d. the manufacturer should furnish the Administrator with technical data descriptive of all structural changes, except those of an obviously minor nature, such changes to be substantiated by test, if necessary, and approved prior to resuming the type inspection.

.054 ISSUANCE OF AIRCRAFT SPECIFICATION

1. Upon satisfactory completion of all reports, tests and inspections required to prove compliance with the airworthiness requirements of the Administrator, an Aircraft Specification will be issued for the type and model of the airplane in question. The Aircraft Specification will certify as to the airworthiness of airplanes of the type in question when manufactured and maintained in accordance with the provisions noted thereon.

.055 ISSUANCE OF TYPE CERTIFICATES

1. A type certificate such as is described in CAR 02 will be issued to the applicant upon compliance with the requirements therein.

.056 AUTHENTICATED DATA

1. As a part of the type certificate, the Administrator will furnish the applicant, upon issuance of such certificate, one set of drawing lists on which the seal of the Administrator is impressed. These lists will show acceptance of the drawings as partial proof of the airworthiness of the type of airplane to which they apply.

.06 CHANGES

1. Change, Repair or Alteration of Individual Certificated Airplanes. Change, repair or alteration of a certificated airplane renders such airplane subject to re-certification as to airworthiness in accordance with CAR 18, but does not affect the type certificate on which the airworthiness certification may have been based. As a general rule extensive revisions of the primary structure should not be undertaken without the cooperation of the airplane manufacturer. Changes which appear to be unimportant might seriously affect the structural safety or flying qualities, making the airplane unsafe. The manufacturer is supplied with complete strength calculations from which information regarding the approved member sizes and material specifications can be obtained. Also, the manufacturer may have already obtained the Administrator's approval of the proposed change.

2. Changes by Holder of Type Certificate. The holder of a type certificate should apply for approval of any specific change or revision of the approved drawings or specifications which affect the airworthiness of the airplane and should submit sufficient technical data in the form of strength calculations and strength tests, or both, to demonstrate continued compliance with the airworthiness requirements hereinafter specified. Corrected pages of the drawing lists, in duplicate, should also be submitted. Alternate installations should be so designated and properly indicated on the drawing lists. If, in the opinion of the Administrator, the changes are such as to affect the performance or operating characteristics, appropriate tests may be required. Upon satisfactory proof that the revisions do not render the airplane type unairworthy the Aircraft Specification may be modified to include airplanes embodying the approved changes and sealed copies of the revised drawing list pages will be returned to the applicant. The manufacturer should maintain a record of the airplane serial numbers to which the changes apply.

3. Changes by Persons Other Than Holder of Type Certificate. Changes such as described above, when made by persons other than the holder of the type certificate, are also subject to the procedure outlined above, except that the written consent of the holder of the type certificate should be obtained if it is desired to refer to technical data originally submitted to the Administrator in connection with type certification. With the consent of both the person making the change and the holder of the type certificate, all airplanes manufactured under the type certificate may be made eligible for such change by an appropriate revision of the pertinent Aircraft Specification.

.060 MINOR CHANGES

1. The procedure to be followed in obtaining approval of minor changes to airplanes manufactured under the terms of a type certificate will largely depend on the nature of the change involved. As soon as time will permit additions will be made to this manual covering certain specific changes in addition to that covered in 2 below.

2. When a tail wheel and tire are appended to a previously approved tail skid installation and the original provisions for shock absorption are left intact, the following procedure should be followed in obtaining approval of the change:

a. Submit the usual file drawings.

b. Substantiate the strength of skid structure and attachment to the fuselage if the point of contact with the ground of the proposed wheel installation is forward of the tail skid shoe contact point. For installations where the contact points coincide or the wheel is to the rear of the skid contact point, no structural investigation is required unless such procedure appears necessary.

c. Obtain inspection of installation and weight check by a representative of the Administrator.

d. Obtain recheck of landing and taxiing characteristics by a representative of the Administrator. No investigation of the status of the tire, strength of the wheel attachment to the skid, or the energy absorption capacity need be made.

.061 MAJOR CHANGES

A GENERAL

1. Major changes in existing designs will usually entail an appreciable expenditure of time and money on the part of the applicant for approval. Care should therefore be taken to determine the status of such changes with respect to the pertinent regulations, prior to any extensive rebuilding or conversion.

B INSTALLATION OF AN ENGINE OF A TYPE OTHER THAN THAT COVERED BY THE ORIGINAL TYPE (OR APPROVED TYPE) CERTIFICATE

1. It is generally understood that the purpose of most changes involving the installation of an engine of a type other than that covered by the original approval is to permit full advantage to be taken of improvements in engine performance which do not involve a material increase in engine weight. This is of direct benefit to the operator of the airplane, as it increases safety of operation and/or performance by improving take-off, climb, single-engine performance, true cruising speeds at altitude, engine reliability, and engine life between overhauls, with few (if any) changes in the aircraft structure. It should be carefully noted that these benefits will be difficult to obtain if the changes made require or involve an increase in the originally approved airplane gross weight or placard speeds. If the changes result in an increase in placard speeds, it will be necessary in any event to reinvestigate the structure for compliance with the flutter prevention measures referred to in CAR 04.404. Before making a change in engine it is always advisable for an owner to contact the manufacturer of the make of airplane involved to learn if the proposed change has ever been approved by the Administrator. If there is a record of approval, it is often a relatively simple matter to revise the airplane to conform with the manufacturer's approved data.

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2. The general procedure to be followed, when the rated power of the engine to be installed exceeds that originally used for design purposes or exceeds the rated power of the engine being replaced, is described in the following paragraphs. It consists, briefly, in substantiating the strength of the engine mount and adjacent structure for the take-off (one minute) power and for the local increase in weight, if any, and in limiting the engine output and indicated speeds for subsequent posting in the aircraft. The engine placard limits differentiate between the power permitted for continuous operation (maximum, except take-off), and that which has been approved for take-off only (take-off, one minute). The following procedure applies to modifications of existing designs but the principles will also apply to new designs under consideration.

3. To expedite handling and to reduce the usual exchange of correspondence to a minimum, the applicant for approval of the change should always supply a complete description of the proposed engine replacement. When an individual airplane is being modified it should be identified in the correspondence as to name of manufacturer, model designation, manufacturer's serial number and identification mark. In addition, a new or revised airplane model designation should be selected to distinguish it from the original model. The current status of the engine to be used, with respect to CAR 13, should be determined prior to the completion of any extensive changes. Field inspection personnel of the Administrator are supplied with this information and they will assist in the determination of the status of the engine in question. Copies of the approved engine specification can be obtained from the Administration's Publications and Statistics Division in Washington. If the details of the powerplant installation are affected, note that the pertinent requirements specified in CAM 04.0320-1(e) and CAR 04.6 call for certain approved file data.

4. The data submitted should include a comparison of the weights of the original and proposed engine installations. Appendix 1 of the Repair and Alteration Manual will be found useful in rechecking the balance. The aircraft specification, copies of which can be obtained from the Administration's Publications and Statistics Division, includes the approved center of gravity range.

5. Changes in engine mount structure and the local effects of an increase in engine weight must, of course, be investigated. The extent of such investigation will depend largely upon the amount of increased power

the applicant desires to use in take-off (one minute) and the remaining operations. See 7 below for references to operation limitations. See CAM 04.0320 for references to the information required on drawings submitted covering the changes made.

6. Airspeed Placard Limits. There are a large number of certificated airplanes in service which do not display the placard speeds specified in the current requirements. These airplane models were approved prior to the application of the 1934 edition of Aeronautics Bulletin No. 7-A in which the requirements for airspeed placards first appeared in the airplane regulations. In these cases when the rated power of the engine being installed exceeds that of the engine installation originally approved, the following airspeed limits should be displayed:

a. Level Flight or Climb: V_L.
b. Glide or Dive: $1.2V_L$. V_L is the actual indicated high speed in level flight obtainable with the power of the engine originally used.

If the applicant for approval wishes to raise these placard limits, there are no objections to his investigation of the case. The current requirements will serve as a guide for determining which components of the airplane and pertinent loading conditions or design criteria involve a consideration of design airspeeds. For cases in which airspeed placard limits were determined as part of the original approval, the use of an engine with rated power in excess of that originally used for design purposes will not require changes of the original airspeed placard limits. However, as previously mentioned, an attempt to increase these placard speeds will represent a revision of the basic structural design data and as such will usually require an appreciable amount of reinvestigation for purposes of determining whether the airplane structure can withstand the air loads incident to the increased performance. As a rule only the airplane manufacturer or an experienced engineer can efficiently make the necessary investigations. The Administrator does not initiate such studies.

7. Engine Placard Limits. The airplanes discussed in the first part of 6 above in most instances do not display the engine placard limits specified in the current requirements. In these cases when the rated power of the engine being installed exceeds that of the engine installation being replaced the following engine operation limits should be displayed:

a. Maximum, except take-off horsepower, not to exceed the output of the originally approved engine installation which is being replaced.
b. Take-off (one minute) horsepower, limited by:

(1) Approved take-off rating of engine. See CAR 04.60, CAR 13 and approved engine specification.
(2) Status of propeller used. See CAR 04.61, CAR 14 and approved propeller specification.
(3) Strength of engine mount structure. See CAR 04.26.
(4) Fuel flow capacity. See CAR 04.625.
(5) Engine cooling requirements. See CAR 04.640.

For cases in which engine placard limits were determined as part of the original approval of the airplane, the use of an engine with rated power different from that of the engine being replaced will require the display of new placard limits corresponding with the maximum permissible output determined by the following:

a'. Maximum, except take-off horsepower, limited by:
(1) Approved rating of engine. See CAR 04.60, CAR 13 and approved engine specification.
(2) Status of propeller used. See CAR 04.61, CAR 14 and approved propeller specification.
(3) Strength of engine mount structure. See CAR 04.26.
(4) Fuel flow capacity tests. See CAR 04.625. (There are a few supercharged installations for which the maximum, except take-off, rating is greater than the take-off rating. Therefore, the maximum, except take-off power, is used in determining the fuel flow required.)
(5) Full power longitudinal stability characteristics with rearmost center of gravity.
(6) Engine cooling tests. See CAR 04.640.
(7) Design power used in original analysis.

b'. Take-off (one minute) horsepower, limited by items listed in b(1) to b(5) above.

8. Inspection and Flight Tests. Following receipt and approval by the Administrator of file data satisfactorily accounting for the change in engine as discussed in the foregoing paragraphs, the usual inspection and a recheck of certain flight tests will be authorized. The extent of the flight tests will depend upon the nature of the replacement with respect to the original approval.

9. It will be of interest to designers to note that provision for future increases in engine power and airplane performance can easily be made in the original design by the following methods:

a. Assume a power loading of 12 pounds per HP in determining the maneuvering load factors. (See Fig. 04-3 of CAR 04.)
b. Design the engine mount, adjacent structure, and powerplant installation for the maximum power which might possibly be used in the future.
c. Assume a design level speed (V_L) considered high enough for all future operations. In this connection it should be noted that speed placards refer to "indicated" airspeeds and that the corresponding actual airspeed may therefore exceed the placard speed at altitudes above sea level.

C CONVERSION OF APPROVED TYPE LANDPLANE OR SEAPLANE TO APPROVED SKIPLANE STATUS

1. There are two distinct steps involved in obtaining the Administrator's approval of an airplane equipped with skis. These are as follows:

a. Approval of the ski model.
b. Approval of the airplane equipped with approved skis.

It should be noted that the approval of a ski and the approval of a ski installation are two separate cases. The Administrator's approval of a ski for a specified static load for quantity production under a type certificate does not imply approval of the ski installed on any certificated airplane. It means only that the ski itself is satisfactory. This is true also in the case of a single set of skis where no type certificate is involved.

2. Approval of the Ski Model. The strength of all skis must be substantiated in accordance with the requirements contained in CAR 15 (see also CAM 15) before they may be used on certificated aircraft, whether or not the designer or manufacturer desires to obtain a type certificate for the skis. The procedure for obtaining an approval for skis is explained in CAR 15.

