joint at each end of each web member of a riveted or bolted truss. No slip need be assumed with welded joints. Whether the deflections are obtained by test or are computed, EI values should be obtained for at least three points in each span of the truss and the average used in the precise formulas. When an external load parallel to the axis of the spar is applied at any section at a point other than the centroid of the chords at that section considered as a unit, it should be treated in the precise formulas as an equivalent combination of an axial load at that centroid and a bending moment.

(3) The loads in the chord members at any section should be computed from $F = PA_c/A \pm M/h$, where P is the total axial load, A_c the area of the chord under consideration, A the sum of the areas of the chords without allowance for rivet holes, M the total bending moment from the precise formulas, and h the distance between the centroids of the chords. Where the axis of the spar is not straight between support joints, M should be increased or decreased by Pe, e being the distance on the unloaded truss from the centroid of the chords considered as a unit at the section under investigation to a line joining the similar centroids at the support sections. Where full scale tests are not practicable, the loads in the web members should be computed from $F = S/\sin\theta$, where θ is the angle between the web member and the axis of the spar and S is the derivative of the total bending moment with respect to x. If the chords are not parallel, S should be corrected by an amount equal to the shear carried by the chords which are cut by the same section as the web member. Where the chord members change section, the web members should be designed to carry an additional load the component of which, parallel to the spar axis, is equal to the part of the total axial load P that must be transferred from one chord to the other. Thus, if the area of the upper chord changes from 0.6 of the total chord area to 0.5 of the total chord area, the added load in the web members will be $0.1P/\cos\theta$. For simplicity, this load may be applied entirely to the web member adjacent to the change of section, when such procedure is conservative for that member.

(D) Design of Chord Members. The column length should be assumed as the center-line distance between truss joints for bending in the plane of the truss, using a restraint coefficient of not more than 2.0. For bending laterally it should be assumed as the distance between drag struts except that:

(1) If the ribs have adequate strength to prevent lateral buckling the distance may be taken as not less than one-half the distance between drag struts.

(2) If the wing covering is metal, suitably stiffened, the bending laterally may be neglected.

(E) Design of Web Members. When there are no eccentricity patterns and the centroid of the rivet group is on the axis of the member, the column length may be assumed to be equal to the center line length of the member. The restraint coefficient used will depend on the type of joint employed but should in no case exceed 2.0. When eccentricity patterns exist or when the centroid of the rivet group is eccentric to the axis of the member, the member should be considered as an eccentrically loaded column of length equal to its true centerline length, the assumed eccentricity of the axial load at each end being taken as the arithmetical sum of the rivet group eccentricity and the distance from the axis of the member to the most distant corner of the eccentricity pattern. (See Paragraph C-1). When a more exact method of analysis is employed, each member should be analyzed for the proper combination of axial load and end moment.

Section 21. Skin-Stressed Wings.

(A) Plywood Covered Wings. Wings that are completely covered with plywood may be designed under the following assumptions:

(1) The covering will carry the shear due to the chord components of the external loads, provided that suitable compression members are installed between the spars, and cut-outs are properly reinforced. The axial loads in the spars due to chord loads should not be neglected in the spar analysis.

(2) If the loads on the spars are computed by means of conventional methods, without reference to the elastic characteristics of the entire structure, it may be assumed that plywood covering, if rigidly attached to the spars and ribs throughout their entire length, will carry 10 percent of the moments on the wing due to the beam components of the air loads. The spars should be designed to carry at least 90 percent of these moments. In case such covering is removable or contains large openings or other discontinuities between the spars on either surface of the wing, proper reduction in the assumed strength of the covering adjacent to such opening should be made. No reduction should be made in the shear loads to be carried by the spars.

(B) Metal-Covered Wings. Because of the lack of uniformity in the types of metal-covered wings in general use, it is recommended that extensive static testing be employed either in lieu of or in conjunction with stress analysis methods. In many cases a proof test to the specified applied load is the only method by which the behavior of the metal covering can be determined. The following points should be considered in investigating the strength of metal covered wings:

(1) The covering should not develop waves or wrinkles of appreciable magnitude at loadings less than approximately half the specified applied load. Where the covering is depended on to resist any part of the bending moment, no wrinkles which impair the strength of the structure should form before the design load is reached.

(2) Methods of analysis involving the use of the elastic axis of the wing are satisfactory if the position of the elastic axis is definitely known. It is usually advisable to eliminate any uncertainty in this respect by assuming different positions for the elastic axis, thereby covering a range in which it is certain to lie.

(3) Analyses of skin-stressed wings involving the strength of sheet and stiffener combinations, or the strength of thin-web girders, should be supplemented by at least one static test of a representative panel in which the design conditions are closely simulated. Such a panel should be relatively large, in order to account for the interaction of various parts of the structure.

(4) On cantilever wings the torsional stiffness should be made as high as practicable, especially for high speed airplanes and for wings which are relatively stiff in bending.

Sec. 22. - Chord Distribution - Wing Ribs.

(A) Wing Ribs. The approximate methods of chord loading outlined in Aeronautics Bulletin No. 7-A for the testing of wing ribs are suitable for conventional two-spar construction but do not accurately represent the true chord distribution. In some cases it is necessary to determine the actual distribution, not only for the 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, and 465 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 immediately obtained for any C_L. Curves of this nature can be obtained directly from the National Advisory Committee for Aeronautics, for several widely-used airfoils.

(B) Leading Edge Loads. On high speed airplanes the leading edge loads developed may be exceptionally severe, particularly the "down" loads which are produced by negative gusts when flying at the limited diving speed. The magnitude of such loads can be estimated, without determining the entire chord distribution, by the method outlined in N.A.C.A. Report No. 413.

(C) Effects of Auxiliary Devices. When a relatively low design speed is used in connection with wing flaps or similar retractable devices, it will not usually be necessary to determine the chord distribution over the entire airfoil. The effect of any device which remains operative up to the design gliding speed of the airplane should be carefully investigated. This applies particularly to auxiliary airfoils and fixed slots.

Section 23 - Lift-Truss Analysis.

(A) Jury Struts. In computing the compressive strength of lift struts which are braced by a jury strut attached to the wing, it is usually satisfactory to assume that a pin-ended joint exists at the point of attachment of the jury strut. The jury strut itself should be investigated for loads imposed by the deflection of the main wing structure. An approximate solution based on relative deflections is satisfactory, except when the jury strut is considered as a point of support in the wing spar analysis, in which case an accurate analysis of the entire structure is necessary.

(B) Redundant Wire Bracing. When two or more wires are attached to a common point on the wing but are not parallel, the following approximate equations may be used for determining the load distribution between wires, provided that the loads so obtained are increased 25 percent in accordance with Section 22 (B) of Aeronautics Bulletin No. 7-A:

$$P_1 = \left[\frac{V_1 A_1 L_1 L_2^3}{V_1^2 A_1 L_2^3 + V_2^2 A_2 L_1^3}\right] B$$

$$P_2 = \left[\frac{V_2 A_2 L_1^3 L_2}{V_1^2 A_1 L_2^3 + V_2^2 A_2 L_1^3}\right] B$$

Where B = beam component of load to be carried at the joint,
P_1 = load in wire 1,
P_2 = load in wire 2,
V_1 = vertical length component of wire 1,
V_2 = vertical length component of wire 2,
A_1 and A_2 represent the areas of the respective wires,
L_1 and L_2 represent the lengths of the respective wires.

The chord components of the air loads on the upper wing and the unbalanced chord components of the loads in the interplane struts and lift wires at their point of attachment to the upper wing should then be assumed to be carried entirely by the internal drag truss of the upper wing.

(C) Indeterminate Wing Cellules. In biplanes which have two complete lift truss and drag truss systems interconnected by an N strut, a twisting moment applied to the wing cellule will be resisted in an indeterminate manner, as each pair of trusses can supply a resisting couple. An exact solution involving the method of least work, or a similar method, can be used to determine the load distribution. For simplicity, however, it is usually assumed that the drag trusses resist only the direct chord loads and that all the normal loads and torsional forces are resisted by the lift trusses. This will usually be conservative for the lift trusses, but does not adequately cover the possible loading conditions for the drag trusses. In the usual biplane arrangement the lower drag truss will tend to be loaded in a rearward direction by the wing moment. Supplementary design Condition VI_2 (Sec. 19 (F)2, Aeronautics Bulletin No. 7-A) therefore represents the most critical condition for the lower drag truss. This condition should be investigated by assuming that a relatively large portion (approximately 75 percent) of the torsional forces about the aerodynamic center are resisted by the drag trusses. In the case of a single-lift-truss biplane, the drag trusses must, of course, resist the entire moment of the air forces with respect to the axis of the lift truss.

Chapter IV - CONTROL SURFACES AND AUXILIARY DEVICES

Section 24. General.

(A) The requirements for the design of control surfaces are outlined in Chapter IV of Aeronautics Bulletin No. 7-A and are shown in tabular form on Figure 19 of this bulletin. The conditions specified are based on the two separate functions of control surfaces: balancing and maneuvering. The requirements are specified so as to account also for the effects of auxiliary control devices, gust loads, and control forces.

(B) The average unit loading normal to any surface is determined by the force coefficient C_N and the dynamic pressure q, as shown by equation 13, Section 3. When dealing with tail surfaces, it is customary to specify the value of C_N for the entire surface, including both the fixed and movable surfaces. The total load so obtained is then distributed so as to simulate the conditions which exist in flight. In the case of ailerons, flaps, or tabs, the value of C_N is usually determined only for the particular surface, without reference to the surface to which it is attached.

(C) The average unit loading is usually assumed to be constant over the span. On account of the nature of the chord distribution curves specified in Figures 5 and 6 of Aeronautics Bulletin No. 7-A, it will be simpler to assume that the unit loading at the hinge line is constant over the span.

Section 25. Balancing Loads.

(A) The balancing loads apply only to the horizontal tail surfaces, as the ailerons and vertical tail surfaces are used only to a small extent for balancing purposes. The use of the vertical tail surfaces for balancing a multi-engined airplane having one engine dead is provided for in Aeronautics Bulletin No. 7-A, Sec. 27(A). See also Sec. 26(F) of this bulletin.

(B) Design condition III or IV is used for determining the balancing load on the horizontal tail, as outlined in Aeronautics Bulletin No. 7-A, Sec. 26(A). Condition III represents a steady pull-out at the design gliding speed. Condition IV is used to obviate the computation of the balancing tail load for steady flight at this speed. These two conditions cover the range of flight attitudes likely to be encountered at the design gliding speed, condition IV being slightly conservative in some cases. This latter condition, however, represents the effects of a "down" gust, which would tend to increase the down load which was acting on the tail before the gust was encountered. The greater balancing load obtained from condition IV is therefore not objectionable, in view of the fact that no definite requirements are specified for gust loads on horizontal tails.

(C) The chord distribution illustrated in Figure 4 of Aeronautics Bulletin No. 7-A is intended to simulate a relatively high angle of attack condition for the stabilizer, in which very high unit loadings can be obtained near the leading edge. The opposite loading required for the elevator in the balancing condition provides for the control force which might be exerted by the pilot to hold the airplane in equilibrium. A minimum value is specified for this control force, as it is possible for such a force to exist even when the net balancing tail load is zero.

(1) In Figure 4 of Aeronautics Bulletin No. 7-A, the load from the elevator is shown as a concentrated load acting at the elevator hinge line. The hinge moment is, of course, resisted by the control system and therefore does not affect the stabilizer.

Section 26. Maneuvering Loads.

(A) General. The specifications for maneuvering loads outlined in Aeronautics Bulletin No. 7-A are intended mainly to place the determination of such loads on a speed - force coefficient basis, to specify values which agree substantially with previous practice, and to provide for the effects of greater airplane speeds. It should be understood that the method is designed for conventional airplanes and that in determining the maneuvering loads the designer should consider the type of service for which the airplane is to be used.

(B) Design Speed. The speed V_p specified in Aeronautics Bulletin No. 7-A, Sec. 25(B), is based on the assumption that the greatest loading on the movable surface and rear portion of the fixed surface will be obtained in a sudden pull-up at some speed in excess of the stalling speed. The formula for V_p is intended to provide for the following features:

(1) V_p cannot be less than the minimum speed of level flight.

(2) Assuming that the size of the surfaces is governed largely by the necessity for adequate control at the minimum speed, the formula tends to reduce the unit loading for the larger control surface areas required when the stalling speed is low.

(3) The high speed of the airplane is included in the formula as a general measure of the magnitude of the maneuvering speed, so that the unit loading will be increased with an increase in high speed.

(4) The factor K_p is an empirical factor to provide for the more severe maneuvers likely to be experienced by small airplanes. This factor is adjusted so as to make the control surface loadings for average airplanes agree approximately with those known to be satisfactory from past experience.

(C) Design Coefficients. The design values of C_N specified in Aeronautics Bulletin No. 7-A represent coefficients which can be attained by deflecting the control surfaces, the highest value representing the largest deflection of the movable surface expected at the design speed. Lower values are used for up loads on the horizontal tail surfaces and for the vertical tail surfaces, as the corresponding control forces are expected to be less in these cases. The numerical values of the coefficients are coordinated with the value of the factor K_p in the equation for design speed and do not represent the maximum coefficients which can be obtained with conventional control surfaces. Higher values may be desirable in certain cases, depending on the purpose of the airplane.

(D) Load Distribution. The chord distributions shown in Figures 5 and 6 of Aeronautics Bulletin No. 7-A represent approximately the type of loading obtained with the movable surface deflected. For tail surfaces, this type of loading is critical for the movable surface and for the rear portion of the fixed surface.

(E) Damping Loads. When a control surface is deflected suddenly, the full maneuvering load tends to build up immediately, after which the airplane begins to acquire an angular velocity. This angular motion causes the direction of the relative air stream over the fixed surface to change, which causes the air load on this surface to build up in a direction such as to oppose the angular rotation of the airplane. This load is concentrated near the leading edge of the fixed surface and is commonly referred to as the damping load. It is provided for in Aeronautics Bulletin No. 7-A as a supplementary condition based on the initial maneuvering condition. (See Secs. 26(C) and 27(B), Aeronautics Bulletin No. 7-A). The damping load is closely related in magnitude to the initial maneuvering load which produces it, so that it is convenient to use the latter loading condition to determine the damping load on the fixed surface. To avoid the necessity for a separate analysis for damping loads, the distribution is made the same as for the balancing loads. In the case of the horizontal surfaces, the damping load therefore acts as a minimum limit for the design of the fixed surface and need not be investigated when the balancing load is critical.

(F) Multi-engined Airplanes. Section 27 of Aeronautics Bulletin No. 7-A specifies that the value of V_p shall not be less than the level flight speed with one engine dead. This ruling is based on the assumption that the unbalanced yawing moment present in such a condition will be balanced by the vertical tail surfaces. In some cases it may be advisable to increase the value of the normal force coefficient to account for features such as engines which are relatively far from the plane of symmetry.

(1) In estimating the speed with one engine dead for use in this manner the following approximate equation may be used:

$$V_p = 0.9\, V_L \left(\frac{n-1}{n}\right)^{\frac{1}{3}}$$

Where V_p = speed with one engine dead.
V_L = normal high speed.
n = total number of engines.

Section 27. Auxiliary Devices.

(A) Wing Flaps. In the design of wing flaps, the critical loading is usually obtained when the flap is completely extended. The requirements outlined in Section 29(A) of Aeronautics Bulletin No. 7-A apply only when the flaps are not used at speeds above a certain predetermined design speed. As noted in Section 18(D)3 of Aeronautics Bulletin No. 7-A, a placard is required to inform the pilot of the speed which should not be exceeded with flaps extended.

(1) The flap position which is most critical for the flap proper may not also be critical for the flap control mechanism and supporting structure. In doubtful cases the flap hinge moment can be plotted as a function of flap angle for various angles of attack within the design range. The necessary characteristic curves should be obtained from reliable wind tunnel tests. The following N.A.C.A. publications pertain to wing flaps in general: Technical Notes 422, 459, 463, 472, 475. Pressure distribution data for split flaps can be found in Technical Note No. 498.

(B) Trailing-Edge Tabs. The effects of tabs on the major control surfaces are accounted for by the methods outlined in Section 29(B) of Aeronautics Bulletin No. 7-A. In determining the normal force coefficient to be used for the computation of the tab loading, the best available wind-tunnel data should be used. N.A.C.A. Report No. 360 can be used for this purpose, pending the publication of more suitable data.

(1) The loading condition specified in Section 29(B)(3) of Aeronautics Bulletin No. 7-A is diagrammatically illustrated in Figure 20(a) of this bulletin. This condition represents the case of the tab load and the control force both acting so as to resist the hinge moment due to the air load on the movable surface. For convenience, the distances and moments can be computed for the neutral position of the movable surface and tab. Actually, the tab load will tend to decrease slightly when the movable surface is deflected, but this effect, being small and difficult to determine rationally, should be neglected.

(2) The condition outlined in Section 29(B)(4) of Aeronautics Bulletin No. 7-A is illustrated in Figure 20(b) of this bulletin. It represents the case of the pilot opposing the hinge moment produced by the tab and hence the maximum control force is specified. As noted in Aeronautics Bulletin No. 7-A this condition will usually be critical only for the movable surface, particularly the members supporting the tab. When these members are loaded by the tab control forces, the resulting combined loads should be accounted for in the design.

Section 28. Stress Analysis.

(A) In analyzing movable control surfaces supported at several hinge points, care should be taken in the use of the "three-moment" equation. In general, the assumption that the points of support lie in a straight line will give misleading results. When possible, the effects of the deflection of the points of support should be approximated in the analysis.

(B) The effects of initial rigging loads on the final internal loads are difficult to predict, but in certain cases may be serious enough to warrant some investigation. In this connection, methods based on least work or deflection theory offer the only exact solution. Approximate methods, however, are satisfactory if based on rational assumptions. As an example, if it appears that a certain counter wire will not become slack before the design load is reached, the analysis can be conducted by assuming that the wire is replaced by a force, acting in addition to the external air forces. The residual load from the counter-wire can be assumed to be a certain percentage of the rated load and will of course be less than the initial rigging load.

Section 29. Flutter Prevention - Balancing.

(A) Static Balance. Satisfactory static balance of a movable control surface is obtained when the center of gravity of the movable structure (excluding control system) is located on the hinge line or in a plane through the hinge line and normal to the mean plane of the surface.

(B) Dynamic Balance. A movable surface is dynamically balanced with respect to a given axis if an angular acceleration of the surface about that axis does not tend to cause the surface to swing about its own hinge line. A control surface which is dynamically balanced about a certain axis will therefore remain "neutral" with respect to a torsional vibration about that axis; that is, it will act substantially as a rigid structure. As the types of flutter likely to be encountered in aircraft structures include both torsional and bending vibration, the type of balancing employed and the choice of a suitable reference axis for any given case will depend on the particular form of flutter to which the component is subject. The axes which pertain to torsional vibration of the fuselage are shown on Figure 21, for the case of a rudder.

(1) In computing the dynamic balance coefficient as outlined in Aeronautics Bulletin No. 7-A, Section 30(G) 6, the surface being investigated should be divided into a relatively large number of portions. The weight of each portion and the perpendicular distance from the center of gravity of the portion to each axis should be determined. The resultant product of inertia of any quadrant is the sum of the individual products of inertia of each item in that quadrant. The product of inertia of any item is the product of its weight and the two distances noted above. Referring to Figure 21, the product of inertia of the item where weight is ΔW is equal to ΔWxy.

(2) The dynamic balance coefficient is an approximate measure of the amount by which a given control surface is out of balance. The convention for signs used in Aeronautics Bulletin No. 7-A corresponds to the assumption that any item, the inertia effect of which would increase the tendency of the control surface to aid flutter, has a positive product of inertia. The positive and negative quadrants are indicated in Figure 21.

(3) In the case of cantilever wings and tail surfaces a type of bending vibration is possible in which the location of the basic axis of rotation is indeterminate, being different for different items and varying with the deflection of the structure. In such cases it is possible to approximate the desired condition by assuming a conservative location for the axis of vibration. In any case, complete dynamic balance will be obtained if each individual portion of the movable surface is statically balanced with respect to the hinge line.

Chapter V - CONTROL SYSTEMS

Section 30. General.

(A) Figure 22 shows a summary of the design conditions for control systems, as specified in Chapter V, Section 31 of Aeronautics Bulletin No. 7-A. A chart of this nature can be used for tabulating the loads used in the stress analysis. Such a table is useful for reference and checking purposes.

Section 31. Special Factors and Limits.

(A) In all cases the applied loads for control systems are specified as 125 percent of the actual loads corresponding to the control surface applied loading, with certain maximum and minimum limits. The factor of 1.25 is used to account for various features, such as:

(1) Differences between the actual and the assumed control surface load distribution.

(2) Desirability of extra strength in the control system to reduce deflections.

(3) Reduction in strength due to wear, play in joints, etc.

(B) The maximum limits are based on the greatest probable control forces which will be exerted by the pilot. These forces can be exceeded under severe conditions, but the probability of this occurrence is very low. The minimum factor of safety of 1.50, which is required in any case, will permit the maximum applied load to be exceeded for a relatively short time without serious consequences.

(C) The minimum limits apply only to cases where the control surface applied loads are relatively small. It is then probable that the minimum control forces will be applied, even though the control surfaces are completely utilized and are against the stops.

(D) The requirement of an extra factor of safety of 1.20 for fittings does not apply in the case of control systems, as the factor of 1.25 provides a sufficient margin and conservative rules are specified for determining allowable bearing stresses in joints. When the control system is designed by either the maximum or minimum control forces it is also unnecessary to use the extra factor of safety for fittings.

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Section 32. Special Considerations.

(A) In some cases involving special leverage or gearing arrangements, the critical loading on the control system may not occur when the surface is fully deflected. For example, in the case of wing flaps the most critical load on the control system may be that corresponding to a relatively small flap displacement, even after proper allowance is made for the change in hinge moment. This condition will occur when the mechanical advantage of the system becomes small at small flap deflections.

(B) An investigation of the strength of a control system includes the various fittings and brackets used for support. In particular, the rigidity of the supporting structure is important, especially in aileron, wing flap, and tab control systems.

(C) Control surface horns are considered a part of the control system. This includes the attachment of the horn to the surface.

Chapter VI - LAND TYPE LANDING GEAR

Section 33. General.

(A) The basic landing conditions are outlined in Aeronautics Bulletin No. 7-A, Sections 35 to 38, and are tabulated on Figure 23 of this bulletin. This chart can be used as a summary of the load factors for the landing conditions, by inserting the actual values used.

(B) The design conditions are chosen so as to cover the various possible types of landings with a minimum amount of investigation. It will usually be found that each different condition is critical for certain members. If the landing gear design is such that it is obvious that a certain condition cannot be critical, such a condition need not be investigated for the landing gear. It will probably be necessary, however, to determine the loads acting on the fuselage for all conditions, for use in the fuselage analysis.

(C) Conventions. In order to simplify the procedure used in analyzing landing gear and float bracing it is recommended that the following conventions be used:

(1) The basic reference axes are designated by V (positive upward), D (positive rearward) and H (positive outward). (For side landing conditions H will be positive outward only with respect to one side).

(2) Tension loads are positive, compression loads negative.

(3) Moments are represented by vectors according to the "right hand" rule.

(4) The basic axes also represent positive moment vectors, each axis being the axis of rotation for the corresponding moment. (Note: changing the sign of a moment reverses the direction of the vector).

(5) In writing the equations of equilibrium, all forces should initially be assumed to be tension (that is, positive). The true nature of the forces will be indicated by the sign of the vector obtained in the final solution.

(6) Moments can be combined vectorially in exactly the same manner as forces and can also be solved for by the same methods.

Section 34. Unsymmetrical Landing Conditions.

(A) Side Landing. (Aeronautics Bulletin No. 7-A, Sec. 37(A)). This condition represents a loading such as would be obtained in a ground loop. It will usually result in critical compressive loads for certain brace members which carry large tension loads in the symmetrical landing conditions. It may also be critical for some fuselage members adjacent to the landing gear. The analysis of the fuselage for this condition will be taken up in Chapter VIII.

(B) One-Wheel Landing. (Aeronautics Bulletin No. 7-A, Sec. 37(B)). This condition does not require additional investigation for the landing gear structure, as the loads are the same as in Level Landing. It represents the "whipping" condition obtained in either of the two following cases:

(1) The airplane is assumed to strike the ground with one wheel only. The initial loading is such as to produce a relatively high angular acceleration, which is resisted by the angular inertia of the airplane about its longitudinal axis through the center of gravity.

(2) After striking the ground on one wheel, or after a landing with considerable side load, the airplane has acquired an angular velocity about its longitudinal axis and tends to roll over on one wheel. By the time the opposite wheel is clear of the ground, any appreciable side load will probably have disappeared, so that the one-wheel landing condition can be used again without modification. Any tendency to continue rolling after the load has been transferred entirely to one wheel will not be likely to increase the load on that wheel, as the kinetic energy of rotation will be converted into potential energy by the rise of the center of gravity.

(C) As the one-wheel landing condition is used only for the design of fuselage and wing attachment members, a detailed explanation of the stress analysis procedure is included in Chapter VIII.

Section 35. Shock Absorption.

(A) The term "shock-absorbing system" usually refers to a pneumatic tire and some form of shock-absorbing strut, interconnected by a system of members which determine the relative motion of each component. Any number of different shock-absorbing systems can be built up from the same shock strut and tire by changing the geometry of the landing gear structure. The characteristics of a shock-absorbing system are not determined by the individual characteristics of the strut and tire alone, but depend on the interaction between these two components. The latter relationship will, of course, be governed by the geometry and design of the landing gear.

(B) As specified in Section 39 of Aeronautics Bulletin No. 7-A, shock-absorbing systems are required to absorb the energy corresponding to a free drop from a certain height without permitting the design load factor to be exceeded. The design load factor is used here in conjunction with a relatively large height of free drop in order to provide a factor of safety with respect to shock-absorption capacity. Dynamic tests are usually necessary for the determination of shock-absorption capacity and characteristics. The capacity of tires alone can usually be obtained from the manufacturer's curves. When a tire is used in conjunction with a shock strut, however, the tire characteristics cannot be used directly in the determination of the characteristics of the entire system. The following points should be noted in connection with tests of shock-absorbing systems:

(1) The maximum loads on the tire and shock strut may not occur at the same instant and are not necessarily of the same relative magnitude.

(2) The "semi-sprung" weight (wheel, tire, and movable portion of shock absorber) should be approximately the same in the test as on the airplane.

(3) Corrections should be made to account for the effects of landing gear design, particularly when the static load on the shock strut is considerably different from the corresponding load on the tire. It is impossible to reproduce exactly the conditions in such a case, when a drop test is made with the shock strut and tire in line. The following procedure is usually sufficiently accurate, where K is the ratio of the load in the strut to the load on the tire:

(a) Use a test weight equal to K times the nominal load on the tire.

(b) The height of free drop should be 1/K times the height specified.

(c) Replace the original tire with a tire having a load-deflection curve, each ordinate (load) of which is K times the original value and each abscissa (deflection) of which is 1/K times the original value, the original values being those for the tire actually used. In addition, the maximum deflection of the test tire should be limited to 1/K times the maximum deflection of the original tire.

(d) These adjustments will cause the actual conditions to be very closely reproduced, except that the impact velocity will be in error. As this factor is of considerable importance in liquid-and-orifice types of shock struts, too large a discrepancy in the impact velocity is undesirable.

(C) As shock struts are usually designed and adjusted to give the highest efficiency with a certain type of tire, any change of tire to a type having a considerably different load-deflection curve or shock absorption capacity will usually require a recheck of the shock-absorption characteristics of the system.

Chapter VII - FLYING BOAT AND SEAPLANE LANDING CONDITIONS

Section 36. General.

(A) The various landing conditions outlined in Chapter VII of Aeronautics Bulletin No. 7-A are summarized on Figure 24. As noted in Aeronautics Bulletin No. 7-A, the requirements for flying boat hulls have not been completely revised, although some revisions have been made in order to simplify certain loading conditions. In particular, the subject of unit pressures on hull and float bottoms warrants further investigation. In the case of large flying boats it is recommended that a preliminary schedule of the proposed loading conditions be submitted before the stress analysis is completed. This will provide an opportunity to make use of the latest available data on this subject and may, in some cases, result in a saving in structural weight.

(B) In certain landing conditions a higher value of the minimum factor of safety is required for some portions of the structure. This is primarily for the purpose of obtaining greater rigidity and to provide for possible variations in the load distribution. In general, whenever the total factor of safety is 1.80 or greater, no further increase is required for fittings. (See Aeronautics Bulletin No. 7-A, Sec. 67(B)). It may be advisable, however, to use an increased factor for fittings which are highly stressed or subjected to reversal of loading, in order to provide for the effects of stress concentration, fatigue, and wear at joints.

Section 37. Design Landing Conditions.

(A) The landing conditions outlined for seaplanes in Sections 44, 45, and 46 of Aeronautics Bulletin No. 7-A correspond, in general, to the conditions used for landplanes. These conditions apply to conventional float installations and in such cases will provide a sufficient range of loadings. Where unconventional types of float bracing are employed it may be advisable to investigate other landing attitudes, depending on the type of loading which appears to be most critical for the structure.

(B) The design landing conditions for flying boats, as specified in Section 47 of Aeronautics Bulletin No. 7-A, are used to provide for several different types of landings with a minimum amount of stress analysis. In particular, the condition specified as "Two-wave Landing" actually provides for two different landing conditions, one in which the bow strikes the water first, the other in which the stern receives the initial impact. The arbitrary assumptions as to reactions and panel-point loads are specified to simplify the analysis and to insure additional strength for those portions of the hull where failure is most likely to occur. The stress analysis of hulls will be taken up in greater detail in Chapter VIII.

(C) No requirements are specified in Aeronautics Bulletin No. 7-A for the landing of flying boats with side load. Such a condition is not likely to be critical for the hull structure as a whole, but in the investigation of bulkheads, etc. it may be advisable to consider the effects of angular acceleration such as that obtained in the "landing with side load" condition for seaplanes (Aeronautics Bulletin No. 7-A, Sec. 46). The analysis of hulls for such conditions corresponds to that used for fuselages and will be described in greater detail in Chapter VIII.

Chapter VIII - FUSELAGE, ENGINE MOUNTS, AND NACELLES

Section 38. General.

(A) Figure 25 summarizes the various conditions which may produce critical loads in fuselage members and indicates the portions of the structure to which each condition usually applies. It is obvious that the analysis for any condition need not be carried beyond a point at which it can be positively shown that no critical loads are produced.

(B) In addition to determining the loads in the main structural members of a fuselage (or hull) the local loads imposed by the internal weights which it supports should not be overlooked. This applies particularly to members which serve both as a critical portion of the primary structure and as a means of support for some item of appreciable weight. The combined stresses should be determined in such cases, except that control system loads need not be combined with the primary loads from the flying or landing conditions.

Section 39. Stress Analysis Procedure.

(A) Weight Distribution. All major items of weight affecting the fuselage should be distributed to convenient panel points in such a way that the true center of gravity of the fuselage and contents is maintained. A suitable vertical division of loads should be included. The following rules should be followed in computing the panel point loads for conventional airplanes.

(1) The weight of an item located between two adjacent panel points of the side trusses should be divided between those panel points in inverse proportion to the distance from them to the center of gravity of the item.

(2) The weight of an item to the rear of the tail post or forward of the front structure should be represented in the table by a load and a horizontal couple at the tail post or front frame.

(3) The weight of an item supported at three or more panel points should be divided between those points by the aid of an investigation and analysis of the method of support, if practicable. Where a rational analysis is not possible, the division may be estimated.

(4) In all cases the moment of the partial panel loads due to any item about an origin near the nose of the fuselage should be equal to the moment of the item about that origin.

(5) All loads may be assumed to lie in the plane of symmetry and to be divided equally between the two vertical trusses of the fuselage.

(B) Balancing (Symmetrical Conditions).

(1) The methods of balancing the airplane are discussed in Section 12. It will, in general, be satisfactory to apply directly the balancing loads found for the various flight conditions. The acceleration factor applied to each item of mass in the fuselage will be the net acceleration as determined from the balancing computations.

(2) The basic inertia force on any item will be parallel to the resultant external applied force and will not, in general, be perpendicular to the thrust line. In certain cases the chord components of the inertia forces (that is, the components along the thrust line or fuselage centerline) can conveniently be combined into a single force and applied at the nose of the fuselage. This procedure permits the use of vertical inertia loads but it should not be used unless it is obviously conservative for the critical fuselage members.

(C) Balancing (Unsymmetrical Conditions). In any condition involving angular acceleration about a given axis, the inertia force applied to the structure by any item of weight is proportional to the mass or weight of the item and its distance from the axis of rotation. Each angular inertia force will act in a direction perpendicular to the radius line between the item and the axis of rotation. In order to facilitate the analysis of a condition involving both linear and angular acceleration, the loads produced by linear acceleration should be determined separately from those produced by angular acceleration. When unbalanced external loads are applied this involves the determination of the magnitude of the net resultant external load and its moment arm about the proper axis through the center of gravity of the airplane. It will usually be satisfactory, in analyses of this nature, to represent the major items of weight, such as wing panels, nacelles, etc.,by assumed concentrated masses at the centers of gravity of the respective items.

(1) Figure 26 illustrates approximate methods by which the fuselage can be balanced for a typical unsymmetrical landing condition (one-wheel landing).

(a) Sketch (a) shows the level landing condition in which the resultant load does not pass through the center of gravity. In such cases it will generally be satisfactory to apply a balancing couple composed of a downward force acting near the nose of the fuselage and an equal upward force acting at the same distance to the rear of the center of gravity. These arbitrary forces can be considered as approximately representing angular inertia forces and they may be divided between the nearest panel points, if desired.

(b) Sketch (b) indicates a satisfactory method of balancing externally applied rolling moments about the longitudinal axis. The forces resisting angular acceleration are assumed to be applied by the wing. The arbitrary location shown is based on the fact that that the effectiveness of any item is proportional to its distance from the center of gravity. The balancing loads may be assumed to be vertical, although they would actually act normal to a radius line through the center of gravity of the airplane. If nacelles or similar items of large weight are attached to the wing, the balancing couples can be divided between nacelles and wing panels in proportion to their effectiveness. This type of balancing applies also to side landing conditions, including those for seaplanes.