3. Approval of an Airplane Equipped with Approved Skis. Certain airplane models are already approved with certain specific approved skis installed. The owner of a certificated airplane of some such model wishing to install skis, need only install skis of the model with which airplanes of his model are approved and his airplane will be approved with the skis installed, upon the satisfactory completion of an inspection of the installation by a representative of the Administrator. Should changes in the landing gear be necessary to accommodate the skis, the owner, of course, must make the changes in accordance with the change data approved by the Administrator. If

the airplane is of a model which has not been approved with the installation of skis of the particular approved model it is desired to install, the procedure hereinafter outlined should be followed:

a. Technical data showing any changes in the landing gear should be submitted to the Administrator for approval. This is not often necessary, as skis are usually designed to attach to the axles in place of the wheels.
b. Upon approval of the change data, if any, the installation must pass a satisfactory inspection by a representative of the Administrator.
c. During this inspection, the representative will obtain the weight of the ski installation and the weight of the wheel installation which has been replaced.
d. Upon completion of a satisfactory inspection, the representative will witness take-offs and landings, and other demonstrations if deemed necessary, of the airplane equipped with skis. The characteristics of the airplane equipped with skis must be acceptable to the Administrator's representative.

4. If the airplane inspected and tested is a standard airplane of a certain model and the skis installed are approved under a type certificate and manufactured under a production certificate or if the skis are manufactured under an approved type certificate, all airplanes of this model will be considered eligible for approval when equipped with skis of the model installed on the airplane inspected. The aircraft specification will identify the approval accordingly.

5. If the skis installed are not approved under an approved type certificate or were not manufactured under a production certificate, each airplane so equipped must undergo the tests of 3d above in order to be eligible for approval. The notes on the pertinent aircraft specification will list this distinction.

.062 CHANGES REQUIRED BY THE ADMINISTRATOR

1. Due to Revised Regulations. The type certificate permits production of aircraft under the terms of the airworthiness requirements in effect at the time of the type approval. Due to progress in the art, however, it may be advisable in rare cases, to require that airplanes being built under a type certificate be made to conform with a requirement made effective subsequent to the issuance of the type certificate.

2. Due to Unsatisfactory Service Experience. When unsatisfactory experiences are encountered in service it is the normal procedure for the manufacturer to prepare a service bulletin and forward it direct to the aircraft owners. Such service bulletins are usually prepared in cooperation with the Administrator. When the difficulty encountered is of sufficient importance to require immediate action an Airworthiness Maintenance Bulletin is prepared by the Administrator and is sent by registered mail to all owners to advise them of the nature of the difficulty, the corrective steps to be taken, and requesting them to contact an authorized representative of the Administrator regarding approval of the changes made. In addition a special note is generally issued as a supplement to the Aircraft Specification as a final check to insure that the particular item has been corrected by the time of the annual inspection.

.1 DEFINITIONS (INCLUDING STANDARD SYMBOLS, VALUES, AND FORMULAS)

A DEFINITIONS ADDITIONAL TO THOSE GIVEN IN CAR 04.1.

1. **Aerodynamic Center, a.c.** 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 per cent 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 CAM 04.129C.

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

3. Equivalent Drag Area, S_D. 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: $S_D = 1.28\ S_E$, where S_E is the equivalent flat plate area).

   a. S_{D_t} = estimated total drag area at high speed, in square feet. When the value of V_L is known or has been estimated, S_{D_t} can be determined by solving Eq. 16 in CAM 04. 1-C for d. When it is desired to estimate S_{D_t} first in order to compute the value of V_L, the equation $S_{D_t} = S_{D_f} + C_D S_W$ can be used. S_W refers to the total wing area exclusive of the area replaced by the fuselage and C_D can usually be assumed to be the minimum wing drag coefficient. Typical values of S_{D_f} (Drag area of airplane less wing) are given in Fig. 6.

4. **Margin of Safety, M.S.** The margin of safety is the percentage or fraction by which ultimate strength of a member exceeds its ultimate load.

   a. A linear margin of safety is one which varies linearly with the ultimate load.
   b. A nonlinear margin of safety is one which is based on stresses which are not proportional to the ultimate load. A nonlinear margin of safety is not a true measure of the excess strength of a member.

B STANDARD SYMBOLS

A -

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

a.c. - aerodynamic center.

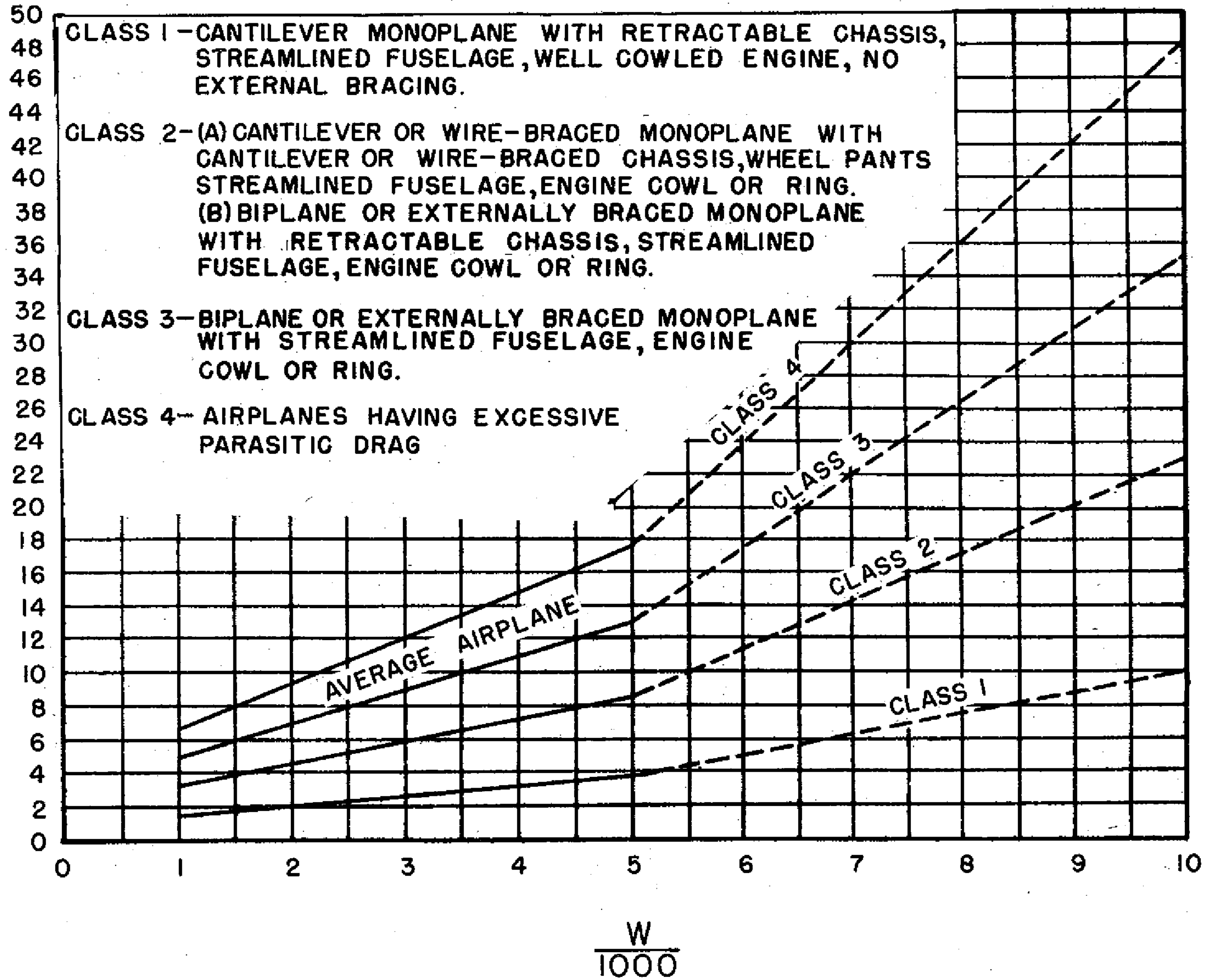


FIG. 6 VARIATION OF FUSELAGE DRAG AREA WITH GROSS WEIGHT
(REF. CAM 04.1-A3)

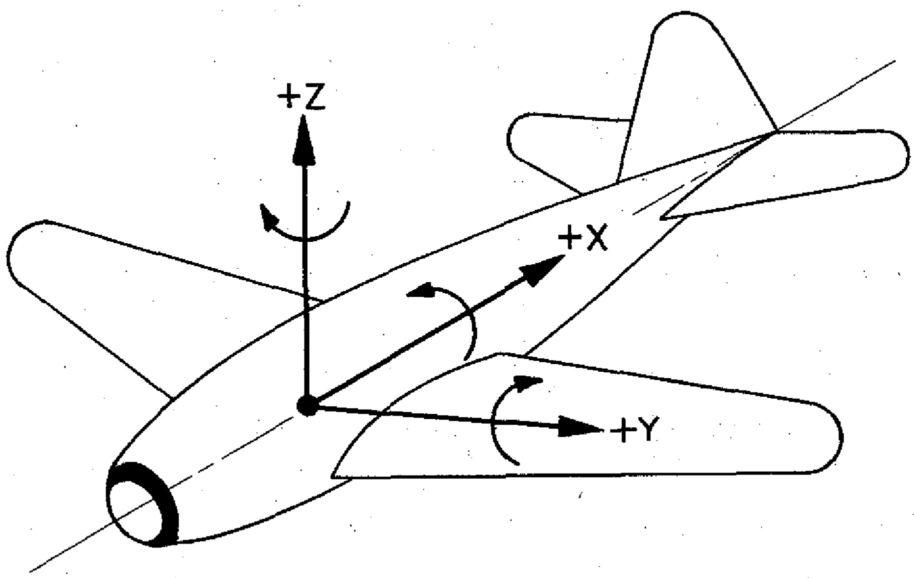


FIG. 18 – CONVENTION OF AXES, FORCES, & MOMENTS.
(REF C.A.M. 07.217–D3)

NOTE: THERE ARE NO FIGURES 19 & 20 IN THIS EDITION, AS THEY WERE DELETED BY A REVISION TO THE PRECEDING EDITION

(6) The data for column 6 is taken from the assumed span distribution curve, Figure 16 or 17. 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.
(7) Column 7 contains the product of column 5 and 6. The sum of this column when divided by the sum of column 5 will give the value K_b, which is the ratio of the mean effective C_N to the value of C_{N_o} (at the root).
(8) Column 8 contains the product of columns 2 and 7. The sum of this column when divided by the sum of column 7 gives the lateral coordinate of the centroid of the semi-wing as indicated below the table.
(9) In column 9 is listed the distance of each section aerodynamic center aft of the YZ plane.
(10) Column 10 contains the product of column 7 and 9. The sum of this column when divided by the sum of column 7 gives the X-coordinate of the centroid as indicated below the table.
(11) In column 11 is listed the distance of each section aerodynamic center above the XY plane.
(12) Column 12 contains the product of columns 7 and 11. The sum of this column when divided by the sum of column 7 gives the Z-coordinate of the centroid as indicated below the table.
(13) Column 13 contains the product of columns 4 and 5. The sum of this column, when divided by the sum of column 5, gives the length of the mean aerodynamic chord. If a location is wanted for the mean aerodynamic chord, it should be drawn on the wing so that its aerodynamic center coincides with the centroid of lift determined by means of the preceding columns.
(14) Column 14 is a list of the section moment coefficients.
(15) Column 15 contains the product of columns 13 and 14. For wings that have no twist, the sum of this column when divided by column 13 gives the average moment coefficient for the wing.

3. In the case of twisted wings a different span distribution exists for each angle of attack. The location of the resultant forces can, however, be determined as in 2 above for any known span distribution.