(c) Sketch (c) shows an approximate method for balancing a moment about a vertical axis. This condition exists in a one-wheel landing. It is conservative (for the wing attachment members) to assume that the balancing couple is supplied entirely by the wing. The magnitude of the unbalanced moment about a vertical axis is, however, relatively small in the design conditions required in Aeronautics Bulletin No. 7-A. In order to secure ample rigidity against loads tending to twist the wing in its own plane, it may sometimes appear advisable to check the wing attachment members or cabane for a greater unbalanced drag load acting at one wheel, or a side load acting at the tail.

(2) It should be noted that the balancing couples shown on Figure 26 will act in addition to the inertia loads due to linear acceleration. For instance, in Figure 26 (b) the load V shown as a reaction at the c.g. actually represents the inertia loads of all the components of the airplane. Those due to the wing weight will act uniformly on each wing panel and will be added arithmetically to the forces representing the angular inertia effects. This applies also to the other cases illustrated.

Section 40. Special Analysis Methods.

(A) Torsion in truss-type fuselages. In analyzing conventional truss-type fuselages for vertical tail surface loads it will be found convenient to make simplifying assumptions as to internal load distribution. The following methods may be used for this purpose, the first method being more conservative than the second.

(1) The entire side load and torque may be assumed to be resisted only by the top and bottom trusses of the fuselage. The distribution to the trusses can be obtained by taking moments about one of the truss centerlines at the tail post.

(2) For the structure aft of the rearmost bulkhead the tail load may be represented by a side load acting at the center of the tail post and a couple equal to this load times its vertical distance from the center of pressure of the vertical tail. The side load may be assumed to be divided equally between top and bottom trusses. For the structure forward of the rearmost bulkhead the tail load may be represented by a side load acting at the center of the tail post and a torque acting at the rearmost bulkhead equal to the tail load times the vertical distance from the

center of pressure of the vertical tail to the center of this bulkhead. This side load may be assumed to be divided equally between top and bottom trusses. The assumption may be made that the torque (not the forces composing the equivalent couple) is divided equally between the horizontal and vertical trusses. The couples acting on the bulkhead and resisted by the top, bottom, and side trusses can then be readily obtained. Stress diagrams should be drawn for the trusses to obtain the loads in the members. The longeron loads should be taken from the diagrams for the horizontal trusses, vertical trusses, or the combined loads from both trusses, whichever are largest. (This arbitrary practice is advisable on account of the uncertainty of the load distribution between trusses).

The diagonals of the rearmost bulkheads, the bulkheads through which the torque is transmitted to the wing, and all bulkheads adjacent to an unbraced bay, should be designed to transmit the total torque. Intermediate bulkheads should be designed to transmit 25 percent of the total torque.

In some cases the loads obtained in the bottom truss members may be quite small. In such cases it should be noted that it is desirable to maintain a high degree of torsional rigidity in the fuselage and that the rigidity of the top truss will be completely utilized in this respect only when the bottom truss is equally rigid.

(B) Engine Torque. In investigating the conditions involving engine torque, the basic torque may be computed from the following formula:

$$T = 63000\ P/N$$

where T = torque in inch pounds,
P = horsepower of engine,
N = propeller speed in revolutions per minute.

(1) The resulting moment is taken care of by an unsymmetrical distribution of load between the wings and by forces in the fuselage cross bracing. In certain cases, especially when geared engines are used, the stresses due to the torque should be computed for all fuselage members affected, the necessary reactions being assumed to act at the connections of the wings with the fuselage. Otherwise the following approximation may be used for nose engines. The torque load is assumed to act down on the engine bearer and to be held in equilibrium by vertical forces acting at the main connections of the wings with the fuselage, the engine bearer and the members of the fuselage side truss being assumed to lie in a single plane parallel to the plane of symmetry.

(2) When a direct-drive engine is carried by engine bearers that are supported at two or more points, the torque load should be divided between the points of support in the same proportions as the weights carried by the engine bearer. Where an engine is supported by a vertical plate or ring, the torque can correctly be assumed to act at the points of attachment. (The dead weight of the engine, however, should be assumed to act at the center of gravity of the engine).

(3) In combining the torque condition with any other loading condition, for a symmetrical structure, the stresses due to torque are to be added arithmetically, not algebraically, to those obtained for the symmetrical loading condition, because if the forces induced by the torque load in any member are opposite in character to those due to the dead weights there will normally be a corresponding member on the opposite side of the fuselage in which the forces due to the torque loads and weights will be of the same character.

(4) In analyzing an engine mount structure, care should be taken to distribute the torque only to those members which are able to supply a resisting couple. For example, in certain structures having three points of support for the engine ring, it may be necessary to divide the entire engine torque into a single couple, applied at only two of the supporting points.

(C) Skin-stressed Fuselages.

(1) The strength of skin-stressed fuselages is affected by a large number of factors, most of which are difficult to account for in a stress analysis. The following are of special importance:

(a) Effects of doors, windows, and similar cut-outs.

(b) Behavior of metal covering in compression and as a shear web, including the effects of wrinkling.

(c) Strength of curved sheet and stiffener combinations, including fixity conditions and curvature in two dimensions.

(d) True location of neutral axis and stress distribution in bending.

(e) Applied and allowable loads for rings and bulkheads.

(2) Unless a fuselage of this nature conforms closely to a previously constructed type, the strength of which has been determined by test, a stress analysis is not considered as a sufficiently accurate means of determining its strength. In all cases, the stress analysis should be supplemented by test data. Whenever possible it is desirable to test the entire fuselage for bending and torsion, but tests of certain component parts may be satisfactory in conjunction with a stress analysis. As this subject is now being investigated by the National Advisory Committee for Aeronautics, the latest information should be obtained from that organization before the stress analysis or test methods are decided upon.

Chapter IX - MISCELLANEOUS STRUCTURAL INFORMATION

Section 41. Fittings.

(A) In the analysis of a fitting it is desirable to tabulate all the forces which act on the fitting in the various design conditions. This procedure will reduce the chances of overlooking some combination of loads which will be critical.

(B) The minimum factor of safety of 1.80 specified for fittings (Sec. 61(A) of Aeronautics Bulletin No. 7-A) is composed of the standard minimum factor of 1.50 and an additional factor of 1.20. The increase of 20 percent is to account for various factors which tend to increase the probability of failure of a fitting, such as stress concentration, eccentricity, uneven load distribution, and similar features. As noted in Aeronautics Bulletin No. 7-A, whenever the total factor of safety for any portion of the structure equals or exceeds 1.80 the fittings included in this portion are not subject to an increase in factor of safety above the value used for the primary members.

Section 42. Wire-Braced Structures.

(A) The requirements of Section 62 of Aeronautics Bulletin No. 7-A, paragraphs (A) to (C), are based on the necessity for proportioning wire sizes so as to prevent an excessive load from being produced in any wire while rigging any other wire. These requirements provide for an average rigging load of 20 percent. This means that when the maximum allowable ratio of rigging loads (two to one) exists between two wires, one will be assumed to be rigged to 13.3 percent, the other to 26.7 percent. If a larger ratio were permitted, such as three to one for instance, there would be a possibility of obtaining an excessively high rigging load in one wire while rigging the other to a relatively low percentage of its rated load.

(B) The requirement outlined in Section 62(D) of Aeronautics Bulletin No. 7-A is based on the fact that a decrease in the angle between a lift wire and a spar, will greatly increase the deflection for a given loading. The formula used is adjusted so as to maintain, approximately, the deflection which would be obtained for a 30 degree angle between the wire and the spar. It will be noted that the value of K becomes 1.0 when the angle is 30 degrees. Since K approaches infinity as the angle approaches zero, it will be found impractical to design wire-braced structures for small angles between the wires and the members which they support.

(1) A specific example of the application of these principles to an airplane wing is found in a biplane cellule in which lift wires are used for both front and rear spars, but which has only one landing wire (or pair of wires). In such a case the landing wire must act as a counter wire for all the lift wires. This means that a relatively high load must be supplied by the landing wire to counteract normal rigging loads in the flying wires. To meet the requirement as to the maximum allowable ratio of rigging loads it is therefore necessary to use a large landing wire, even though its design load from the flying conditions is comparatively small. In this example it will also be noted that the drag truss wires will be loaded by rigging the flying wires. Obviously, the drag truss wires should be strong enough to prevent excessive rigging loads from being built up.

Section 43. Strength of Materials.

(A) In general, the strength properties and methods of computing allowable stresses outlined in the current edition of the Army "Handbook of Instructions for Airplane Designers" Section II, Part V, are acceptable. Certain extracts from these requirements and additions and deviations are noted below.

(B) Effects of Welding.

(1) Methods of correcting for the effects of welding are stated on pages 232 to 234, inclusive, of the above reference.

(C) Combined Bending and Compression.

The allowable stresses for chrome-molybdenum steel tubes (of any heat-treatment) subjected to combined bending and compression can be determined from Figure 27 as follows:

(1) Locate the point on the chart representing the values of f_c/F_c and f_b/F_b computed from the applied stresses. (Illustrated as intersection of dotted lines on Figure 27).

(2) Draw a straight line through this point and the origin. (Shown as diagonal dotted line on Figure 27).

(3) Read the values of allowable f_c/F_c or f_b/F_b at the intersection of this diagonal line with the proper "B" curve.

(4) Compute the margin of safety from the ratio of the allowable to the applied f_c/F_c or f_b/F_b. (Note: The true margin of safety is obtained in either case, that obtained with the larger figures being the more accurate).

(5) In evaluating coefficients, the following definitions are used:

F_b = modulus of rupture in bending (See Fig. 137-1, Army Handbook).
F_o = yield point.

Note: If desired, F_b may be assumed to be $[1.35 - .01\ (D/t)]$ x (Ult. Tensile Strength),

Where D = diameter of tube,
t = thickness of tube.

(6) Figure 27 represents, in non-dimensional form, the basic curves from which Figure 136 of the Army Handbook was obtained. These basic curves may be used for any degree of heat-treatment for which the strength characteristics are known and should also be used in any case where the modulus of rupture is different from 120,000 pounds per square inch. The curves include the effect of secondary bending and are based on the type of primary bending moment curve produced by lateral loading at the third points of a member. The method of obtaining the margin of safety is based on the assumption that an increase in the basic applied loads will cause the axial and primary bending stresses to increase proportionately.

(7) If the effects of secondary bending were included in the determination of f_b, it is probable that the curves shown on Figure 27 would tend to converge into an approximately straight line, the equation of which would be

$$\frac{f_b}{F_b} + \frac{f_c}{F_c} = 1.0,$$

where f_b is the total bending stress, including secondary bending.

This method may be used to determine the strength of a member, but the margin of safety so obtained is not linear and must be recomputed if there is a change in the basic loading.

(D) Combined Bending, Torsion, and Compression.

As an approximate method, pending the development of more accurate formulas, the effects of torsion may be accounted for by the following equation:

$$\frac{f_b}{F_b} + \frac{f_c}{F_c} + \frac{f_s}{F_s} = 1.0$$

Where f_s is the shear stress due to torsion alone.
F_s is the torsional modulus of rupture for pure torsion. (See Fig. 138 of Army Handbook)
Other symbols are explained in paragraphs (B) and (C).

(1) In using the above equation, a tube will be considered satisfactory if the value of the terms is not greater than 1.0.

(E) Allowable Loads for Columns. The allowable loads for columns of standard shapes can be determined from standard column formulas or charts. When columns of unconventional cross-section are employed, tests should be made to determine the nature of the column curve, particularly in the region where local buckling occurs. Such tests should satisfactorily cover the range of sizes and dimensions which are used in actual construction. It is usually preferable to use pin-ended specimens, as the exact effects of end fixity are difficult to determine. Corrections for material variations can be introduced by the use of non-dimensional coefficients such as those used in N.A.C.A. Technical Note. No. 307. (Figure 27 is taken directly from this publication and makes use of the same coefficients).

(F) Bearing of Bolts on Wood. The allowable bearing load for bolts bearing on wood at an angle other than 90 degrees with respect to the grain should be determined from Figures 120 and 121 of the Army Handbook by means of the following formula:

$$N = \frac{PQ}{P \sin^2 \theta + Q \cos^2 \theta}$$

where N = allowable resultant bearing load,
P = allowable load parallel to the grain,
Q = allowable load perpendicular to the grain
θ = the angle of the resultant load with the grain.

(G) Standard Sizes and Ratings.

(1) The following table gives the rated strength of standard tie-rods:

Size	Rated Strength Pounds	Size	Rated Strength Pounds
No. 6-40	1,000	No. 3/8-24	8,000
No. 10-32	2,100	No. 7/16-20	11,500
No. 1/4-28	3,400	No. 1/2-20	15,500
No. 5/16-24	6,100		

(2) The rated strength of hard aircraft wire is shown in Figure 28.

Chapter X - NON-STRUCTURAL INFORMATION

Section 44. Performance.

(A) The performance requirements outlined in Sections 73 and 74 of Aeronautics Bulletin No. 7-A are based on standard atmospheric conditions. Variations from standard conditions should be accounted for by acceptable methods. The procedure outlined in Section II, Part IV of Army "Handbook of Instructions for Airplane Designers" is satisfactory for this purpose.

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TABLE I

AERODYNAMIC CHARACTERISTICS OF AIRFOILS

1	2	3	4	5	6	7	8
	C_{Lmax}	$C_{Mc/4}$	a	m	C_c		
Airfoil	Max. lift coef- fici- ent.	Moment Coeffi- cient about 1/4 chord point where $C_L = 0$	Aero- dynam- ic cen- ter in percent chord	$\frac{dC_L}{d\alpha}$ per rad. A.R. 6	Max. rear- ward chord coef- fici- ent	Max. thick- ness, per cent chord	Mean camber, per- cent chord with re- spect to true chord
Clark Y	1.56	-0.068	24.2	4.10	0.0203*	11.7	3.8
YM-15	1.58	-.068	24.1	4.13	.0223**	15.0	4.0
YM-18	1.49	-.065	23.6	4.10	.0236**	18.0	4.0
Curtiss C-72	1.62	-.084	23.8	4.23	.0230*	11.7	4.0
Gottingen 387	1.56	-.95	23.9	4.27	.0301*	15.1	5.9
398	1.57	-.083	24.4	4.20	.0253*	13.8	4.9
N-22	1.60	-.074	25.0	4.25	.0229*	12.4	4.3
N.A.C.A. 0006	.88	0	24.3	4.28	.0065	6.0	0
0012	1.53	0	24.1	4.25	.0083	12.0	0
2212	1.60	-.029	24.6	4.31	.0102	12.0	2.0
2409	1.51	-.044	24.7	4.31	.0093	9.0	2.0
2412	1.62	-.044	24.6	4.25	.0099	12.0	2.0
2415	1.55	-.040	24.3	4.25	.0112	15.0	2.0
2418	1.43	-.037	24.0	4.16	.0127	18.0	2.0
4412	1.65	-.089	24.5	4.22	.0158	12.0	4.0
CYH	1.47	-.027	24.5	4.24	.0126*	11.7	3.1
M-6	1.40	+.002	25.0	4.26	.0094	12.0	2.4
M-12	1.25	-.022	25.0	4.03	.0113	12.0	2.1
R.A.F. 15	1.21	-.052	23.2	4.16	.0100***	6.4	2.6
U.S.A. 27	1.59	-.077	23.7	4.11	.0194	11.0	5.4
35-A	1.48	-.111	23.4	4.18	.0410	18.2	7.3
35-B	1.69	-.076	24.5	4.29	.0217	11.6	4.6

* Tangent chord line used as reference.
** Chord line based on original Clark Y chord.
*** Arbitrary chord line.

From tests in the variable density tunnel of the National Advisory Committee for Aeronautics.

COMPUTATION OF AIRFOIL CHARACTERISTICS

1	C_L	−1.0	−.8	−.6	−.4	−.2	0	.2	.4	.6	.8	1.0	1.2	1.4	1.6	1.8	2.0	2.2
2	α_6																	
3	$\Delta\alpha = 18.24\ KC_L$																	
4	$\alpha =$ (2) + (3)																	
5	C_{D_6}																	
6	$\Delta C_{Di} = .318\ KC_L^2$																	
7	$C_D =$ (5) + (6)																	
	$C_{D\,ext.} =$ (19) + (20)																	
8	COS α = COS (4)																	
9	SIN α = SIN (4)																	
10	C_L COS α = (1) × (8)																	
11	C_D SIN α = (7) × (9)																	
12	$C_N =$ (10) + (11)																	
13	C_L SIN α = (1) × (9)																	
14	C_D COS α = (7) × (8)																	
15	$C_C =$ (14) − (13)																	
16	C.P. = C.P.$_6$																	
	C.P.$_{ext.}$ = $a - C_{Ma}$/(12)																	
17	$C_{M c/4} =$ (.25 − (16)) × (12)																	
18	$C_{Ma} =$ (17) + (a − .25) × (12)																	
19	$C_{Di} =$ (6)/KR																	
20	$C_{D_o} =$ (7) − (19)																	
21	ΔC_{L_u} (BIPLANE)																	
22	ΔC_{L_ℓ} (BIPLANE)																	
23	$C_{L_u} =$ (1) + (21)																	
24	$C_{L_\ell} =$ (1) + (22)																	

R = K = $1/R - 1/6$ = a = **TABLE — II** ΔC_{L_u} ΔC_{L_ℓ} =

TABLE III

DETERMINATION OF RESULTANT AIR FORCES

NO.	ITEM	SEMI-SPAN Root						Tip
(1)	Span = b	0						
(2)	Chord = C							
(3)	R_b							
(4)	$R_b\ C = 3 \times 2$							
(5)	$R_b\ C\ b = 4 \times 1$							
(6)	x							
(7)	$R_b\ Cx = 4 \times 6$							
(8)	$C^2 = 2^2$							
(9)	C_M							
(10)	$C_M\ C^2 = 9 \times 8$							

K_b =

$\bar{b}$ =

$\bar{x}$ =

MAC =

TABLE IV

BALANCING COMPUTATIONS

(See Fig. 13 for symbols)

No.	Item	V_L = ft/sec. I	II	V_g = ft/sec. III	IV
(1)	W = gross weight, pounds				
(2)	$q = .00119\ V^2$				
(3)	s = 1/A_W				
(4)	q/s = 2/3				
(5)	n_1 = applied wing load factor				
(6)	C_N = 5/4				
(7)	α where $C_L = C_N$				
(8)	C_C				
(9)	n_{x_1} = 8 x 4				
(10)	$n_{x_4} = F_{pr}$/1				
(11)	C'_m = design moment coefficient				
(12)	m_1 = 11 x 4				
(13)	n_3 = tail load factor				
(14)	n_2 = 5 - 13 = net load factor				
(15)	n_{x_2} = 9 - 10 = chord load factor				
(16)	T = 1 x 13 = tail load				

TABLE V

COMPUTATION OF NET UNIT LOADINGS (CONSTANTS)

		Stations Along Span				
1	Distance from root, inches					
2	C'/144 = (chord in inches) /144					
3	f, fraction of chord					
4	r, " " "					
5	b = r - f = 4 - 3					
6	a, fraction of chord (a.c.)					
7	j, " " " *					
8	e = unit wing wt., lbs/sq.ft.*					
9	r - a = 4 - 6					
10	a - f = 6 - 3					
11	r - j = 4 - 7					
12	j - f = 7 - 3					
13	C'/144 b = 2 / 5					

* These values will depend on the amount of disposable load carried in the wing.