Table II

Determination of Point of Application of Resultant Airloads on a Wing
(Ref. CAM 04-217-D)

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
Strip No.	y	Δy	C	$C\Delta y$	R_b	$R_b C\Delta y$	$yR_b C\Delta y$	x_{ac}	$x_{ac}R_b C\Delta y$	Z_{ac}	$Z_{ac}R_b C\Delta y$	$C^2\Delta y$	$C_{M_{ac}}$	$C_{M_{ac}}C^2\Delta y$
Ref.				(3) x (4)		(6) x (5)	(2) x (7)		(9) x (7)		(11) x (7)	(4) x (5)		(14) x (13)
Σ														

$\bar{y} = \dfrac{\Sigma(8)}{(7)} =$

$\bar{x} = \dfrac{\Sigma(10)}{\Sigma(7)} =$

$\bar{z} = \dfrac{\Sigma(12)}{\Sigma(7)} =$

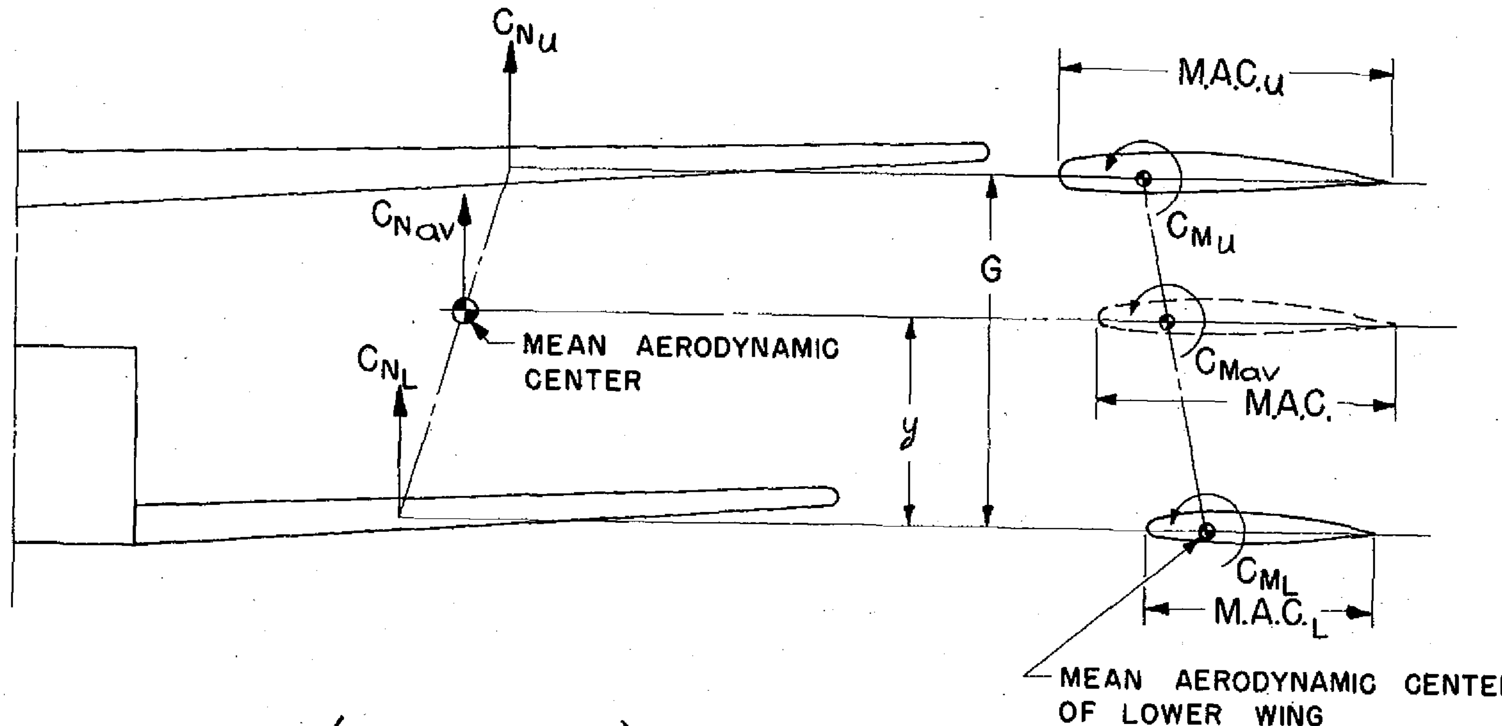
$\bar{C} = \dfrac{\Sigma(13)}{\Sigma(5)} =$

$C_M = \dfrac{\Sigma(15)}{\Sigma(13)} =$

$K_b = \dfrac{\Sigma(7)}{\Sigma(5)} =$

Check:

Wing Area $= \dfrac{2}{144}\Sigma(5) =$



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

(b) $$M.A.C. = \frac{(M.A.C.)_u S_u + (M.A.C.)_L S_L}{S_u + S_L}$$

(c) $$C_{M_{av}} = \frac{C_{M_u} S_u (M.A.C.)_u + C_{M_L} S_L (M.A.C.)_L}{S_u (M.A.C.)_u + S_L (M.A.C.)_L}$$

FIG.21 RESULTANT FORCES ON A BIPLANE

(REF. CAM 04.217-E)

E RESULTANT FORCES ON BIPLANES.

1. The mean aerodynamic center location and the value of the mean aerodynamic chord for each wing panel can be found as outlined in CAM 04.217-D. When wing flaps or other auxiliary high lift devices 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 Table II. 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 Fig. 21:

a. 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), Fig. 21.
b. 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), Fig. 21.
c. 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), Fig. 21.

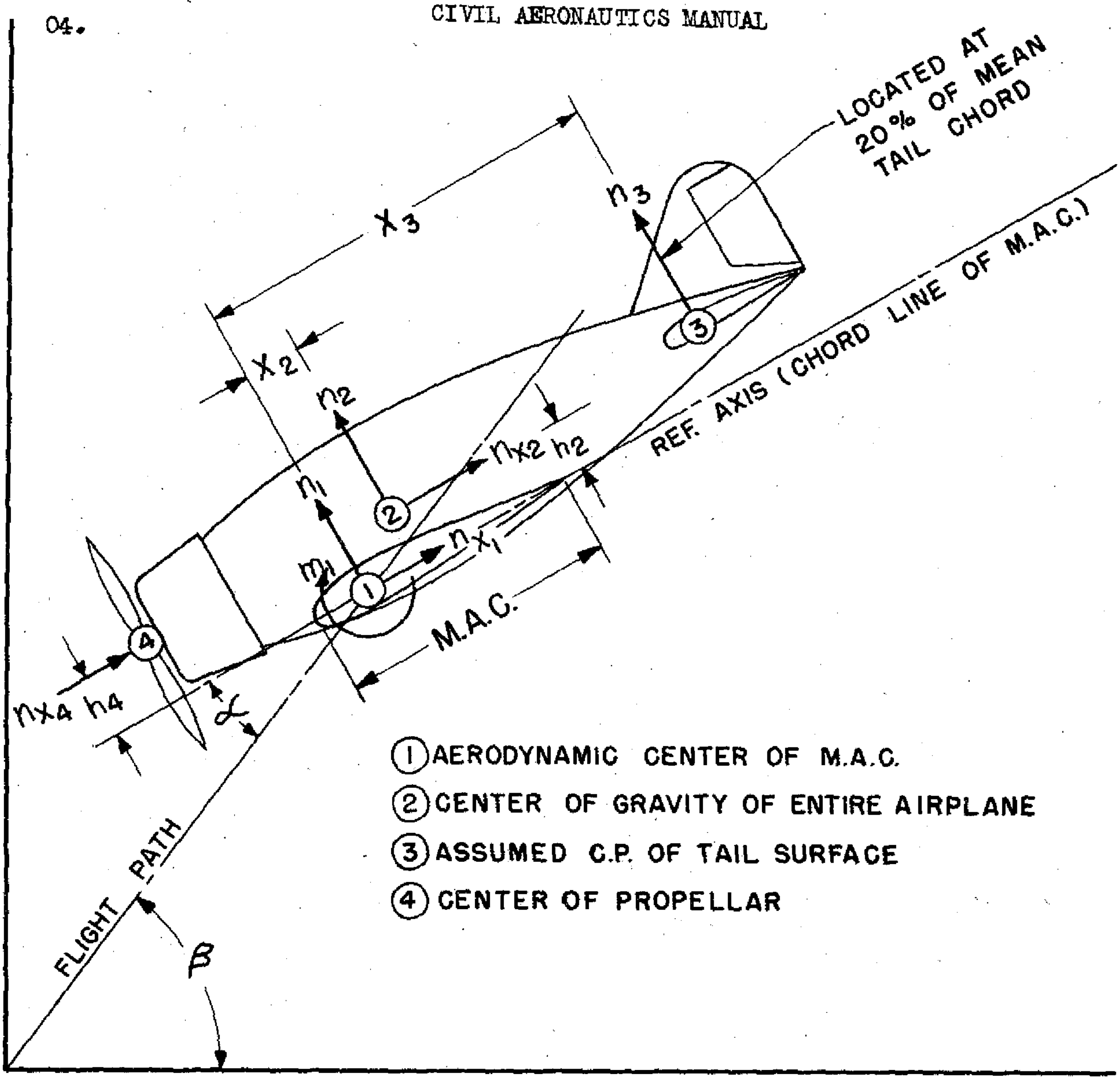
2. The mean aerodynamic center of a biplane, as determined above, is based on the relative values of the normal forces acting on each wing. When the average normal force coefficient for the entire biplane is near zero, the relative loading on the wings varies over a wide range and the mean aerodynamic 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:

a. A single location may be assumed for the mean aerodynamic center for all the balancing conditions.
b. When the investigation of two different span distributions is required, the more nearly constant span distribution may be used in determining the mean aerodynamic center and MAC.
c. The computations may be made for an average value of $C_N = 0.5$, unless the biplane has an unusual amount of stagger or decalage, or is otherwise unconventional.
d. 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.

.218* BALANCING LOADS.

A GENERAL.

* It will be noted that there is no .218 section in CAR 04. The subject of balancing loads has, however, been assigned this number in order to provide better continuity within the Manual.



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.

FIG.22 BASIC FORCES IN FLIGHT CONDITIONS

(REF. CAM 04.218-B)

1. 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 if 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 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 exactly opposite direction and with a force equal to "n" times its weight.

2. 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 flight conditions. The usual expedient in the case of the symmetrical flight 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 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 balancing tail load may consist entirely of a balancing air load from the tail surfaces.

B BALANCING THE AIRPLANE.

1. The following considerations are involved in balancing the airplane:

   a. 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.
   b. It is assumed that the limit load factors specified for the basic flight conditions are wing load factors. A solution is therefore made for the net load factor acting on the whole airplane. The value so determined can then be used in connection with each item of weight (or with each group of items) in analyzing the fuselage. For balancing purposes the net factor is assumed to act at the center of gravity of the airplane.
   c. 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 20 per cent of the mean chord of the entire tail surface. This arbitrary location may also be considered as the point of application of inertia forces resulting from angular acceleration, thus simplifying the balancing process.
   d. In Fig. 22 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,

TABLE III

BALANCING COMPUTATIONS

(See Fig. 22 for symbols)
(Ref. CAM 04.218-B3)

			V_L = mph		V_g = mph	
	No.	Item	I	II	III	IV
See CAM 04.218-B1 through B5	(1)	W = gross weight, pounds				
	(2)	$q = .00256\ V^2$				
	(3)	s = 1 / S				
	(4)	q/s = 2 / 3				
	(5)	n_1 = applied wing load factor				
	(6)	C_N = 5 / 4				
	(7)	C_L corresponding to 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				
See CAM 04.218-B6	(17)	C_{mt} = moment coefficient of airplane less tail				
	(18)	ΔC_m = 17 - 11				
	(19)	Δm_1 = 18 x 4				
	(20)	Δn_3 = 19 / $(x_3 - x_2)$				
	(21)	ΔT = 1 x 20				
	(22)	T' = 16 + 21				

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 Fig. 22 to include the additional

2. As shown in Fig. 22, a convenient reference axis is the basic chord line of the mean aerodynamic wing chord. (The basic chord line is usually specified along with the dimensions of the airfoil section.) The determination of the size and location of the MAC is outlined in CAM 04.217-D. In determing the vertical location of the aerodynamic center of the MAC (point 1 of Fig. 22) the vertical position of the MAC in relation to the wing root chord, or other similar reference line, should be considered.

3. A tabular form will simplify the computations required to obtain the balancing loads for various flight conditions. A typical form for this purpose is shown in Table III. In using Fig. 22 and Table III the following assumptions and conventions should be employed:

a. <u>If known distances or forces are opposite in direction from those shown in Fig. 22, 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_{x_4} 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 conditions of positive acceleration the solution should give a negative value for n_2, as the inertia load will be acting downward. The convention for m_1 corresponds to that used for moment coefficients; that is, when the value of C_M is negative m_1 should also be negative, indicating a diving moment.

b. <u>All distances should be divided by the MAC</u> before being used in the computations.

c. The propeller thrust should be assumed to act along the thrust axis.

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

4. Computation of Balancing Loads. In Table III 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 design 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. 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 CG will require a corresponding change in the values of x_2 and h_2 on Fig. 22.

a. When the full-load center of gravity position is variable the airplane should be balanced for both extreme positions unless it is apparent that only one is critical. In certain cases it may also be necessary to check the balancing tail loads required for the loading conditions which produce the most forward and most rearward center of gravity positions for which approval is desired.

5. The following explanatory notes refer by number to items appearing in Table III;

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

(5) n_1 = limit load factor required for the condition being investigated. (See CAR 04.21).

(8) Determine C_c as specified in CAR 04.21. See also eq. 8, CAM 04.1-C.

(10) Propeller thrust, F_{pr}, should be determined from Eq. 15, CAM 04. 1-C for conditions at V_L. For conditions at V_g assume $n_{x_4} = 0$.

(11) The value of C'_m is specified in CAR 04.21. For a biplane see CAM 04.217-E of this bulletin. See also CAM 04.217-D in cases involving wing flaps.

(13) The net tail load factor, n_3, is found by a summation of moments about point (2) of Fig. 22, 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 explanatory notes apply only when the set-up shown in Fig. 22 is used. If a different distribution of external loads or a different system of measuring distances is employed, the computations should be correspondingly modified.

6. The preceding paragraphs 1-5 and items 1-16 in Table III cover the determination of the balancing loads, without consideration for the moment which may be contributed by the fuselage and nacelles. The following explanatory notes refer by number to items appearing in Table III which provide for the determination of tail loads with consideration for fuselage moment effects, as required by CAR 04.2210.

(17) C_{mt} is the total moment coefficient about the c.g. of the airplane less tail, as determined from a wind tunnel test. When such test results are not available this item can be omitted, as other provisions to cover cases of this type are given in item (18) following. It will be noted that this coefficient is based on the design wing area and the mean aerodynamic chord.

(18) ΔC_m is the increment in moment coefficient due to the fuselage and nacelle moments, also based on design wing area and mean aerodynamic chord. When data on item (17) is not available, ΔC_m can be assumed equal to -0.01.

(22) T' is the tail load considering fuselage and nacelle moment effects.

.22 CONTROL SURFACE LOADS.

.220 GENERAL.

1. The requirements for the design of control surfaces as outlined in CAR 04.22 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.

2. The average unit loading normal to any surface is determined by the force coefficient C_N and the dynamic pressure q, as shown by Eq. 13, CAM 04. 1-C. 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.

3. 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 CAR Figs. 04-4, 04-5 and 04-6, it will be simpler to assume that the unit loading at the hinge line is constant over the span

4. Although there are no specific chord loading conditions for control surfaces specified in CAR 04.22, such surfaces should be designed to withstand a reasonable amount of chord load in either direction. A total chord load equal to 20 percent of the maximum normal load may be used as a separate design condition. The distribution along the span may be made proportional to the chord, if desired. Tests for this condition are not required unless the structure is such as to indicate the advisability of such tests.

.2210 BALANCING (HORIZONTAL SURFACES).

1. The balancing loads should be applied to the horizontal tail surfaces, as the ailerons and the 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 CAR 04.2220.

2. An acceptable method for accounting for fuselage and nacelle moments in the determination of the balancing tail loads is given in CAM 04.218-B6 and CAM 04 Table III. When wind tunnel tests have been used in this process the tail loads T' in item 22 of this table may be used for design purposes. When, however, the -0.01 moment increment has been used in lieu of wind tunnel tests to account for fuselage and nacelle moments, the balancing loads to be used for design purposes should be taken as either item 16 or item 22 in CAM 04 Table III, whichever are most severe. This is to allow for a possible range of fuselage and nacelle moment coefficients.

3. The chord distribution illustrated in CAR Fig. 04-4 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 the pilot might need to exert to hold the airplane in equilibrium.

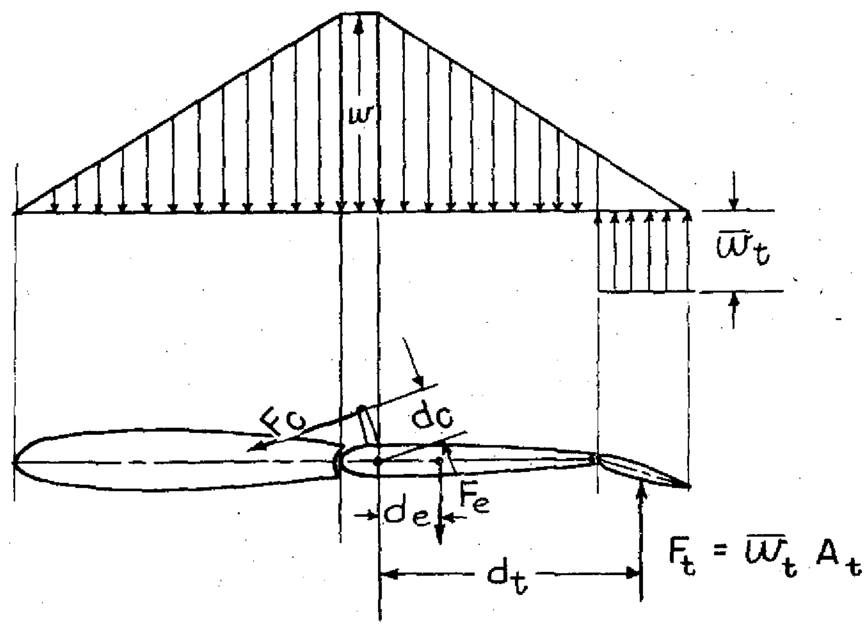
4. In CAR Fig. 04-4, the load from the elevator is shown as a concentrated load acting at the elevator hinge line. The hinge moment is, of course, resisted by the control system and therefore does not affect the stabilizer. It will be noted in CAR 04.2210(b) that the opposite elevator load, P, may be assumed equal to zero when the balancing load is obtained with flaps deflected (Conditions VII and VIII). This is based on the improbability of the pilots having to push on the elevator control in order to obtain balance with flaps down.

.2211 MANEUVERING (HORIZONTAL SURFACES)

1. The requirements for maneuvering loads outlined in CAR 04 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 effebts of increasingly greater airplane speeds. It should be understood that the method is designed for application to 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.

2. The design values of C_N specified in CAR 04 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.

3. The chord distribution shown in CAR Fig. 04-5 represents 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.



$$F_e = \frac{F_t\, d_t + F_C\, d_C}{d_e}$$

F_C = CONTROL SYSTEM FORCE

F_t = TOTAL TAB LOAD

F_e = TOTAL ELEVATOR LOAD

FIG. 23 TAB LOADING CONDITION

(REF. CAM 04.2213)

.2212 DAMPING (HORIZONTAL SURFACES)

1. 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 CAR 04 as a supplementary condition based on the initial maneuvering condition. 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.

.2213 TAB EFFECTS (HORIZONTAL SURFACES).

1. The loading condition specified in CAR 04.2213 is diagrammatically illustrated in Fig. 23. This condition represents the case of the tab load and the control force both acting so as to resist the hinge moment due to 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, can be neglected.

.2220 MANEUVERING (VERTICAL SURFACES).

1. The comments in CAM 04.2211 in regard to horizontal surfaces also apply, in general, to the vertical surfaces.

2. It is specified that the value of V_p shall not be less than the level flight speed with one engine dead. This 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. In estimating the speed with one engine dead the following approximate equation may be used:

$$V_p = 0.9\, V_L \cdot \left[\frac{n-1}{n}\right]^{1/3}$$

Where V_p = speed with one engine dead.

V_L = normal high speed.

n = total number of engines.

.2221 DAMPING (VERTICAL SURFACES).

1. The comments of CAM 04.2212 in regard to horizontal surfaces also apply, in general, to the vertical surfaces.

.2222 GUSTS (VERTICAL SURFACES)

1. The following points should be noted in connection with this requirement:

a. This gust condition applies only to that portion of the vertical surface which has a well defined leading edge. The total effective area for this condition is therefore the sum of the fin and rudder areas which lie behind such leading edge. In cases where the fin fairs gradually into the fuselage the leading edge is considered to be well defined for those longitudinal sections through the fin and rudder which have thickness-chord ratios of .20 or less. For the purposes of this requirement the "fin" is considered to include any rudder balance area ahead of the extended trailing edge of the fin.

b. The chord distribution specified in CAR Fig. 04-6 is applicable to those cases in which the mean chords of the effective fin and rudder areas are of approximately the same magnitudes. When this figure is used it should be noted that $\overline{w}$ refers to the average limit pressure over the total effective area of the vertical surface. The total load acting is therefore equal to $\overline{w}$ times the total effective area. This load is, however, applied to the fin only, in accordance with the specified distribution.

c. When the mean chords of the effective fin and rudder areas are of considerably different magnitude, the chord distribution for a symmetrical airfoil should be used. This distribution can be obtained from the curve marked "experimental mean" of Fig. 11, NACA Technical Report No. 353.

.224 WING FLAPS.

1. In the design of wing flaps, the critical loading is usually obtained when the flap is completely extended. The requirements outlined in CAR 04 apply only when the flaps are not used at speeds above a certain predetermined design speed. As noted in CAR 04.743, a placard is required to inform the pilot of the speed which should not be exceeded with flaps extended. Reference should be made to current NACA Reports and Notes for acceptable flap data.

.230 GENERAL.

1. In all cases the limit loads for control systems are specified as 125 per cent of the actual loads corresponding to the control surface limit loading, with certain maximum and minimum control force limits. The factor of 1.25 is used to account for various features, such as:

a. Differences between the actual and the assumed control surface load distribution.

b. Desirability of extra strength in the control system to reduce deflections.

c. Reduction in strength due to wear, play in joints, etc.

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2. The maximum control force limits are based on the greatest probable 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 ultimate factor of safety of 1.5, which is required in any case, will permit the maximum limit load to be exceeded for a relatively short time without serious consequences.

3. The minimum control force limits apply only to cases in which the control surface limit loads are relatively small. The minimum control forces may be applied when the control surfaces are completely utilized and are against the stops.

4. The requirement of the multiplying 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.

.234 FLAP AND TAB CONTROL SYSTEMS.

1. It should be noted that 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.

2. The following design conditions apply to crank and twist type controls for airplanes certificated in the "Transport" category:

a. From the cockpit control to the control system stops, tab control systems should be designed to withstand the following limit loads:
   (1) A torque of 133 inch-pounds applied to the control knob in the case of twist controls.
   (2) A torque given by the relation T = 100 R applied to the control wheel or crank in the case of controls that are not operated by a twisting motion. In this category will fall cranks, levers, and handwheels with a well-defined rim which can be grasped for turning.

b. From the control system stops to the tab, tab control systems should be designed to withstand limit loads corresponding to 125 percent of the limit load used for the design of the tab. In the case of multiple tabs or multiple surfaces each incorporating a tab, 125 percent of the limit load should be applied to all tabs simultaneously.

c. Care should be taken to secure a rugged connection between the tab and the irreversible unit.