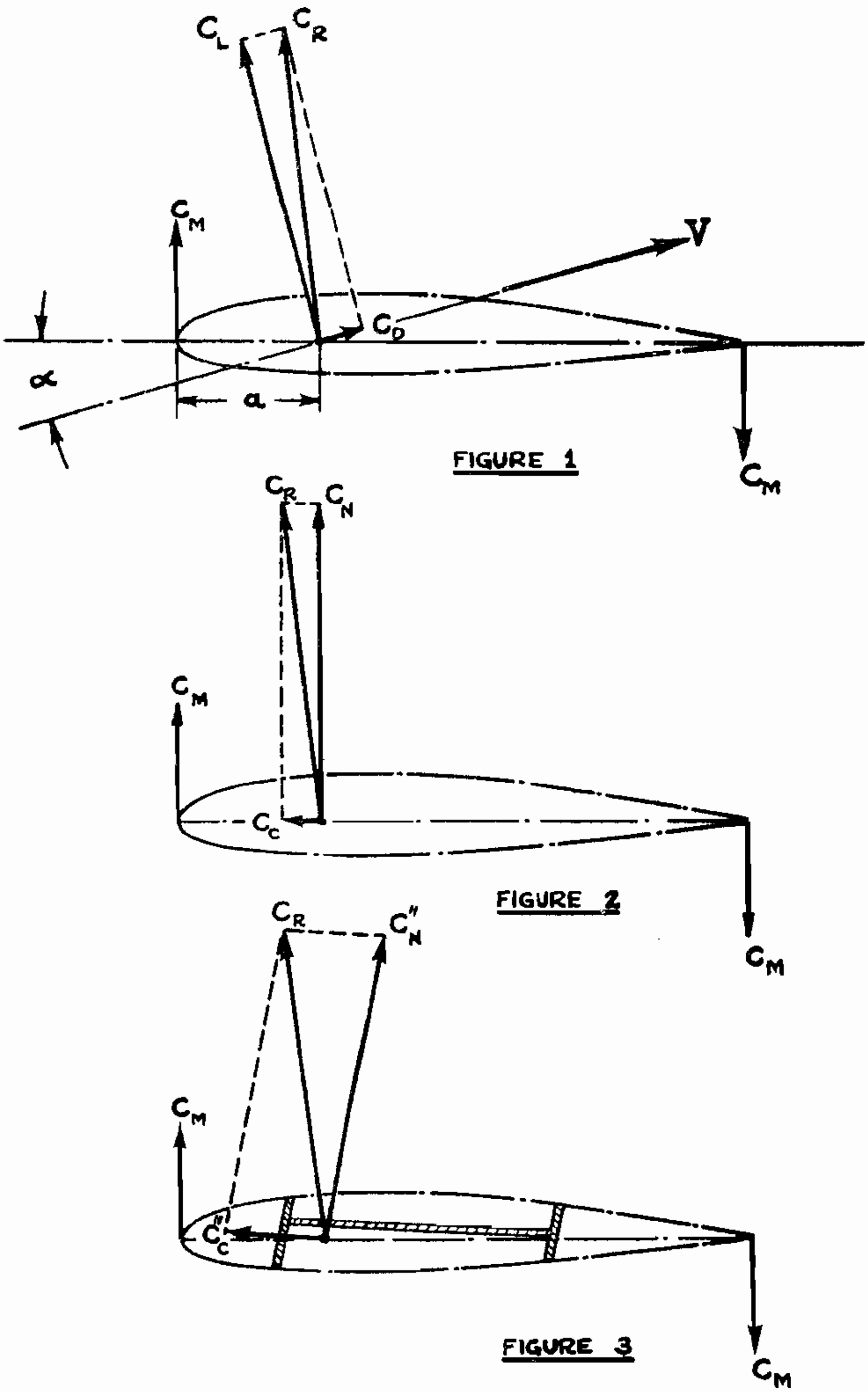
TABLE VI

COMPUTATION OF NET UNIT LOADINGS (VARIABLES)

CONDITION ----

q	$C_{N_I(etc)}$	C'_C	C'_M or C.P.!	n_z	n_{x_z}

		(Refer also to Table V)	Distance b from root					
	14	$C_{N_b} = C_{N_I(etc)} \times R_b/K_b$						
	15	C_{M_a} (variation with span)						
Front Spar	16	14 x 9						
	17	16 + 15						
	18	17 x q						
	19	n_z x 8 x 11						
	20	18 + 19						
	21	y_f = 20 x 13, lbs/inch						
Rear Spar	22	14 x 10						
	23	22 - 15						
	24	23 x q						
	25	n_z x 8 x 12						
	26	24 + 25						
	27	y_r = 26 x 13, lbs/inch						
Chord Load	28	C_C (variation with span)						
	29	28 x q						
	30	n_{x_z} x 8						
	31	29 + 30						
	32	y_c = 31 x 2, lbs/inch						