240 GENERAL.

1. Tail Wheel Type Gear. The basic landing conditions outlined in CAR 04.24 for conventional land type gear are tabulated in Fig. 24. This chart can be used as a summary of the load factors for the landing conditions by inserting the actual values used.

2. Nose Wheel Type Gear. The following design conditions have been found acceptable in certain cases of nose wheel gear. It is emphasized, however, that all unusual features of a particular design should be investigated to insure that all possible critical loadings have been considered. See also CAM 04.340 for a discussion of energy absorption tests.

a. Three-Wheel Landing with Vertical Reactions. The minimum limit load is specified in CAR 04 Fig. 04-10. The value of the sum of the static ground reactions shall be the weight of the airplane less landing gear. The total load shall be divided between the front and rear gear in inverse proportion to the distances, measured parallel to the ground line, from the CG of the airplane less landing gear to the points of contact with the ground. The load on the rear gear shall be divided equally between wheels. Loads shall be assumed to be perpendicular to the ground line in the three-wheel landing attitude, with all shock absorbing units and tires deflected to one-half their total travel unless it is apparent that a more critical arrangement could exist. The critical positions of the CG shall be investigated. The minimum ultimate factor of safety shall be 1.5.

b. Three-Wheel Landing with Inclined Reactions. The minimum limit load factor is specified in CAR 04 Fig. 04-10. The resultant of the ground reactions shall be a force lying in the plane of symmetry and passing through the CG of the airplane less landing gear. The basic value of the vertical component of the resultant force shall be equal to the weight of the airplane less landing gear. The horizontal component shall be 25 per cent of the vertical, acting aft. The total force shall be so divided between the front and rear gear that the resultant moment acting on the airplane will be zero. The load on the rear gear shall be divided equally between wheels. The shock absorbers and tires shall be deflected to the same degree as in condition a above. The critical positions of the CG shall be investigated. The minimum ultimate factor of safety shall be 1.5.

c. Two-Wheel Landing with Vertical Reactions - Nose Up, The minimum limit load factor is specified in CAR 04 Fig. 04-10. The airplane shall be assumed to be in an extreme nose-up attitude. The gross weight of the airplane less the rear gear shall be assumed to act at the rear wheels in a direction perpendicular to the ground line. The total load shall be divided equally between the two rear wheels. The resultant moment on the airplane shall be balanced by inertia forces. The shock absorbers and tires shall be deflected to the same degree as in condition a above. The minimum ultimate factor of safety shall be 1.5.

LANDPLANE LANDING CONDITIONS
FOR TAIL WHEEL TYPE GEAR (SEE CAR 04.24)

CONDITION	LEVEL	3-POINT	SIDE (1)	ONE-WHEEL LANDING (1)	BRAKED
REFERENCE CAR 04.	.241	.242	.243	.244	.245
LOAD FACTOR n (2) (Limit)	$2.80 + \frac{9000}{W + 4000}$ (3) $3.00 + 0.133(W/S)$	Same as Level	.667	.5 Level	1.33
ATTITUDE	Propeller Axis Horizontal	3-Point	3-Point (4)	Propeller Axis Horizontal	3-Point (4)
VERTICAL COMPONENT	nW (5)	nW (5)(7)	nW (6)	nW (5)	nW (5)
REARWARD COMPONENT	Resultant Thru CG (8)	Zero	.55 nW	Resultant (7)(8) Thru CG (Side View)	.55 Vertical
SIDE COMPONENT	Zero	Zero	nW (Inward)	Zero	Zero
SHOCK STRUT DEFLECTION	50% Travel (9)	50% Travel (9)	Static Position	50% Travel	50% Travel
TIRE DEFLECTION	50%	50%	25%	50%	25%

(1) Components act on one wheel only.
(2) Need not exceed 4.33.
(3) Use smaller value. See also Note (2) above.
(4) Reaction at tail equals zero.
(5) W is gross weight less wheels and chassis.
(6) W is gross weight.
(7) Distributed to wheels and skid so that moments about CG equals zero.
(8) Need not exceed 25% vertical component.
(9) Unless apparent more critical conditions exists.

FIG. 24 LANDPLANE LANDING CONDITIONS FOR TAIL WHEEL TYPE GEAR

d. Two-Wheel Landing with Inclined Reactions - Nose up. The minimum limit load factor is specified in CAR 04 Fig. 04-10. The airplane shall be assumed to be in an extreme nose-up attitude. The resultant force shall be determined in the same manner as in condition b above except that the gross weight of the airplane less the rear gear shall be used. The total load shall be divided equally between the two rear wheels. The resultant moment on the airplane shall be balanced by inertia forces. The shock absorbers and tires shall be deflected to the same degree as in condition a above. The minimum ultimate factor of safety shall be 1.5.

e. Two-Wheel Landing with Inclined Reactions - Nose down. The minimum limit load factor is specified in CAR 04 Fig. 04-10. The airplane shall be assumed to be in a nose-down attitude with the front wheel just off the ground. The resultant force shall be determined in the same manner as in condition b above except that the weight of the airplane less the rear gear shall be used. The total load shall be divided equally between the two rear wheels. The resultant moment on the airplane shall be balanced by inertia forces. The shock absorbers and tires shall be deflected to the same degree as in condition a above. The critical position of the CG shall be investigated. The minimum ultimate factor of safety shall be 1.5.

f. Two-Wheel Landing with Brakes - Nose down. The minimum limit load factor shall be 1.33. The airplane shall be assumed to be in a nose-down attitude with the front wheel just off the ground. The gross weight of the airplane less the rear gear shall be assumed to act at the rear wheels in a direction perpendicular to the ground line. In addition, a horizontal aft component equal to .55 times the vertical shall be applied at each wheel at the points of contact with the ground. The total load shall be divided equally between the two rear wheels. The resultant moment on the airplane shall be balanced by inertia forces. The tires shall be assumed to have deflected not more than one-quarter the nominal diameter of their cross-section, and the deflection of the shock absorbers shall be the same as in condition a above. The minimum ultimate factor of safety shall be 1.5.

g. Side Drift Landing. The minimum limit load factor is specified in CAR 04 Fig. 04-10. The attitude of the airplane, the vertical components of the landing gear reactions, and the deflections of the shock absorbers and tires shall be the same as in condition a above. In addition, a horizontal aft component and a side component, each equal to .25 times the vertical component, shall be applied at each wheel at the points of contact with the ground. The resultant moment on the airplane shall be balanced by inertia forces. The minimum ultimate factor of safety shall be 1.5.

h. Side Drift Landing with Brakes. The minimum limit load factor shall be 1.0. The attitude of the airplane, the static ground reactions on the front and rear gear, and the deflections of the shock absorbers and tires shall be the same as in condition a above. The total load on the rear gear shall, however, be applied entirely on one wheel. In addition, a side component equal to .75 times the vertical component shall be applied at each wheel at the points of contact with the ground. The side load at the rear wheel shall be assumed to act inward and the side load at the nose wheel shall be assumed to act in the same direction. A horizontal aft component equal to .55 times the vertical component shall be applied at the point of contact with the ground of each wheel equipped with brakes. (It should be noted that one rear wheel is not loaded). The resultant moment on the airplane shall be balanced by inertia forces. The minimum ultimate factor of safety shall be 1.5.

i. One-Wheel Landing. An investigation of the fuselage structure is required for a one-wheel landing in which only those loads obtained on one side of the fuselage in condition e above are applied. The resulting limit load factor is therefore one-half of the minimum limit load factor specified in CAR 04 Fig. 04-10. (This condition is identical with condition e above insofar as the landing gear structure is concerned). The minimum ultimate factor of safety shall be 1.5.

3. Ski Gear. As noted in CAR 04.2410 the ground loads for ski gear are the same as for wheel gear. However, the strength of skis and ski pedestals must be substantiated in accordance with the requirements of CAR 15.12. See also CAM 15.12. Approval of ski installations is covered in CAM 04.061. The Canadian ski gear requirements, which are of interest to manufacturers contemplating export to Canada, are listed in Inspection Handbook, Chapter XII.

4. Special Considerations. When lower limit and ultimate load factors are used under the provisions of CAR 04.240, adequate provision should be made to likewise hold the taxiing accelerations to lower values. Consideration should also be given to the fact that with such gear there is a tendency to make landings with a higher rate of descent than is common with gear developing higher factors. When lower factors are used in the case of rubber shock absorbers, special rulings should be obtained from the Secretary. When lower factors are used with oleo type gear the following practice has been found acceptable:

a. Such lower design load factors should never be less than one half the conventional values.

b. A margin between the design load factor and the load factor developed in the drop test should be shown. This margin should be at least 20 per cent (of the design load factor) at the one half value noted in a above, and may decrease linearly to zero as the conventional design load factors are reached.

c. The use of such lower ultimate load factors should be justified by drop tests in which the complete landing gear is used.

The provisions of a and b above can be expressed by the formulas given below. The maximum permissible developed load factor is

$$n_1 = \frac{2n_o\, n}{3n_o - n},$$

and the minimum required ultimate load factor for use in the analyses is

$$n = \frac{3n_o\, n_1}{2n_o + n_1},$$

but n should not be less than $0.5n_o$, where

n_o = ultimate load factor (Value from CAR Fig. 04-10 times 1.5),

n = minimum required ultimate load factor for use in the analysis,

n_1 = maximum permissible load factor developed in the drop test.

.2411 ENERGY ABSORPTION.

1. The definition of stalling speed V_s used in drop height calculations is given in CAR 04.113. If accurate flight test data for the airplane in question, or for a very similar airplane, are available, such data may be used as a basis for calculating the power-off stalling speed. However, the determination of speeds in the flight tests used in this connection should not involve an extensive extrapolation of the airspeed calibration. See also CAM 04.340.

.2420 ENERGY ABSORPTION.

1. See CAM 04.340 for general discussion.

.243 SIDE LOAD.

1. This condition represents a loading such as would be obtained in a ground loop.

.244 ONE WHEEL LANDING.

1. This condition represents the "whipping" condition obtained in either of the two following cases:

a. The airplane strikes 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.
b. 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.

2. This condition does not require an additional investigation of the landing gear structure as the loads are the same as in level landing.

.246 SIDE LOADS ON TAIL WHEEL OR SKID.

1. It is required that suitable assumptions shall be made to cover side loads acting on tail skids or tail wheels which are not free to swivel or which can be locked or steered by the pilot. In such cases it will be satisfactory to consider a side load acting alone and having a limit value equal to one-fourth the limit load acting on the tail skid (or wheel) in the three point landing condition (CAR 04.242). This side load should act normal to the plane of symmetry at the center of contact of the skid (or wheel) and the ground. The attitude of the airplane and the deflections of the tire and shock absorber unit should be assumed the same as in the three point landing condition. The minimum ultimate factor of safety should be 1.5.

2. It is also recommended that this side load condition be applied to swiveling tail wheel units with the modification that the wheel is assumed to be rotated 90 degrees from the plane of symmetry and the side load to be applied through the center of the axle.

.250 GENERAL.

1. The basic water landing conditions are tabulated in Fig. 25. This chart can be used as a summary of the load factors for the landing conditions by inserting the actual values used.

2. The landing conditions outlined for float seaplanes 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. When 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.

3. 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 CAR 04 Table 04-7). 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.

SEAPLANE LANDING CONDITIONS See CAR 04.25			
COMPONENT	FLOAT (1)		
CONDITION	Inclined Reaction	Vertical Reaction	Side Landing
REFERENCE CAR 04	.251	.252	.253
n (Limit)	4.20 (2)	4.33 (2)	4.0
VERTICAL REACTION	nW (3)	nW (3)	nW (3)
REARWARD REACTION	1/4 Vertical	0	0
SIDE REACTION	0	0	1/4 Vertical
RESULTANT	Through CG Less Floats and Bracing		In plane through CG and perpendicular to propeller axis
FACTOR OF SAFETY	1.85 (4) 1.50 (5)		1.50
ATTITUDE	Propeller axis or reference line horizontal		

(1) For float requirements see CAR 04.257 and CAR 15.11
(2) Need not exceed 3.00 + .133(W/S).
(3) W is gross weight less floats and bracing.
(4) For float attachments and fuselage carry-thru members.
(5) For remaining structural members.

FIG. 25 SEAPLANE LANDING CONDITIONS

3. In those cases when it would be otherwise unnecessary to calculate the moment of inertia of the airplane about its lateral axis, the specified condition may be replaced by a simplified loading by making some conservative assumptions. To simplify the computation of inertia loads, an 'average' linear acceleration factor can be used. The approximate method then consists of applying to the keel at a point one-tenth of the distance from the bow to the step a load equal to $n_b W_e = 1/2 n_s W_e$ and computing the inertia loads over the forward portion of the hull by using a load factor of $0.65n_b$ applied vertically, together with a horizontal component of $0.50n_b$. This would result in an unbalanced bending moment and shear by the rearward portion of the hull. Since this condition is not likely to be critical for the rearward portion of the structure, these unbalanced forces and moments need not be applied to the rear portion for design purposes. The 'simplified' system for the bow loading condition is illustrated by Fig. 25b.

4. The use of a horizontal component of $.50n_b$ in the 'simplified' bow loading condition insures adequate forward-acting inertia loads for local design.

5. To avoid excessive local shear loads, the water reaction may be distributed over the hull bottom. The area to be used in determining pressures (for comparison with those specified in CAR 04.2541(a)) should be the projected area of the hull bottom on a plane which is normal to the resultant water load, and which intersects the bottom of the keel at a point one-tenth of the distance from the bow to the step.

.2544 STERN LOADING CONDITION

1. To simplify the computations and to decrease the amount of work required, the area to be used with the pressures specified in CAR 04.2541(a) may be taken as the projected area of the hull bottom on the plane defined in CAM 04.2542-3.

.2545 SIDE LOADING CONDITION

1. In cases where the specified condition appears to yield unreasonable results, alternative procedures may be used, subject to acceptance by the Administrator.

2. To avoid excessive local shear loads, the water reaction may be distributed over the hull bottom. The area to be used in determining pressures (for comparison with those specified in CAR 04.2541(a)) should be the projected area of the hull bottom on the plane defined in CAM 04.2542-3. The load to be used in determining pressures should be the vertical component of the resultant load.

.2542 STEP LOADING CONDITION

1. The local and distributed pressure requirements are intended to take care of the hull bottom as such, and therefore it is not too important to correlate the maximum impact factor with bottom pressure. This is further justified by the fact that the maximum impact factor does not occur until a considerable portion of the bottom has been submerged, at which time the bottom pressures have dropped considerably below the maximum values likely to be obtained locally.

2. The step loading condition is critical for the hull in shear and bending, and also may produce maximum downward inertia loads from nacelles, etc. The calculations can be considerably simplified if the resultant load is assumed to pass vertically through the center of gravity. Although the load may act at some point other than the c.g. and may actually be inclined rearward, these refinements will have very little effect on the shears and bending moments in the hull structure and may therefore be neglected. In order to provide adequate strength against forward inertia loads coming from wings, nacelles, etc., a rearward acting load is included in the bow loading condition.

3. To avoid excessive local shear loads and bending moments near the point of water load application, the water load may be distributed over the hull bottom. The area to be used in determining pressures (for comparison with those specified in CAR 04.2541(a)) should be the projected area of the hull bottom on a horizontal plane which intersects the bottom of the keel at the front step.

.2543 BOW LOADING CONDITION

1. The most severe upward shear loads and bending moments for the forward portion of the hull structure are probably caused by an impact load near the bow. Such a loading condition is likely to be obtained in landing or in take-off from rough water. A simplified procedure to cover this condition is discussed below. More rational methods may, of course, be used, subject to acceptance by the Administrator.

2. Considering the arbitrary nature of the hull loading conditions, it seems reasonable to dispense with numerous refinements in specifying the loading condition and to apply a concentrated load at some specified point in an arbitrarily chosen direction. Therefore, the bow impact load is applied to the keel at a point one-tenth of the distance from the bow to the step, and in an upward and rearward direction at an angle of 30° from the vertical. See Figure 25a. As this loading condition will produce a combination of vertical horizontal, and angular accelerations, all items of weight will produce inertia loads accordingly.

.2540 LOCAL BOTTOM PRESSURES

1. Since it is assumed that local pressures are applied only over a limited area at any instant, it is necessary to set up some value for such an area. Any solution to this problem will necessarily be an arbitrary one, therefore, it is desirable to consider the practical side of the picture. If the designer is held strictly to a certain shape and size of loaded area, it might complicate the structural analysis problem. To avoid this, a more flexible requirement is set up to the effect that the area shall be taken such as to cause the greatest local stresses in the adjacent structure. In no case need this area exceed a value of 2.0 square feet.

2. To illustrate the above, if a conventional plate-stringer bottom is used, the area could be taken as that determined by the stringer and frame spacing. This would be critical for the plate. To check the stringer and the attachment of the stringer to the frame, and to produce a maximum local load on the frame, the pressure would be applied to one stringer over an area of the same value, assuming the resulting load to be resisted entirely at the stringer attachment.

3. There have been several failures in actual practice of various hull components due to negative pressures aft of the front step. However, since data concerning negative pressures are so limited, it is not deemed advisable to specify any design criteria for this condition.

.2541 DISTRIBUTED BOTTOM PRESSURES

1. Although the bottom plating and stringers will be designed by local pressure, major members such as cross-frames and the keel will be critical when larger areas are loaded. Water tests indicate that average pressures over relatively large areas are considerably less than the "peak" local pressures. The requirement for distributed pressure consists of applying simultaneously over the entire hull bottom one-half of the pressure values required for local pressures.

2. The distributed pressure requirements are not intended to design the shell structure of the hull (sides in shear, etc.) but apply to belt frames, keels, bulkheads, and the attachment of the cross frames to the sides of the hull structure.

3. See CAM 04.2540-3.

SEAPLANE LANDING CONDITIONS See CAR 04.25			
COMPONENT	FLOAT (1)		
CONDITION	Inclined Reaction	Vertical Reaction	Side Landing
REFERENCE CAR 04	.251	.252	.253
n (Limit)	4.20 (2)	4.33 (2)	4.0
VERTICAL REACTION	nW (3)	nW (3)	nW (3)
REARWARD REACTION	1/4 Vertical	0	0
SIDE REACTION	0	0	1/4 Vertical
RESULTANT	Through CG Less Floats and Bracing		In plane through CG and perpendicular to propeller axis
FACTOR OF SAFETY	1.85 (4) 1.50 (5)		1.50
ATTITUDE	Propeller axis or reference line horizontal		

(1) For float requirements see CAR 04.257 and CAR 15.11
(2) Need not exceed 3.00 + .133(W/S).
(3) W is gross weight less floats and bracing.
(4) For float attachments and fuselage carry-thru members.
(5) For remaining structural members.

FIG. 25 SEAPLANE LANDING CONDITIONS

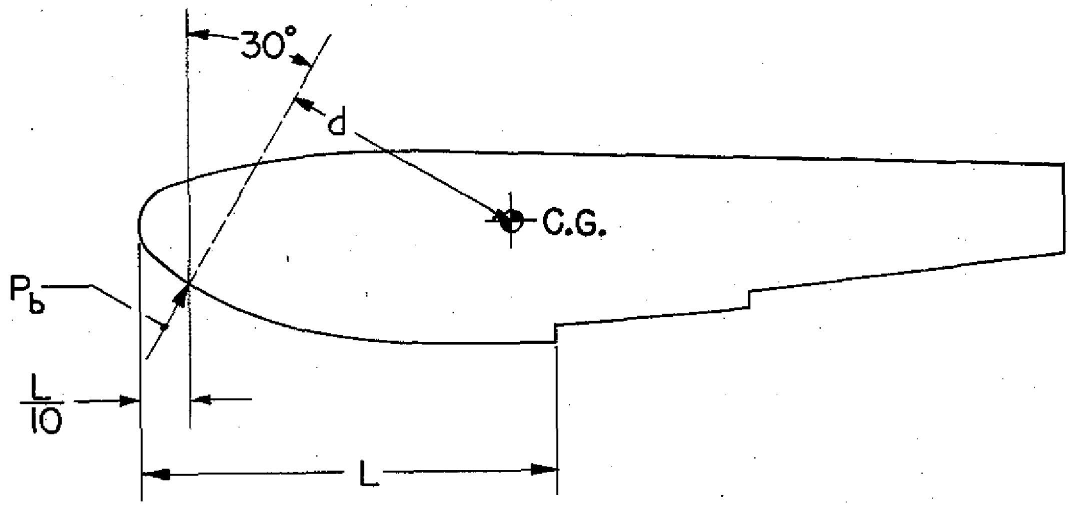

FIG. 25a APPLICATION OF BOW LOAD

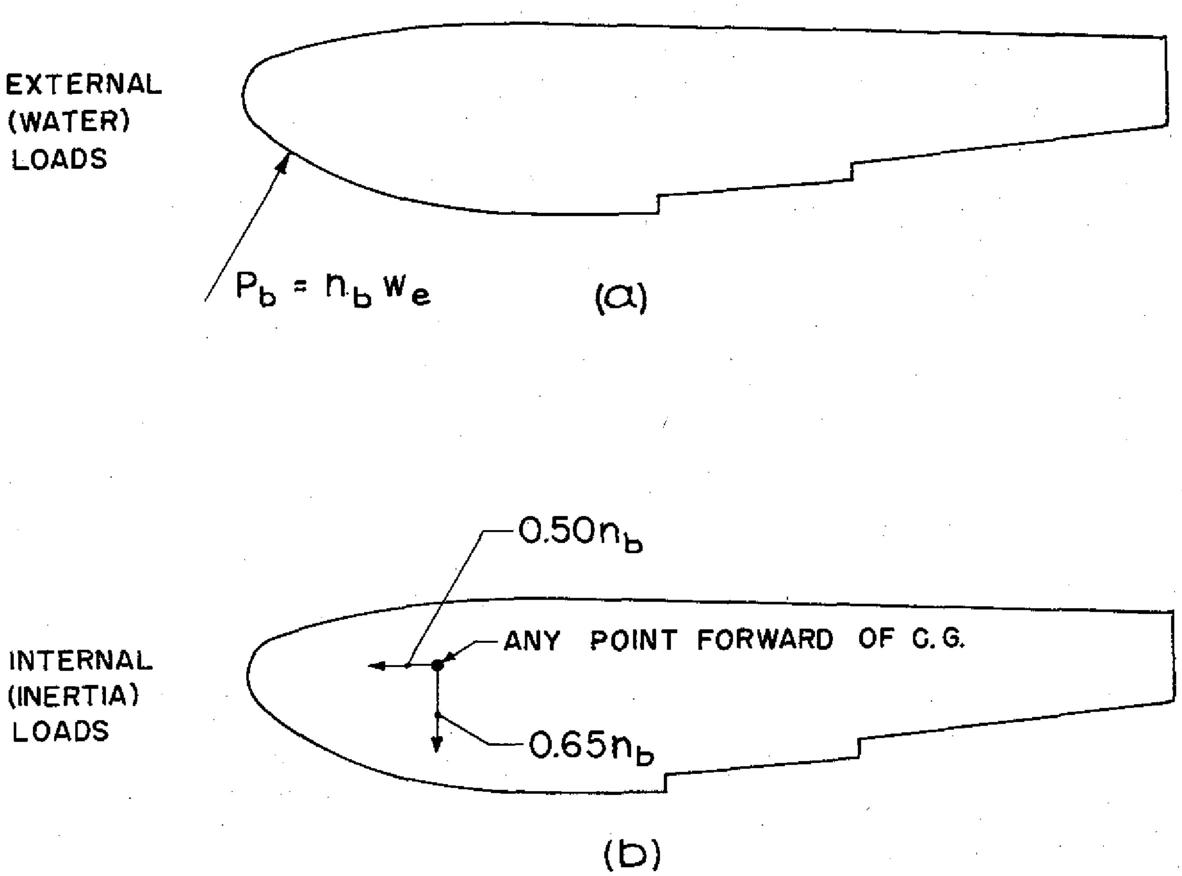

FIG. 25b SIMPLIFIED BOW LOADING CONDITION

.266 RIGGING LOADS.