FIGS. 1, 2, and 3.– ILLUSTRATION OF AIRFOIL FORCE COEFFICIENTS

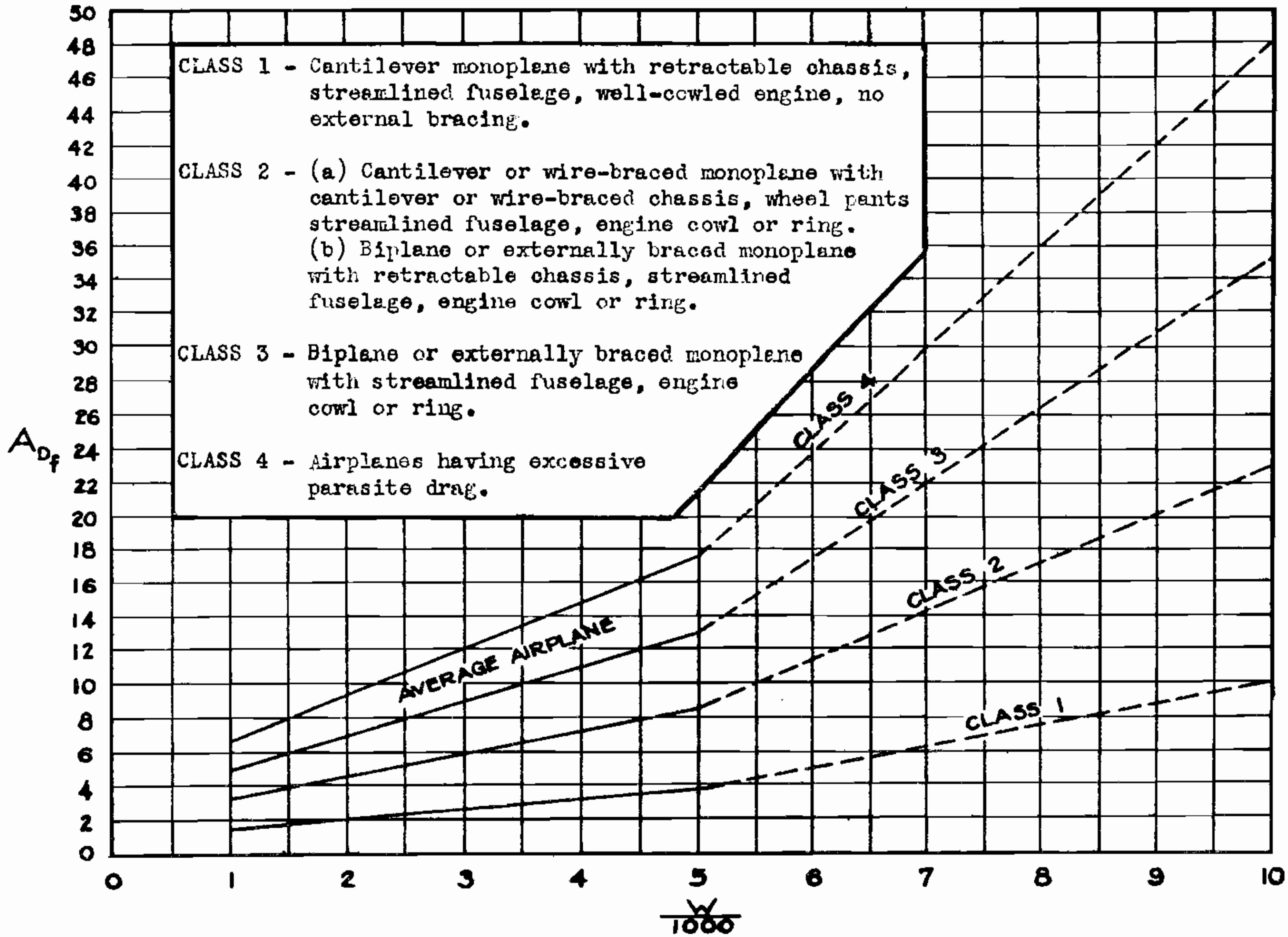


FIGURE 4 - VARIATION OF FUSELAGE DRAG AREA WITH GROSS WEIGHT

CORRECTED AIRFOIL CHARACTERISTICS

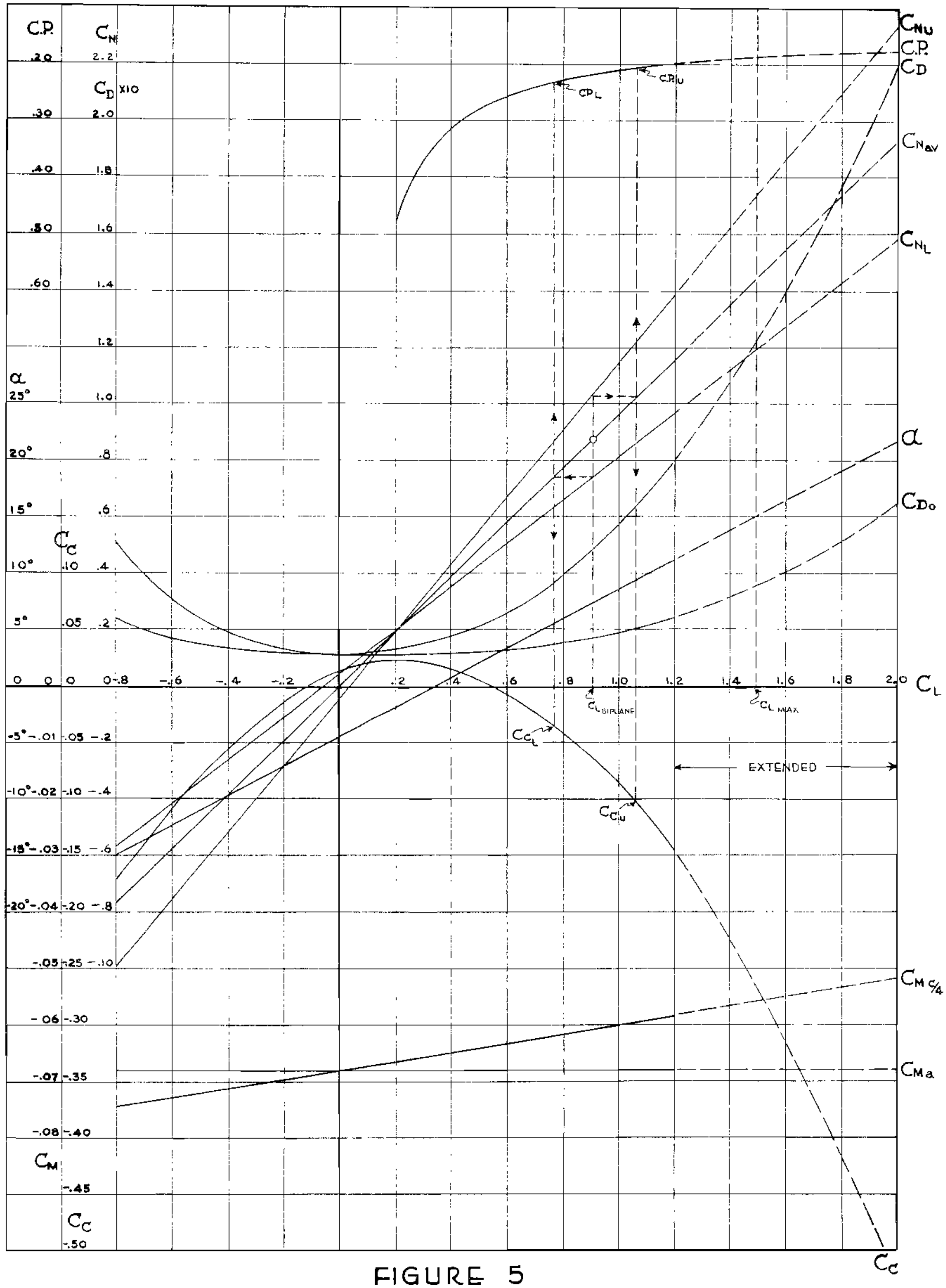


FIGURE 5

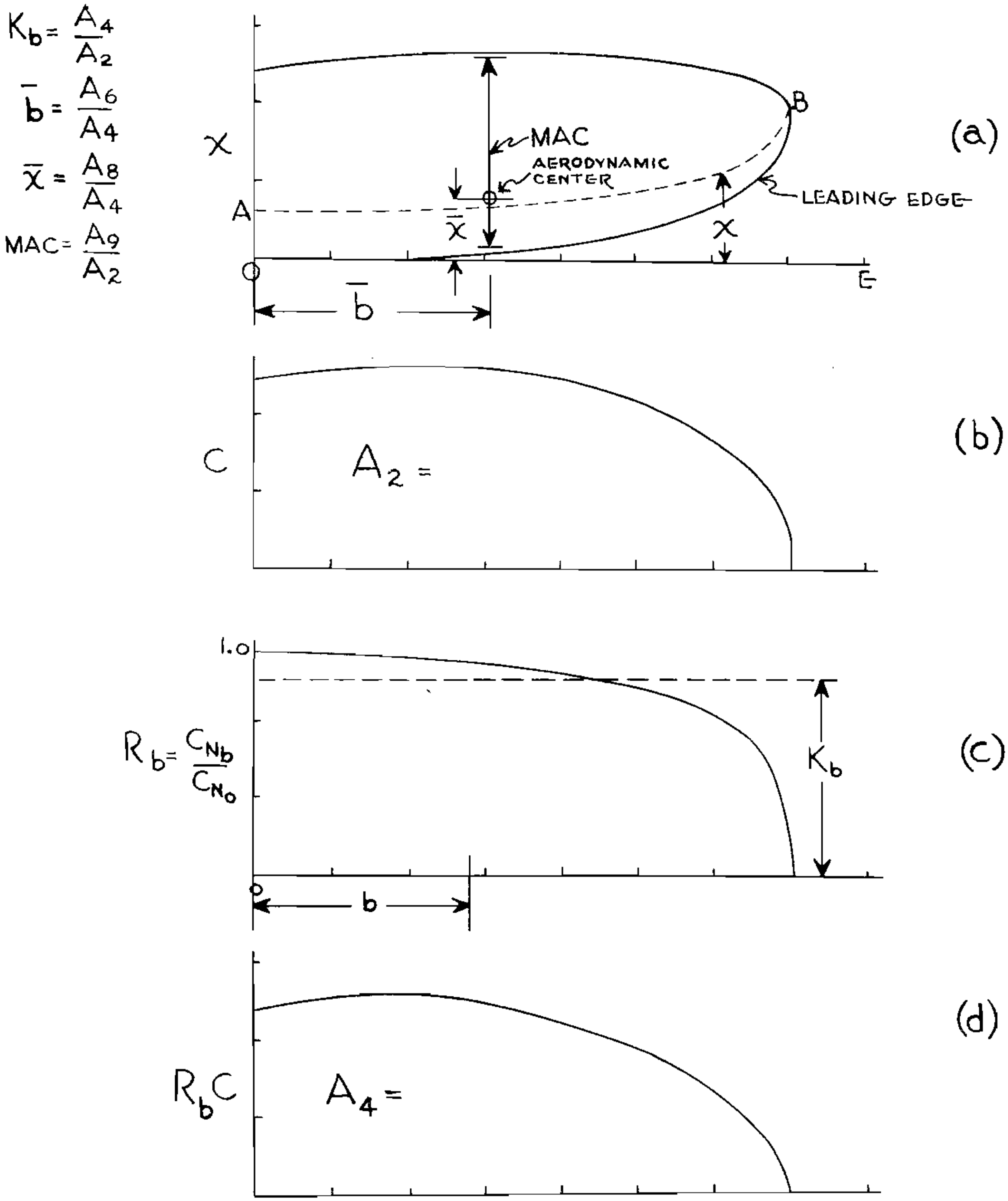


FIG. 6 - DETERMINATION OF EFFECTIVE NORMAL FORCE COEFFICIENT

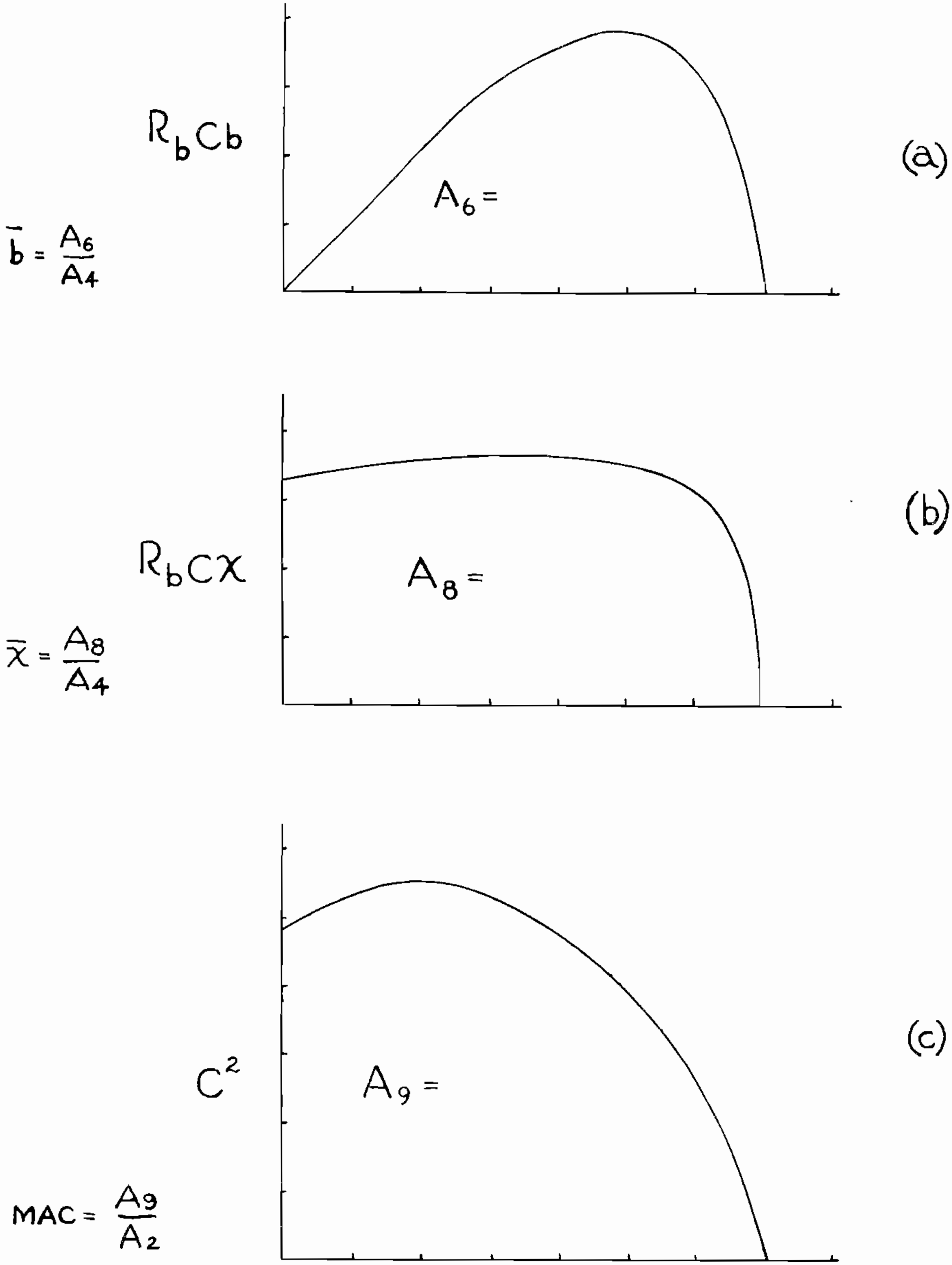


FIG. 7 - DETERMINATION OF MAC AND MEAN AERODYNAMIC CENTER

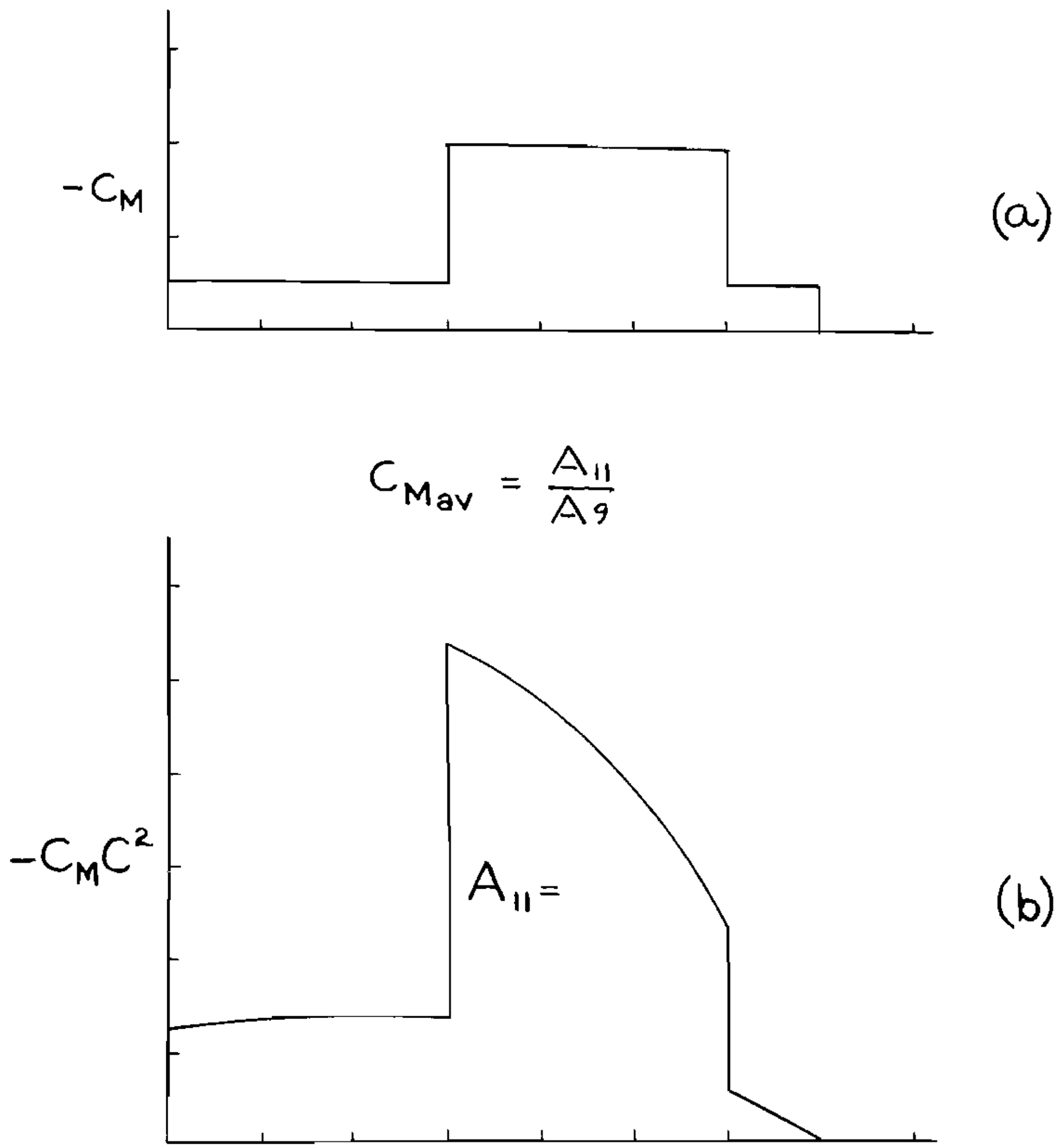
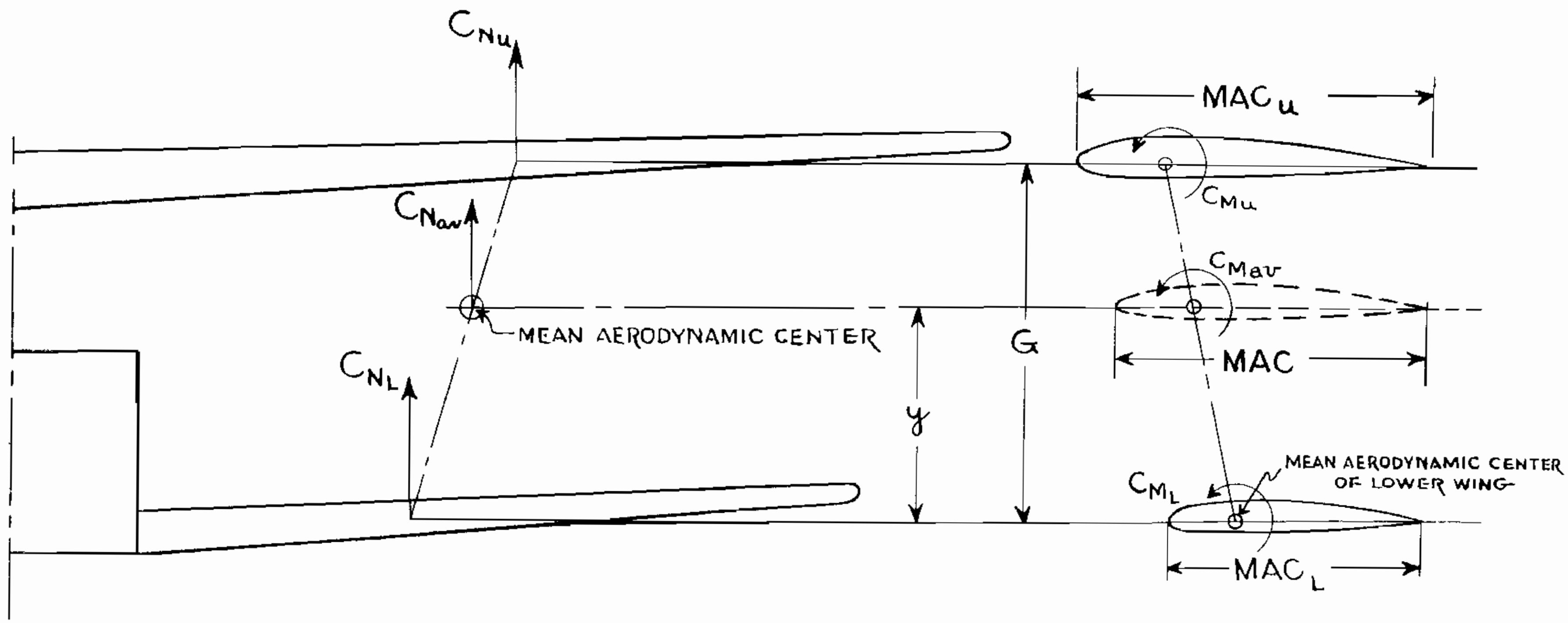


FIG. 8 - DETERMINATION OF MEAN EFFECTIVE MOMENT COEFFICIENT



(a) $$y = \left(\frac{C_{N_u} A_u}{C_{N_u} A_u + C_{N_L} A_L} \right) G$$

(b) $$MAC = \frac{(MAC)_u A_u + (MAC)_L A_L}{A_u + A_L}$$

(c) $$C_{M_{av}} = \frac{C_{M_u} A_u (MAC)_u + C_{M_L} A_L (MAC)_L}{A_u (MAC)_u + A_L (MAC)_L}$$

FIG. 9 – RESULTANT FORCES ON A BIPLANE

DESIGN FLYING CONDITIONS

	BASIC				SUPPLEMENTARY			MODIFICATIONS	
CONDITION	I	II	III	IV	V	VI$_{(1)}$ [1]	VI$_{(2)}$	I$_{(1)}$ [2]	III$_{(1)}$
PERTINENT FOR	WINGS, FUSELAGE, TAIL, AS SPECIFIED				EXT. BRACING	DRAG TRUSS	REAR SPAR	SPARS	REAR LIFT TRUSS
TITLE	POS. HAA	NEG. HAA	POS. LAA	NEG. LAA	INVERTED	DIVING		POS. HAA	NEG. HAA
REFERENCE AB 7-A	SEC. 15 (B)	SEC. 15 (C)	SEC. 15 (D)	SEC. 15 (E)	SEC. 19 (A)	SEC. 19 (B) (1)	SEC. 19 (B) (2)	SEC. 18 (A)	SEC. 18 (B)
REFERENCE AB 26	SEC. 11 (B)	SEC. 11 (C)	SEC. 11 (D)	SEC. 11 (E)	SEC. 15 (B)	SEC. 15 (C) (1)	SEC. 15 (C) (2)	SEC. 14 (B)	SEC. 14 (B)
DESIGN SPEED	V_L	V_L	V_g	V_g	V_L	V_g	V_g	V_L	V_L
Δn [3] a)	$.036 m_6 K_R \frac{V_L}{\Delta}$		$.018 m_6 K_R \frac{V_g}{\Delta}$		$.50 \Delta n_{Ia}$				
		Δn_{Ia}		Δn_{IIIa}					
b)	$\left[.77 + \frac{32000}{W+9200} \right] K_1$		$.60 \Delta n_{Ib}$		$.25 \Delta n_{Ib}$				
LOAD FACTOR n	$1+\Delta n$ [4]	$1-\Delta n$	$1+\Delta n$ [5]	$1-\Delta n$	$-1-\Delta n$ [6]	ZERO	$C_N \frac{q}{\Delta}$	n_I	n_{III}
LIFT COEFFICIENT	$n \frac{\Delta}{q}$	$n \frac{\Delta}{q}$	$n \frac{\Delta}{q}$	$n \frac{\Delta}{q}$	$n \frac{\Delta}{q}$	ZERO	$C_N @ C_{C_{MAX}}$	C_{N_I}	$C_{N_{III}}$
MOMENT COEFFICIENT		C_M	$C_M - .01$	$C_M - .01$	$C_{M_a} = 0$	C_M	$C_M - .01$		$-.06$ [7]
CENTER OF PRESSURE	(8)	C.P.			.25	C.P.		(9)	
CHORD COEFFICIENT	$C_C - 0.20 C_N$ [10]	C_C [11]	C_C [11]	C_C [11]	ZERO	$C_{C_{MAX}} + 0.01$	$C_{C_{MAX}} + .01$	$C_C - 0.20 C_N$ [12]	$C_{C_{III}}$

(1) IF CONDITION VI_2 IS INVESTIGATED, NEGLECT VI_1

(2) MAY BE CRITICAL FOR REAR SPAR AND FOR TENSION IN FRONT SPAR

(3) WHERE TWO VALUES ARE GIVEN, USE THE GREATER

(4) SHALL NOT BE LESS THAN 2.50

(5) SHALL NOT BE LESS THAN 2.00

(6) MINIMUM NEGATIVE VALUE = 1.50

$$K_R = \frac{4}{3 + 6/R}$$

K_1 FROM AB 7-A FIG 1

(7) APPLIES TO PORTION OF WING INCORPORATING AILERONS

(8) MOST FORWARD BETWEEN C_{N_I} AND $C_{L\,MAX}$

(9) MOST REARWARD BETWEEN C_{N_I} AND $C_{L\,MAX}$

(10) USE GREATER NEGATIVE VALUE

(11) WHEN C_C IS POSITIVE OR LESS THAN -0.02, IT MAY BE ASSUMED EQUAL TO ZERO

(12) USE SMALLER NEGATIVE VALUE

ALL LOADS AND FORCES ARE APPLIED

FIGURE 10

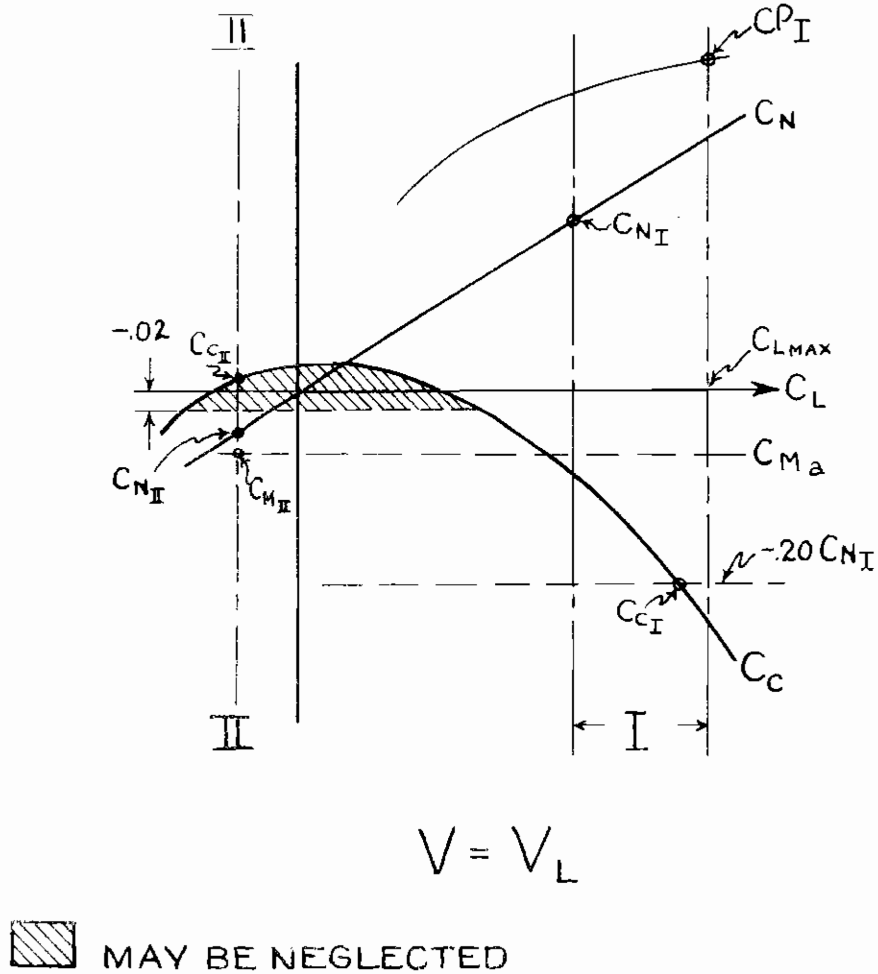


Fig. 11 Conditions I & II

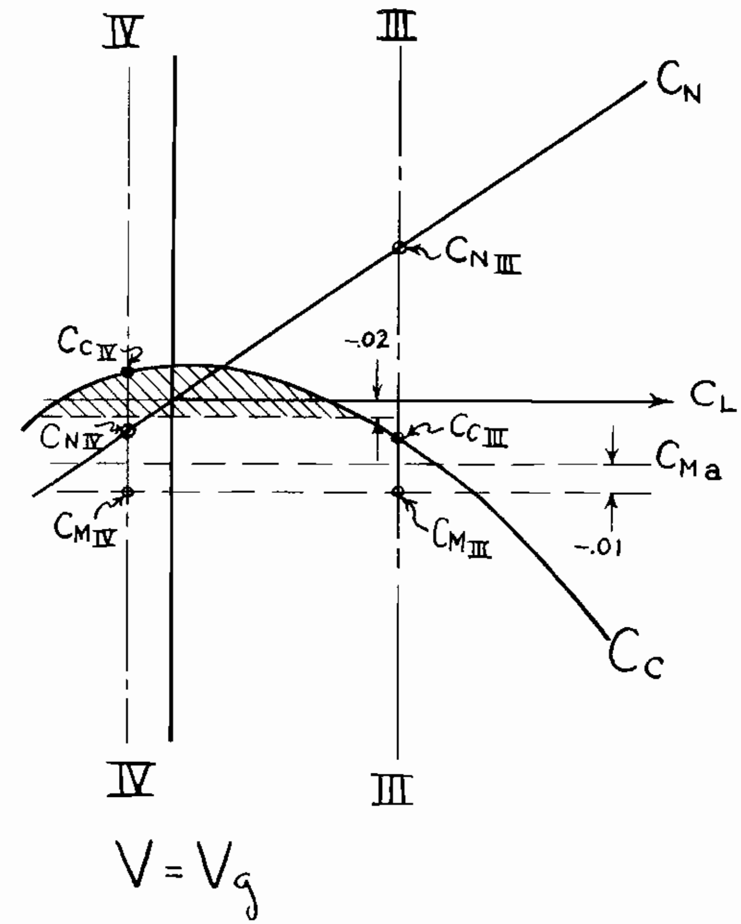


Fig. 12 Conditions III & IV

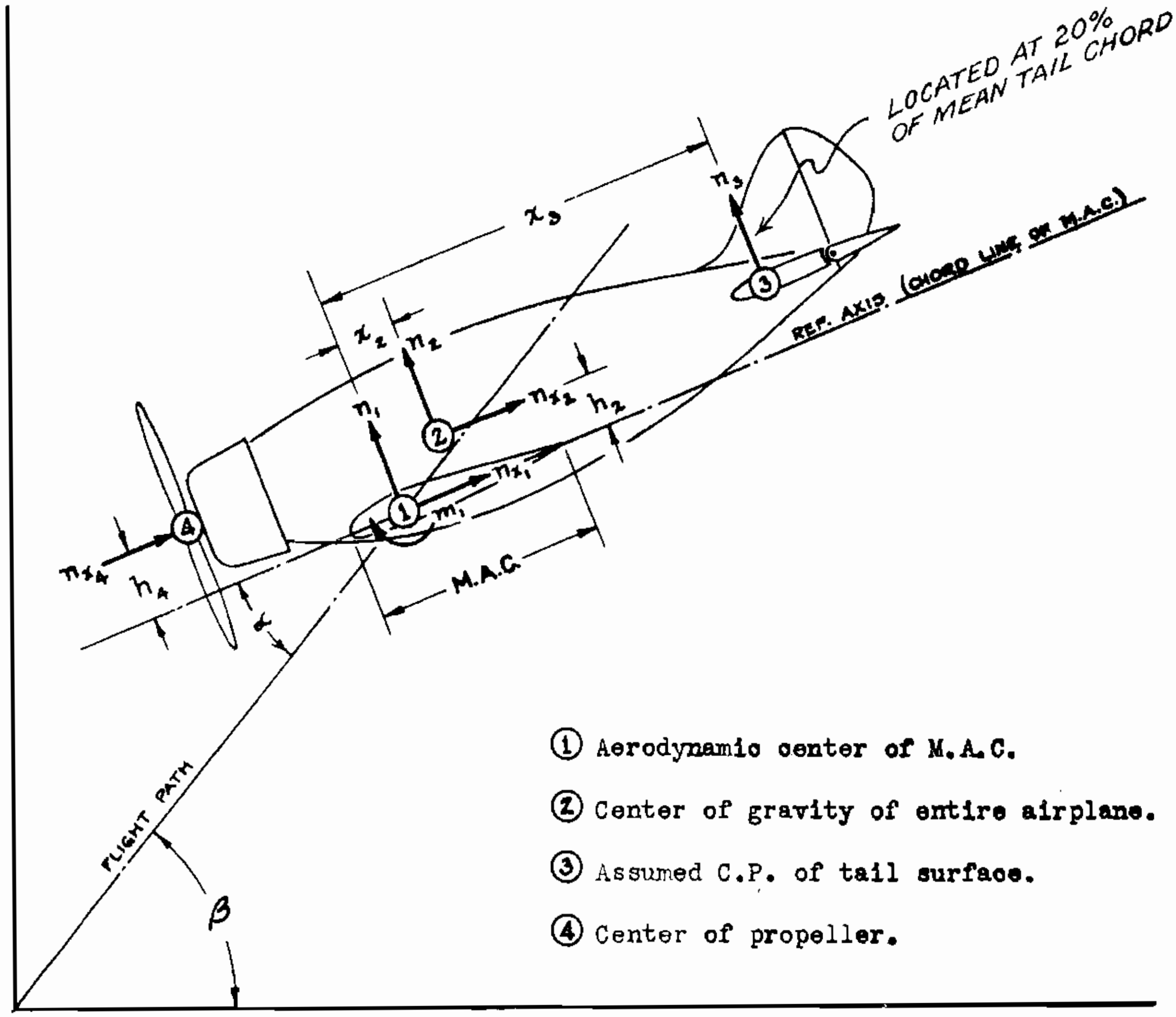


FIGURE 13 - BASIC FORCES IN FLIGHT CONDITIONS.

α = angle of attack, degrees (shown positive).

β = gliding angle, degrees.

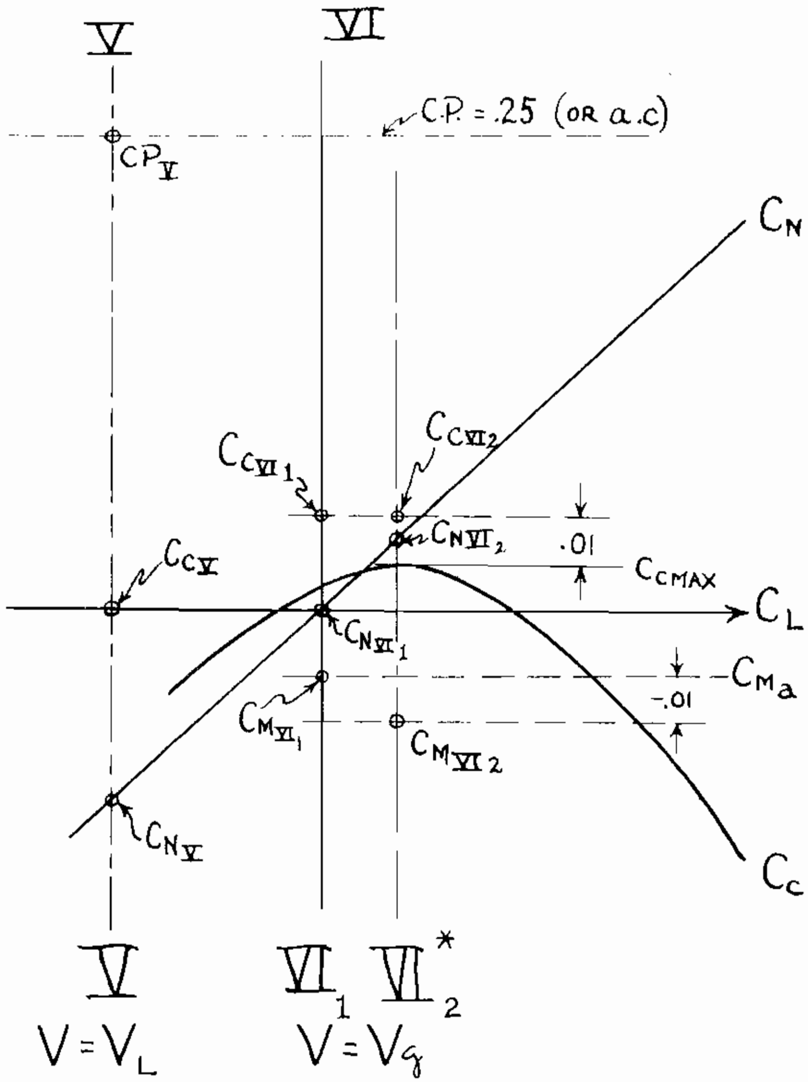
n = force/W (positive upward and rearward).

m = moment/W (positive clockwise as shown).

x = horizontal distance from (1) (positive rearward).

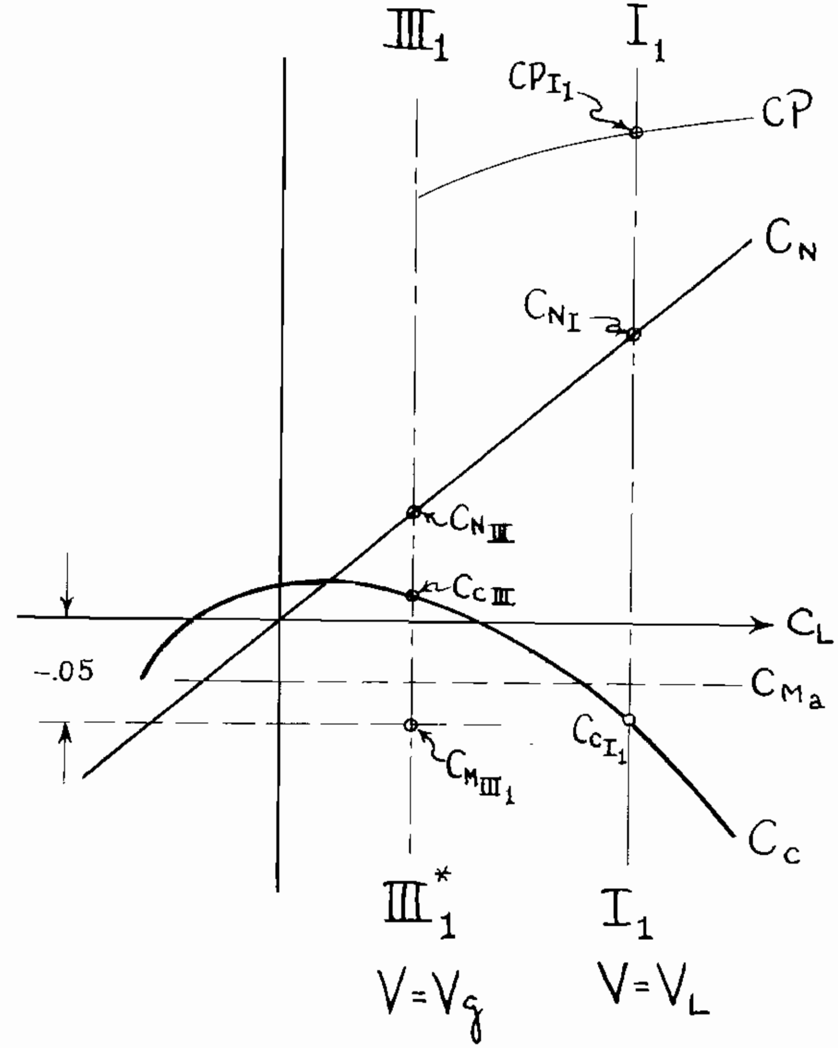
h = vertical distance from (1) (positive upward).

All distances are expressed in terms of the M.A.C.



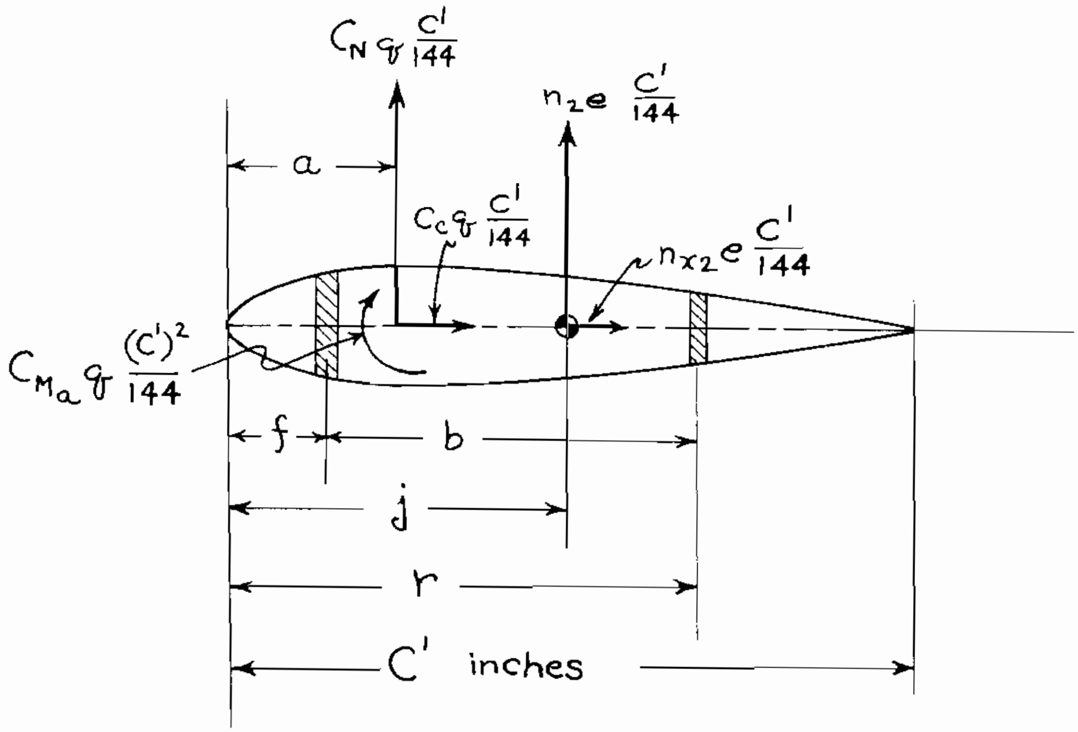
* IF CONDITION VI 2 IS INVESTIGATED, NEGLECT CONDITION VI_1

Fig. 14 Conditions V & VI



* SEE AB 7-A SEC 18(B)(1)