1. The requirements are based on the necessity for proportioning wire sizes so as to prevent an excessive load being produced in any wire while rigging any other wire. They 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.

2. 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 of 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 may 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.

.271 FITTINGS.

1. As noted in the requirement a fitting is so defined as to include the bearing on the connected parts. This includes the bearing of bolts on spars.

.272 CASTINGS.

1. The additional ultimate factor of safety for castings is to account for the reduction in strength due to internal imperfections and also for the difference between the actual physical properties of the casting and the properties of cast test bars. It should be noted that when this factor is used, the 50% stress reduction specified in ANC-5 for casting materials may be disregarded. Consideration will be given to reduction in the specified ultimate factor of safety when suitable means of internal inspections are used and when, in addition, it can be shown that such means of inspection will result in the acceptance for use of only those castings having a definite value of minimum strength at the critical sections.

74 WIRES AT SMALL ANGLES.

1. The requirement 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 so adjusted 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.

77 CONTROL SURFACE HINGES AND CONTROL SYSTEM JOINTS.

1. It will be noted that it is unnecessary to prove the ultimate strength of ball and roller bearings if the limit load does not exceed the manufacturer's non-Brinell rating. If, however, the ultimate factor of safety of the bearing is proved, consideration will be given to the use of a yield factor of safety of less than 1.0 with respect to the manufacturer's non-Brinell rating provided that such use is substantiated by tests.

79 1. The purpose of the requirement is to provide additional strength for that portion of the wing structure which transmits the main landing gear reactions to the fuselage. It applies to all parts of the wing affected, including fittings of a type the failure of which would impair the strength of the wing in flight.

PROOF OF STRUCTURE.

000 1. Acceptable methods for computing the allowable loads and stresses corresponding to the minimum mechanical properties of various materials are given in the Army-Navy-Commerce Publication ANC-5, "Strength of Aircraft Elements", obtainable from the Superintendent of Documents, Washington, D.C., for 35¢.

01 COMBINED STRUCTURAL ANALYSIS AND TESTS.

1. The results of load tests as referred to in the requirement may be interpreted as the results of tests on similar structures when such tests are applicable.

022 1. Detailed recommendations as to acceptable methods of conducting structural tests are contained in Inspection Handbook, Chapter VIII, "Test Procedure".

2. Since it is required that the determination of test loads, the apparatus used in tests, and the methods of conducting tests shall be satisfactory to the Administrator, it is strongly recommended that, in the case of structural tests on all major units, the above items be fully covered by a report submitted to and approved by the Administrator before the actual tests are conducted.

1 PROOF OF WINGS

A DETERMINATION OF SPAR LOADING

1. 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.

2. 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 Fig. 26 and are <u>all expressed as fractions of the chord</u> at the station in question.

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TABLE IV

COMPUTATION OF NET UNIT LOADINGS (CONSTANTS)

(Ref. CAM 04.32-A3)

		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.

(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 limit load factor representing the inertia effect of the whole airplane acting at the CG. 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 CAM 04.218.

3. The computations required in using the above method are outlined in Tables IV and V, 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 curve. Fig. 18c is used for this purpose, together with the value of K_b obtained for this figure, as outlined in CAM 04.217-D.

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 CP by the following equation, using item numbers from Tables IV and V:

$$C'_{M_a} = (14) \times ((6) - CP')$$

d. 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.

e. 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).

TABLE V

COMPUTATION OF NET UNIT LOADINGS (VARIABLES)

(Ref. CAM 04.31-A3)

CONDITION ----

q	$C_{N_{I(etc)}}$	C'_C	C'_M or C.P.	n_2	n_{x_2}

		(Refer also to Table IV)	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_2 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_2 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_2} x 8						
	31	29 + 30						
	32	y_c = 31 x 2, lbs/inch						

B DETERMINATION OF RUNNING CHORD LOAD

1. 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 = chord coefficient at each station. The proper sign should be retained throughout the computations.
q = dynamic pressure for the condition being investigated.
n_{x2} = net limit 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 CAM 04.218. Note that when C_C is negative, n_{x2} will be positive.
e and C' are the same as in CAM 04.31-A2.

2. The computations for obtaining the chord load are outlined in Table V, Items 28 to 32. The following points should be noted:

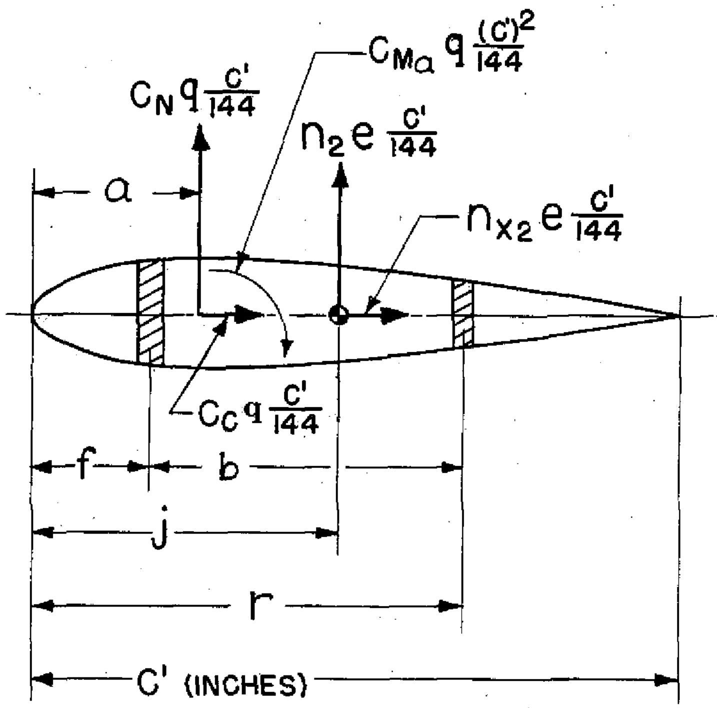
a. The value of C_C, item 28, can usually be assumed to be constant over the span. The only variation required is in the case of partial-span wing flaps or similar devices.
b. The relative location of the wing spars and drag truss will affect the drag truss loading produced by the chord and normal air forces. This can easily be accounted for by correcting the value of C_C as indicated in CAM 04.129-A2 and Fig. 10.

3. It is often necessary to consider the local loads produced by the propeller thrust and by the drag of items attached to the wing. The general rules concerning these items are outlined in CAM 04.217-C. 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 and 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.

C DETERMINATION OF RUNNING LOAD AND TORSION AT ELASTIC AXIS

1. 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.

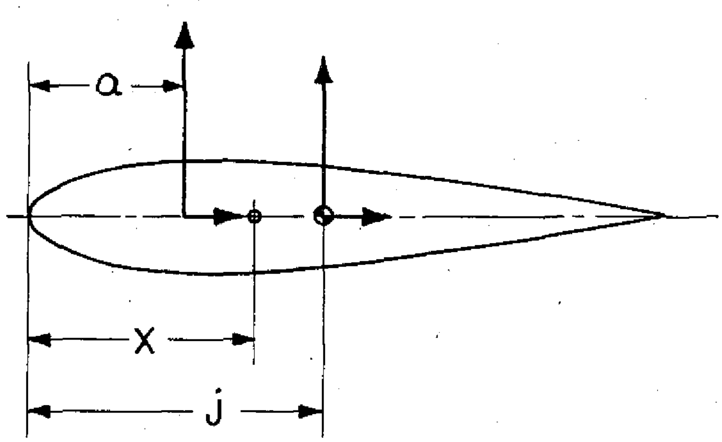
2. As shown in Fig. 27, x denotes the location of the reference axis, expressed as a fraction of the chord. The net running load along the locus of the points x and the net running torsion about these points are found from the following equations:



ALL VECTORS SHOWN IN POSITIVE SENSE
(REF. CAM 04.31-A2)

FIG. 26

UNIT SECTION OF A CONVENTIONAL 2-SPAR WING



(REF. CAM 04.31-C2)

FIG. 27

SECTION SHOWING LOCATION OF ELASTIC AXIS

FIGS. 26 & 27

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

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

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 CAM 04.31-A. (As noted previously, n_2 will always be negative in positively accelerated conditions.)

3. The computations required for this form of analysis can be conveniently carried out through the use of tables similar to Tables IV and V. The items appearing in each table would be changed to correspond to the equations given in 2 above. The computation of the running chord load can be made in the manner outlined in CAM 04.31-B.

D LIFT-TRUSS ANALYSIS

1. 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.

2. 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 per cent.

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

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

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, and
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.

3. 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 assumption is usually 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. Design Condition VI (CAR 04.2136) 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 per cent) 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.

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

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

2. Set-up. The wing should be mounted on the fuselage or a suitable jig, either of which should be anchored solidly to the floor or wall to prevent movement or displacement of the wing. The landing gear should be blocked on the airplane. The torque load may be applied to the wing tip through several beams clamped to the wing as near to the tip as is practical, such as the outermost drag truss compression rib location. The platform cables should be attached to the torque beams an equal distance forward and aft of the elastic axis of the wing. This axis may be located experimentally by rocking the torque beam and noting the nodal point on the wing chord as viewed from the tip. Typical set-ups are shown in Figure 27a. Care should be taken to see that the strength of the local wing structure at the points of application of the torque loads from the beams is adequate. For conventional two spar wood wings, it is advisable to apply the load directly to the spars through wood blocks rather than attempt to carry the load through a rib to the spars. Wings which are to be fabric covered should be tested uncovered, unless a certain amount of conservatism is considered in comparing the results with Figure 32, in order to simulate the conditions found in service.