Fig. 15 Conditions I_1 & III_1



ALL VECTORS ARE SHOWN IN POSITIVE SENSE

FIG. 16

UNIT SECTION OF A CONVENTIONAL 2-SPAR WING

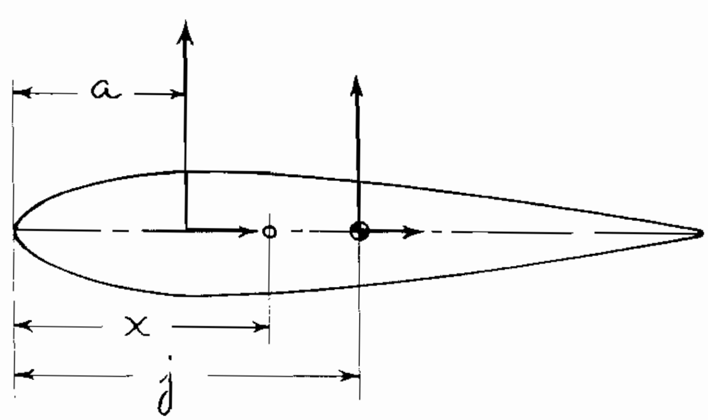


FIG. 17

SECTION SHOWING LOCATION OF ELASTIC AXIS

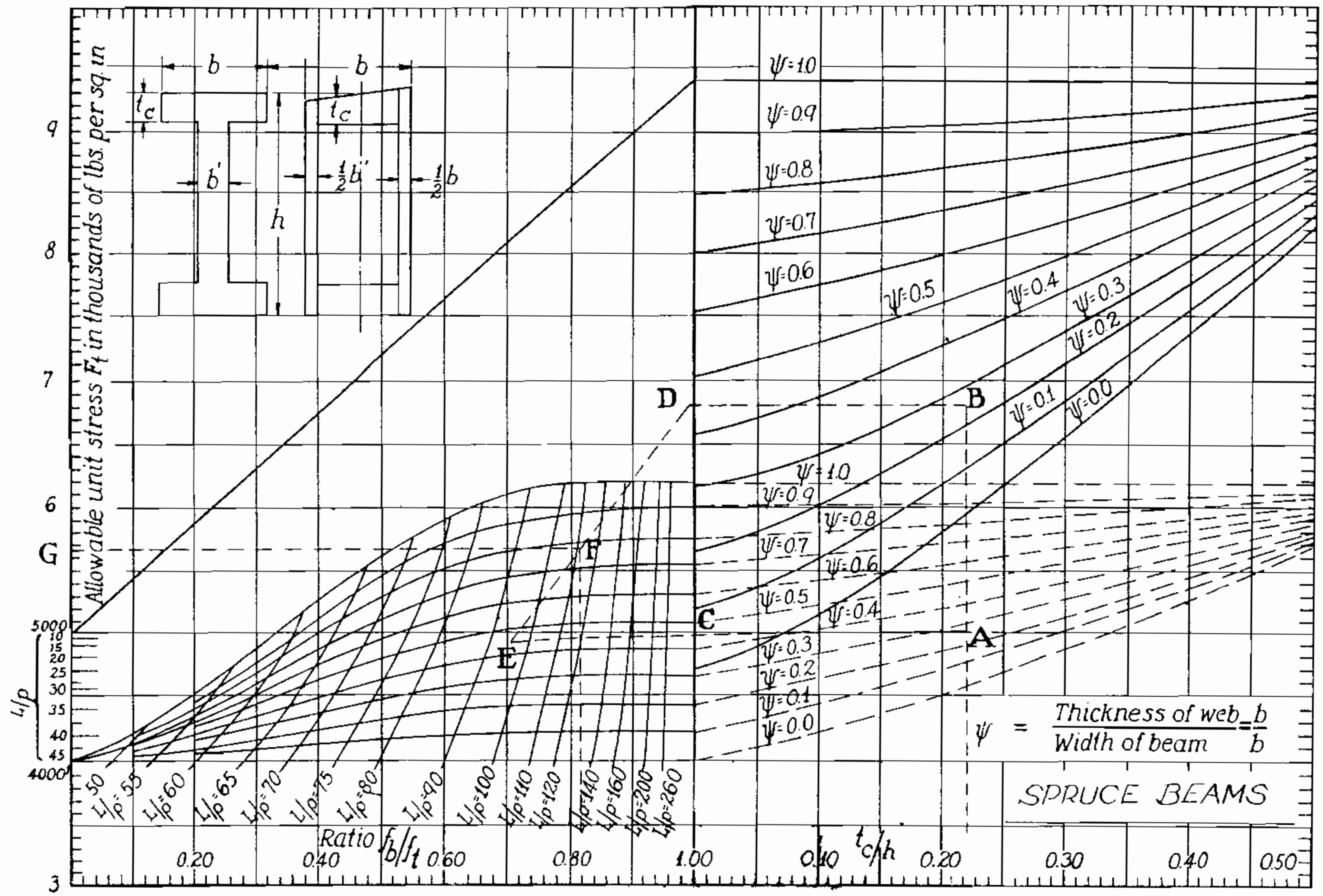


FIG. 18 - ALLOWABLE STRESSES FOR WOOD SPARS

CONTROL SURFACE DESIGN CONDITIONS

SURFACE	HORIZONTAL (1)			VERTICAL (1)		AILERON (1)	FLAP	TAB
CONDITION	BALANCING	MANEUVERING	DAMPING	MANEUVERING	DAMPING	MANEUVERING	BALANCING	BALANCING
REFERENCE AB 7-A	SEC. 26 (A)	SEC. 26 (B)	SEC. 26 (C)	SEC. 27 (A)	SEC. 27 (B)	SEC. 28 (A)	SEC. 29 (A)	SEC. 29 (B)
DESIGN SPEED	V_g	V_p		V_p (8)		V_p	V_f	V_L
C_N		-0.55 $+0.35$ (2)		0.45		± 0.45	+1.60 (MAX)	(3)
$\overline{w}$ (NET)		$-V_p^2/1530$ $+V_p^2/2400$		$\frac{V_p^2}{1870}$		$\pm\frac{V_p^2}{1870}$	$+\frac{V_f^2}{525}$	$C_{N_{tab}}\frac{V_L^2}{840}$
LOAD	$T + P$ (5)	$\overline{w} A_t$ (4)	SAME AS LOAD ON STABILIZER IN PREVIOUS CONDITION	$\overline{w} A_t$ (4)	SAME AS LOAD ON FIN IN PREVIOUS CONDITION	$\overline{w} A_a$	$\overline{w} A_f$	$\overline{w} A_{tab}$
LIMITS MAX.	(6)	(9)		(9)		(13)		
LIMITS MIN.	(7)	(10)		(11)	$\frac{V_L}{13}\left[\frac{3}{4+6/R}\right]$ (12)	(11)		
DISTRIBUTION	FIG. 4	FIG. 5	FIG. 4	FIG. 5	FIG. 4	FIG. 6	(14)	UNIFORM

$$V_p = V_s + K_p (V_L - V_s) \qquad K_p = 0.15 + \frac{5400}{W+3000} \text{ (15)}$$

(1) FOR EFFECTS OF TABS, SEE AB 7-A SEC. 29 (B).
(2) PLUS FORCE ACTS UP, NEGATIVE DOWN.
(3) SEE NACA TEST DATA.
(4) SUBSCRIPT DENOTES TOTAL SURFACE.
(5) T FROM CONDITION III OR IV, WHICHEVER GREATER, P=0.4T
(6) P CORRESPONDS TO OPPOSITE LOAD ON ELEVATOR WITH 150 LB. FORCE ON CONTROL COLUMN.
(7) P CORRESPONDS TO OPPOSITE LOAD ON ELEVATOR WITH MINIMUM FORCE ON CONTROL COLUMN [SEE AB 7-A SEC 31 (2)]
(8) FOR MULTI-ENGINED AIRPLANES, V_p FOR VERTICAL SURFACE $= 0.9 V_L\left[\frac{n-1}{n}\right]^{\frac{1}{3}}$
(9) HINGE MOMENT CORRESPONDING TO 200 lb. CONTROL COLUMN FORCE.
(10) $\overline{w}$ EQUALS ±15 LBS./SQ.FT., EXCEPT (7) MAY BE CRITICAL FOR ELEVATOR.
(11) $\overline{w}$ EQUALS 12 LBS./SQ. FT.
(12) ASSUME R = 2.0 WHEN IT IS LESS.
(13) HINGE MOMENT CORRESPONDING TO 80 LB. CONTROL COLUMN FORCE.
(14) USE C.P. FROM NACA TEST DATA.
(15) MINIMUM EQUALS 0.5

ALL FORCES AND LOADS ARE APPLIED

FIG. 19

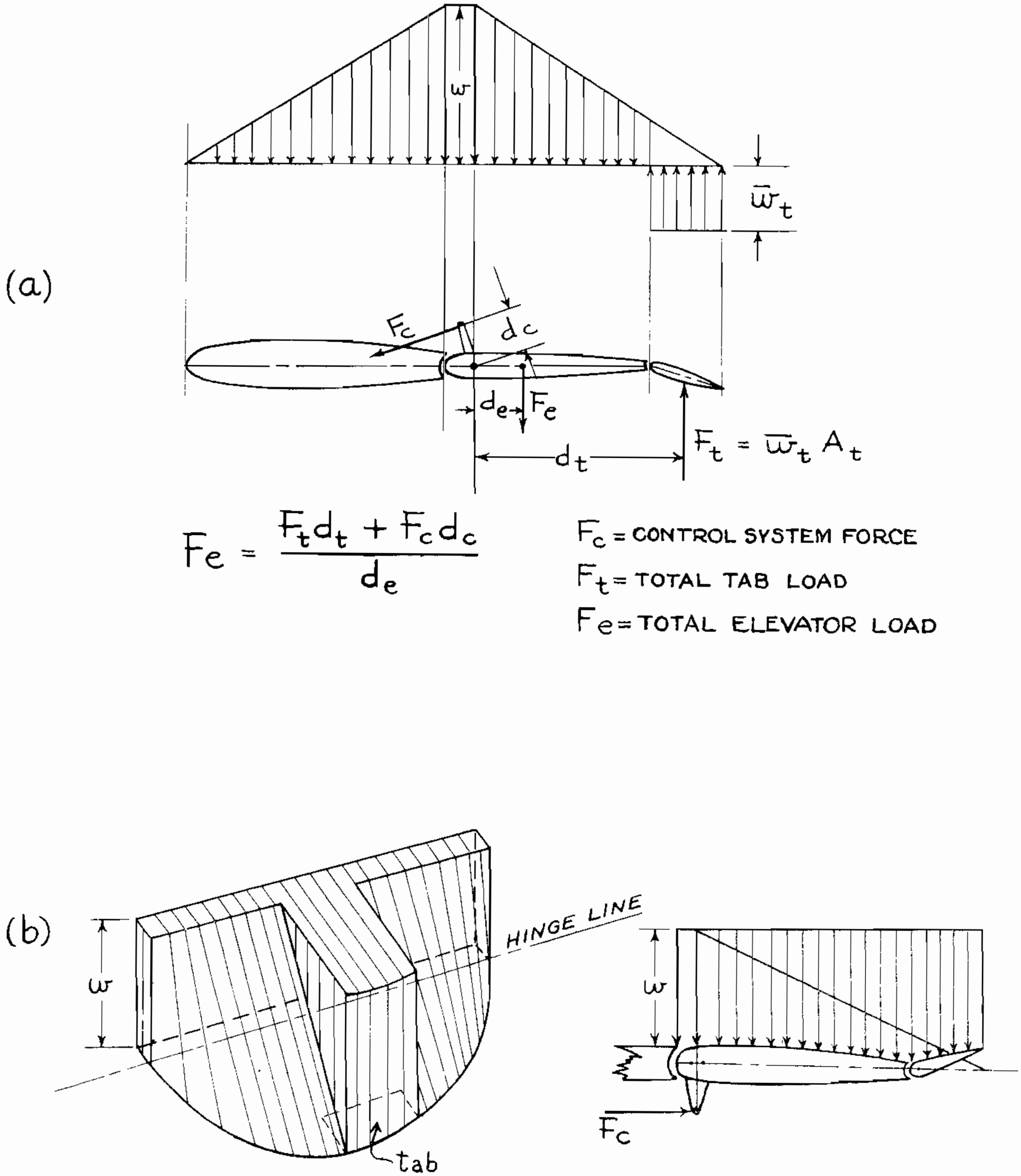


FIG 20

CONTROL SURFACE LOAD DISTRIBUTION WITH TABS

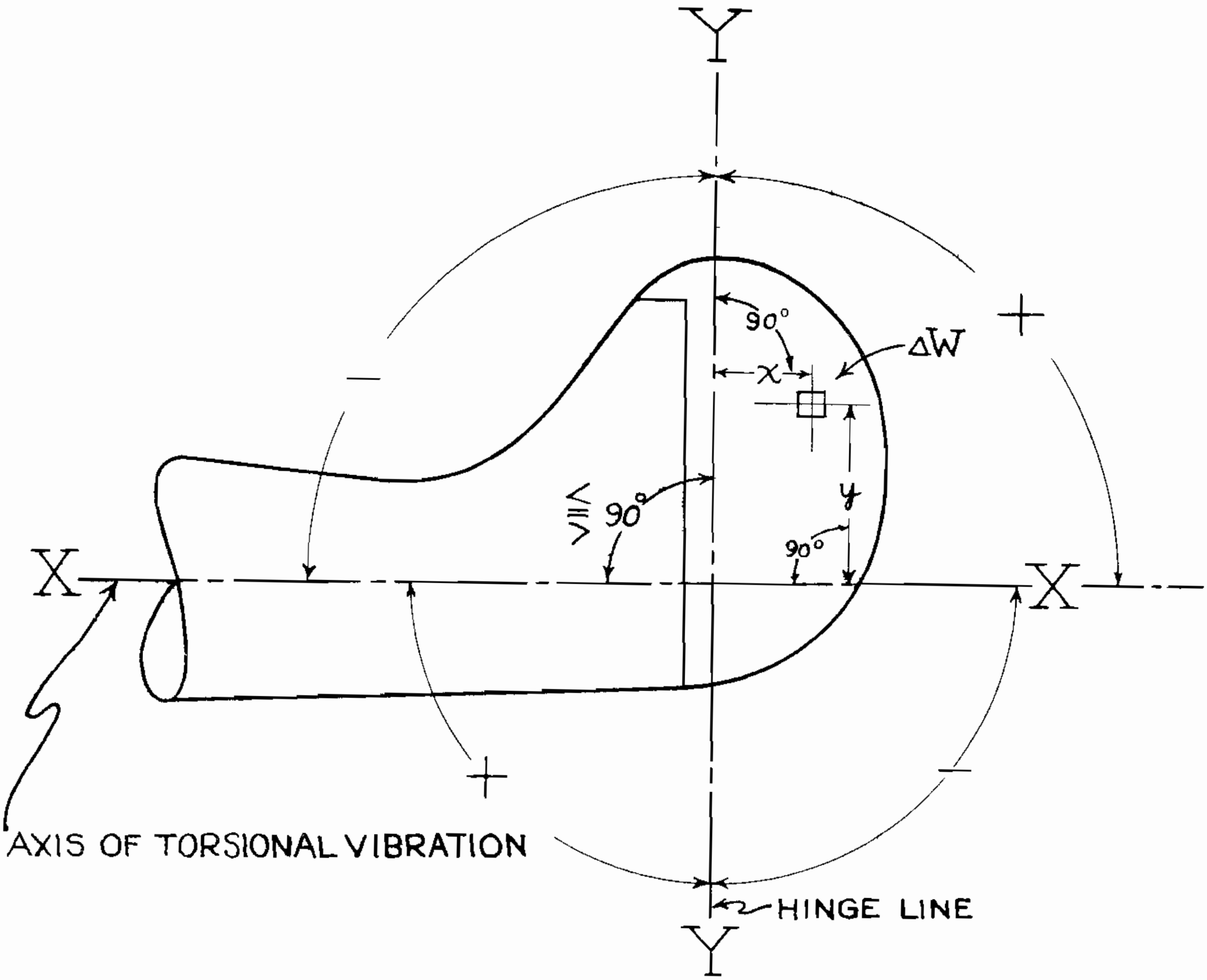


FIG. 21

DYNAMIC BALANCING OF CONTROL SURFACES

CONTROL SYSTEM DESIGN CONDITIONS

SYSTEM	ELEVATOR	RUDDER	AILERON (1)	FLAP OR TAB
REFERENCE AB 7-A	SEC. 31 (A)	SEC. 31 (B)	SEC. 31 (C)	SEC. 31 (D)
REFERENCE AB 26	SEC. 30-32	SEC. 30-32	SEC. 30-32	SEC. 30-32
CONTROL FORCE CORRESPONDING TO	1.25 CRITICAL ELEVATOR LOAD	1.25 CRITICAL (2) RUDDER LOAD	1.25 CRITICAL AILERON LOAD	1.25 CRITICAL FLAP OR TAB LOAD
LIMITING MAXIMUM CONTROL FORCE (LBS.)	200	200	80	
LIMITING MINIMUM CONTROL FORCE (LBS.)	70+.06 [W-500] (3)	130	30+.02 [W-500] (4)	

ALL LOADS AND FORCES ARE <u>APPLIED</u>

(1) SEE ALSO AB 7-A SEC. 31 (C)(3) AND (4)

(2) WHEN ENGINES NOT IN PLANE OF SYMMETRY, DESIGN CONTROL FORCE IS 200 LBS.

(3) NEED NOT EXCEED 130

(4) NEED NOT EXCEED 50

FIG. 22

LANDPLANE LANDING CONDITIONS

CONDITION	LEVEL	3—POINT	SIDE (1)	BRAKED
REFERENCE AB 7-A	SEC. 35	SEC. 36	SEC. 37	SEC. 38
REFERENCE AB 26	SEC. 33	SEC. 33	SEC. 34	SEC. 33
LOAD FACTOR n (2) (APPLIED)	$2.80 + \frac{9000}{W+4000}$ (3) $3.00 + 0.135\,S$ (4)	SAME AS LEVEL	$\frac{1}{10}$ LEVEL	1.33
ATTITUDE	PROPELLER AXIS HORIZONTAL	3—POINT	PROPELLER AXIS HORIZONTAL	3—POINT (5)
VERTICAL COMPONENT	nW (6)	nW (7)	ZERO	nW
REARWARD COMPONENT	RESULTANT (8) THRU C.G.	ZERO	ZERO	.55 VERTICAL
SIDE COMPONENT	ZERO	ZERO	nW	ZERO
SHOCK STRUT DEFLECTION	50% TRAVEL (9)	SAME	SAME	SAME
TIRE DEFLECTION	50%	SAME	SAME	25%

(1) SIDE COMPONENT ACTS ON ONE WHEEL ONLY (INWARD)
(2) NEED NOT EXCEED 4.33
(3) FOR AIRPLANES OF OVER 1000 LBS. GROSS WEIGHT
(4) FOR AIRPLANES OF UNDER 1000 LBS. GROSS WEIGHT
(5) REACTION AT TAIL EQUALS ZERO
(6) W IN ALL CASES IS GROSS WEIGHT LESS WHEELS AND CHASSIS (C.G. CORRESPONDS)
(7) DISTRIBUTED TO WHEELS AND SKID SO THAT MOMENTS ABOUT C.G. EQUAL ZERO
(8) NEED NOT EXCEED 25% VERTICAL COMPONENT
(9) UNLESS APPARENT MORE CRITICAL CONDITION EXISTS

FIG. 23

86060—34——8

SEAPLANE AND FLYING BOAT LANDING CONDITIONS

COMPONENT	FLOAT			HULL		
CONDITION	INCLINED REACTION	VERTICAL REACTION	SIDE LANDING	STEP LANDING	TWO-WAVE LANDING	BOTTOM LOADING
REFERENCE AB 7-A	SEC 44	SEC 45	SEC 46	SEC 47(A)	SEC 47 (B)	SEE AB 7-A SEC 48
REFERENCE AB 26	SEC 37 (A)	SEC 37 (A)	SEC 37 (A)	SEC 37(B)	SEC 37 (B)	
n (APPLIED)	4.20 (1)	4.33 (1)	4.0	5.33	1.0	
	$3.00 + .133\,s$ (2)					
VERTICAL REACTION	nW (3)	nW (3)	nW (3)		nW	
REARWARD REACTION	$\frac{1}{4}$ VERTICAL	0	0		0	
SIDE REACTION	0	0	$\frac{1}{4}$ VERTICAL	0	0	
RESULTANT	THROUGH CG LESS FLOATS & BRACING			nW	THROUGH STEP	
FACTOR OF SAFETY	1.85 (4) 1.50 (5)		1.50	1.50	1.50	1.50
ATTITUDE	PROP. AXIS OR REFERENCE LINE HORIZONTAL					

(1) FOR SEAPLANES OF 1000 LBS. OR MORE GROSS WEIGHT
(2) FOR SEAPLANES OF LESS THAN 1000 LBS. GROSS WEIGHT
(3) W IS GROSS WEIGHT LESS FLOATS AND BRACING
(4) FOR FLOAT ATTACHMENTS AND FUSELAGE CARRY-THRU MEMBERS
(5) FOR REMAINING STRUCTURAL MEMBERS

FIGURE 24

FUSELAGE DESIGN CONDITIONS

	I	II	III	IV	V	VI	IU	IIIU	VU	HT	VT	LL (2)	3-P	SL	NO	T	I-T	SE	UE	TO	
SIDE TRUSSES	o	o	o	o						o	o	o	o								SIDE TRUSSES
FORWARD TRUSSES					o										o	o	o	o	o		FORWARD TRUSSES
FORWARD BULKHEADS																o	o	o			FORWARD BULKHEADS
TOP AND BOTTOM TRUSSES											o										TOP AND BOTTOM TRUSSES
REAR BULKHEADS											o										REAR BULKHEADS
ENGINE MOUNTS															o	o	o	o	o		ENGINE MOUNTS
NACELLES	o	o	o	o								o				o	o	o	o		NACELLES
MEMBERS ADJ. TO WING (1)	o	o	o	o	o	o	o	o	o	o	o									o	MEMBERS ADJ. TO WING (1)
MEMBERS ADJ. TO L.G. (1)												o	o	o							MEMBERS ADJ. TO L.G. (1)
	POS HAA FLYING	NEG HAA FLYING	POS LAA FLYING	NEG LAA FLYING	INVERTED FLIGHT	SPECIAL DIVING	UNSYMMETRICAL POS HAA FLYING	UNSYMMETRICAL POS LAA FLYING	UNSYMMETRICAL INVERTED FLIGHT	HORIZONTAL TAIL (MANEUVERING)	VERTICAL TAIL	LEVEL LANDING (2)	THREE POINT LANDING	SIDE LANDING	NOSING OVER	TORQUE	POS HAA FLYING PLUS TORQUE	SIDE LOAD ON ENGINE MOUNT	UP LOAD ON ENGINE MOUNT	COMPLETE TURN-OVER	

1) INCLUDES CROSS MEMBERS, BULKHEADS, AND TOP AND BOTTOM TRUSSES.

2) CHECK ALSO MEMBERS ADJACENT TO WING AND LANDING GEAR FOR REACTION FROM ONE WHEEL ONLY IN LEVEL LANDING CONDITION.

THIS CHART IS A GUIDE TO THE CONDITIONS LIKELY TO BE CRITICAL FOR THE MEMBERS NOTED.

FIGURE 25

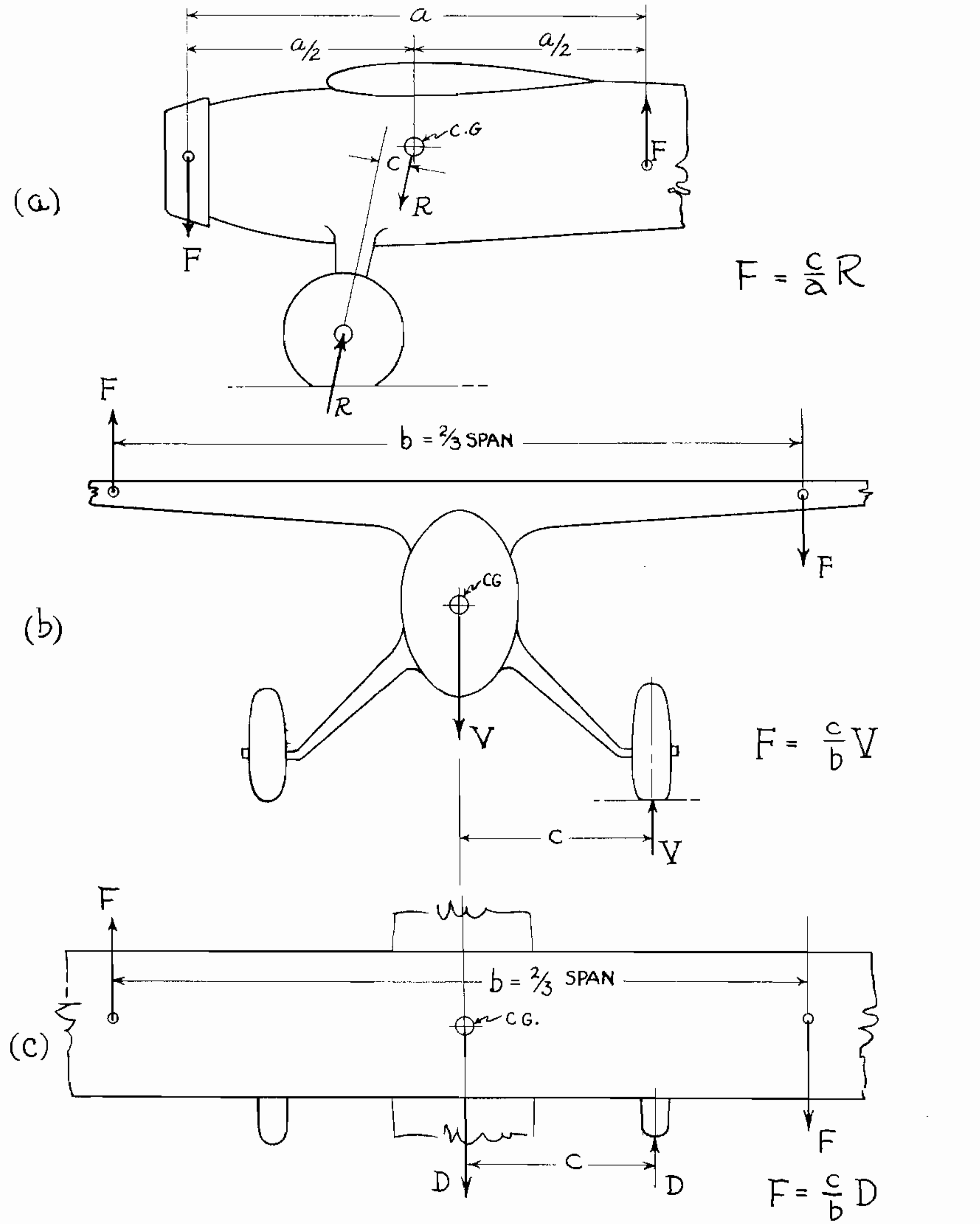
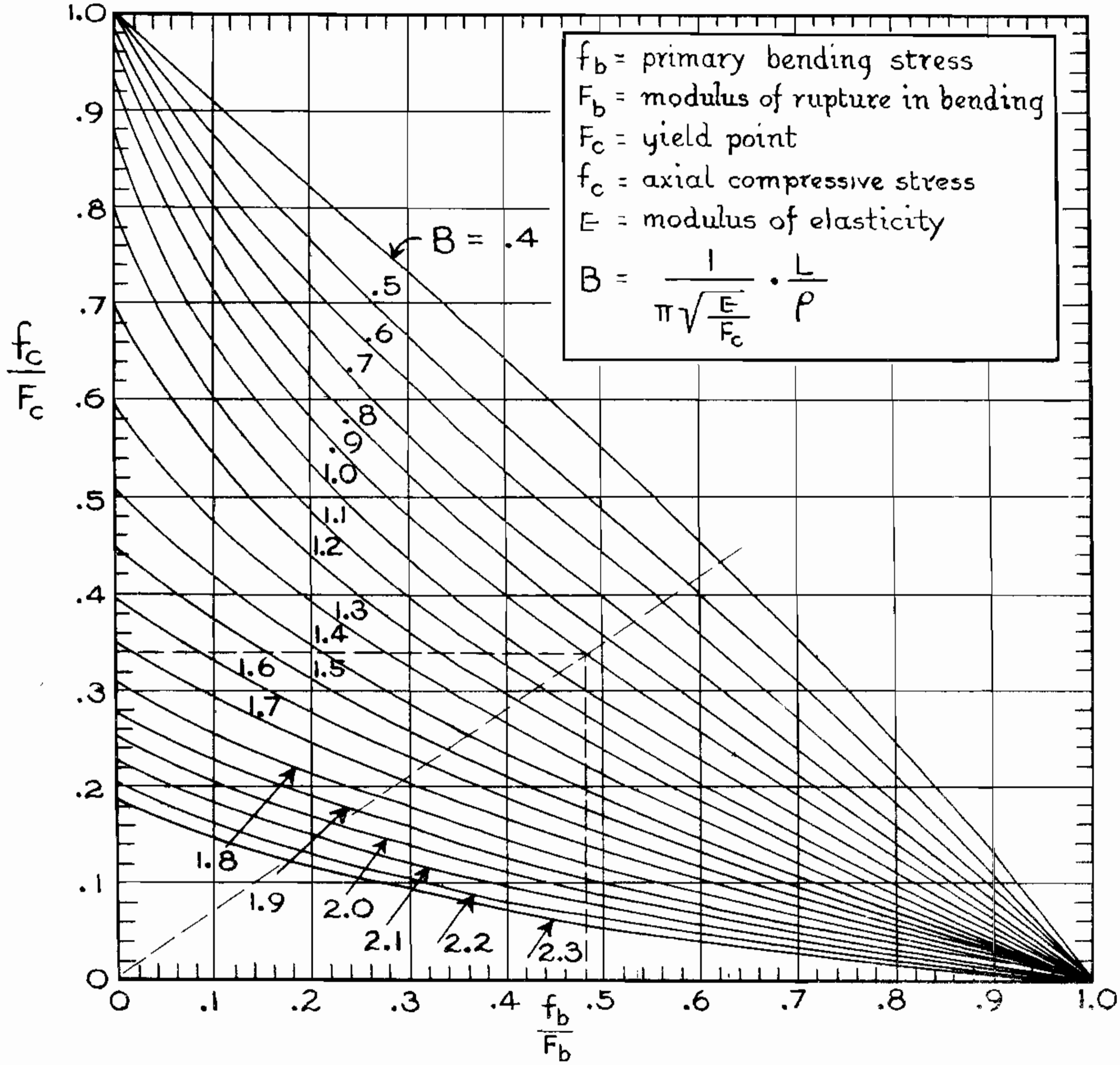


FIG. 26 METHODS OF BALANCING FUSELAGE FOR UNSYMMETRICAL LOADS



BASIC CHART FOR CHROME MOLYBDENUM STEEL TUBES SUBJECTED TO COMBINED BENDING AND COMPRESSION.

FIG. 27

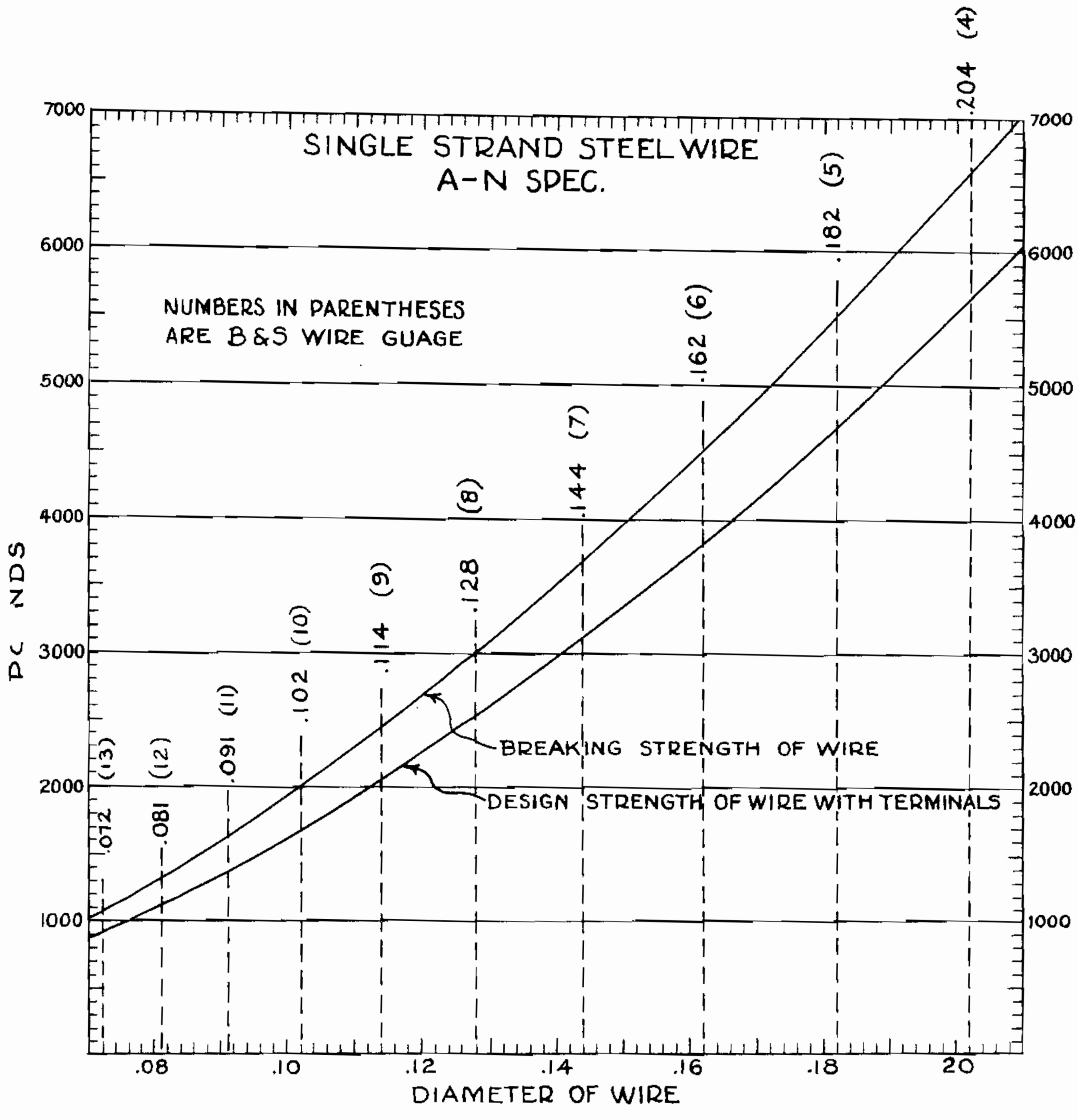


FIG. 28 STRENGTH OF AIRCRAFT WIRE