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

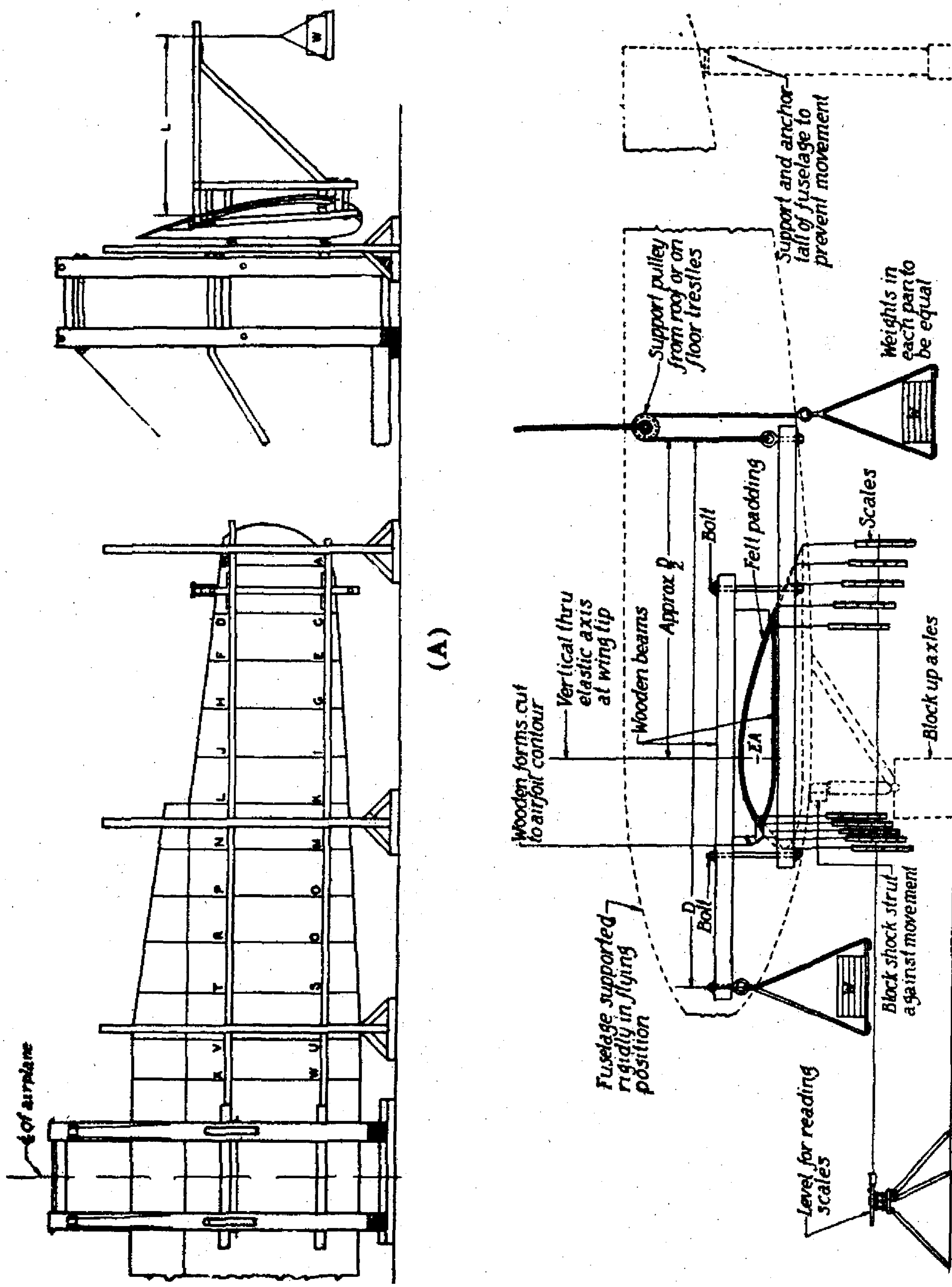


FIG. 27 a SET-UP FOR TORSIONAL TEST OF WING

3. Loading. The following procedure may be used:

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

b. Add a sufficient amount of weight to each platform until readable deflections are obtained. In general, for most aircraft from 1500 lbs. to 25,000 lbs. gross weight, it will be found desirable to make this first torque moment (in.-lbs.) equal numerically to twice the gross weight of the airplane. For aircraft below 1500 lbs. gross weight and biplanes, 70% of the above values may be used as a first trial. Care should be taken to include the tare weights of the platforms in the torque computations.

c. Take readings of all scales.

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

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

f. The data to be recorded are: the load applied; its lever arm; the deflection readings at selected points; and the exact location of these points both along the span and along the chord of the wing. It would be desirable to use a table such as shown on page .3-11 which would include all computations necessary for determining C_{TR}.

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

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

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

or

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

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

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

WING TORSION TEST OF__________ MODEL__________ SERIAL NO.__________

DATE__________ TORQUE ARM__________ inches

BY__________ LOCATED__________ inches from wing tip

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

DEFLECTION READINGS OF__________ WING (in.)

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

DEFLECTIONS OF__________ WING (in.)

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

TOTAL DEFLECTION OF__________ WING (in.) = F + R

1. W_1							
2. W_2							
3. W_3							

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

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

1. W_1							
2. W_2							
3. W_3							

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

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

1. W1						
2. W2						
3. W3						

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

1. W1						
2. W_2						
3. W_3						

Remarks:

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

where

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

$d\theta$ = Angle of twist in degrees, in length dL (in inches), caused by a torque of M inch pounds. Referring to the curve of angle of twist (θ) vs. semi-span (L) shown in Figure 27b, it will be seen that $\frac{d\theta}{dL}$ = slope of the tangent drawn to the curve at any given point. Hence, it is only necessary to draw the required tangent to the curve at the value of L at which the C_{TR} is desired and obtain $\frac{d\theta}{dL}$ to use in the above formula for C_{TR}.

It is very important that the tangent line be drawn accurately. This can best be done by first drawing the reflected curve to the point of tangency (original curve may be drawn on transparent paper and used reversed or the tangent spotted in directly by use of a small mirror), and then by bisecting the resulting angle, as shown in Figure 27b for a wing section 60 inches from the wing tip. The tangent line should be extended to both axes so that the slope of the line may be computed accurately which in this example is equal to $d\theta_{60} / dL_{60}$.

C_{TR} should be computed for each of the three torque conditions used at a number of points along the wing semi-span and plotted against the distance from the wing tip (L). This curve will show the variation of torsional rigidity throughout the semi-span and may be used for purposes of comparison with wings similarly tested. See CAM 04.404-2 and Figure 32.

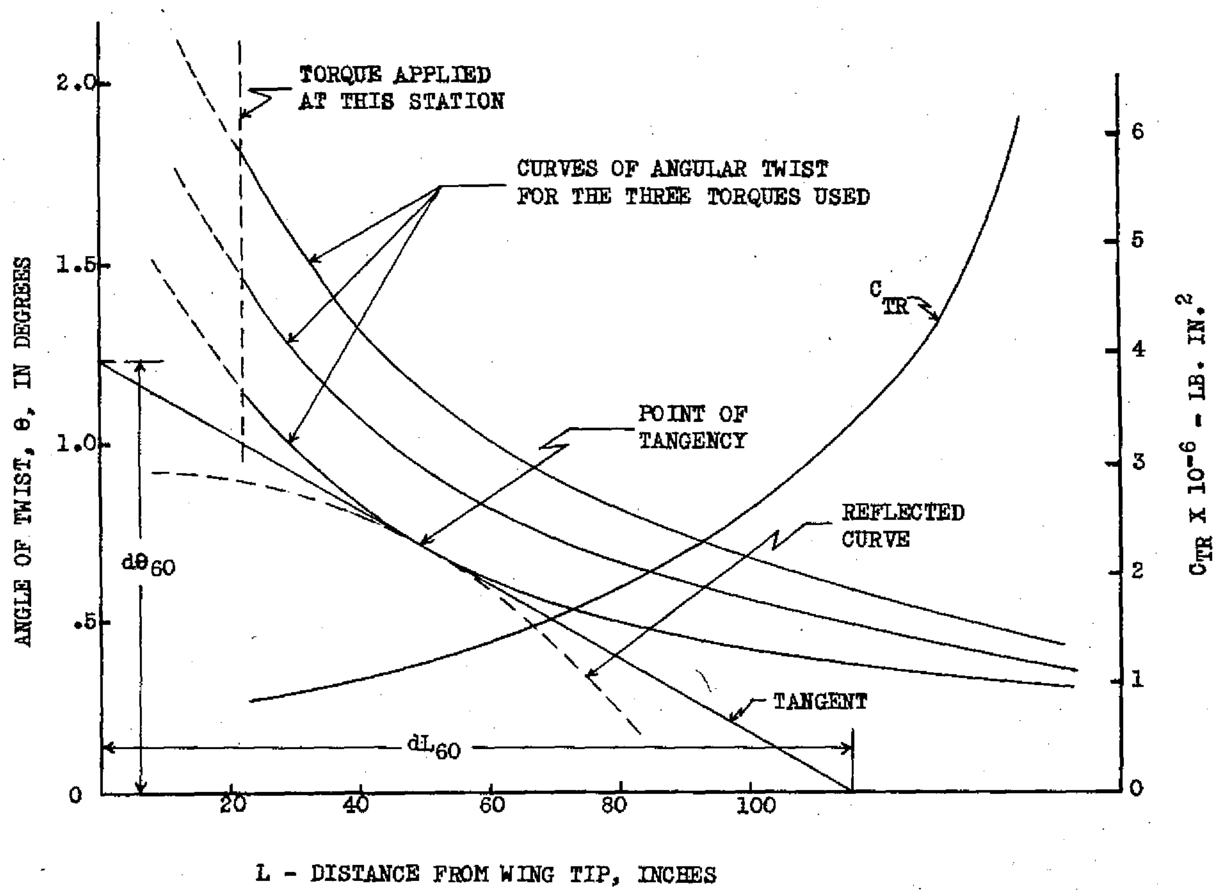
.311 BEAMS.

A WOOD SPARS.

1. The allowable total unit stress in spruce members subjected to combined bending and compression is covered in ANC 5, Section 2.41.

B METAL SPARS - GENERAL.

1. 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 are in the same ratio as are 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 computed values of EI which are accurate within ± 5 per cent.



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

FIG. 27b PLOT OF RESULTS OF TORSION TESTS

2. 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 chord by riveting; 0.75 for beams with webs having lightening holes of such shape that the beam cannot be analyzed as a truss.

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 beams 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. When 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 in 04.311-B1. 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 in 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. When 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 is 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.

4. 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:

a. 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.
b. If the wing covering is metal, suitably stiffened, the bending laterally may be neglected.

5. 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 a member, such 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. When a more exact method of analysis is employed, each member should be analyzed for the proper combination of axial load and end moment.

D THIN-WEB METAL SPARS.

1. 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 information on this subject see NACA Technical Note No. 469. The analysis should cover the attachment of the web to the flanges.

E STRESSED-SKIN WINGS.

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

a. 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 that cut-outs are properly reenforced. The axial loads in the spars due to chord loads should not be neglected in the spar analysis.

b. 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 per cent of the moments of the wing due to the beam components of the air loads. The spars should be designed to carry at least 90 per cent of these moments. When such covering is removable or contains large openings or other discontinuities between the spars on either surface of the wing, proper reduction in 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.

2. **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 limit 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:

a. Methods of analysis involving the use of the elastic axis of the wing are acceptable 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.

b. Analyses of skin-stressed wings involving the strength of sheet and stiffener combinations, or the strength of thin-web girders, should be supplemented by data on 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.

.3110 SECONDARY BENDING

1. 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 ultimate loads, rather than on the limit loads.

.3111 LATERAL BUCKLING.

1. 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 per cent. 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 per cent may be assumed.

.313 RIBS

A TEST REQUIREMENTS.

1. The rib tests required should at least cover the positive high angle of attack condition (Condition I) and a medium angle of attack condition. The total load to be carried by each rib should equal 125 per cent of the ultimate load over the area supported by the rib. For the medium angle of attack condition, the load factor should be taken as the average of the ultimate load factors for conditions I and III.

2. The leading edge portion of the rib may be very severly loaded in conditions II and IV. An investigation of the maximum down loads on this portion should be made when V_g exceeds 200 mph. (See CAM 04.217-B2). When this requirement does not apply, it should be demonstrated that the rib structure ahead of the front spar is strong enough to withstand its portion of the test load acting in the reverse direction. A test for this condition will be required in the case of a rib which appears to be weak.

3. No less than two ribs should be tested in either loading condition. For tapered wings a sufficient number of ribs should be tested to show that all ribs are satisfactory.

B TEST LOADINGS.

1. The following loadings are acceptable for two-spar construction when the rib forms a complete truss between the leading and trailing edges. (See CAM 04.217-B1 for other cases.)

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

(1) PD (Pressure Distribution) classification - this is expressed by a capital letter followed by a two digit number such as C 10, B 11, D 12, etc. For the present purpose, only the number portion of the classification need be considered.

(2) $C_{m_{a.c.}}$ - moment coefficient about the aerodynamic center.

(3) Camber - in percent chord. (This is necessary only in the case of airfoils having a "12" pressure distribution classification.)

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

b. For the medium angle of attack condition 16 equal loads should be used on ribs of chord greater than 60 inches, 8 equal loads for chords less than 60 inches. In either case the total load shall be computed as specified in CAM 04.313-A1. When 16 loads are used, they shall be applied at 8.34, 15.22, 19.74, 23.36, 26.60, 29.86, 33.28, 36.90, 40.72, 44.76, 49.22, 54.08, 59.50, 65.80, 73.54 and 85.70 per cent of the chord. When 8 loads are used they shall be so arranged as to give comparable results.

2. When the lacing cord for attaching the fabric passes entirely around the rib, all of the load should be applied on the bottom chord.

3. When the covering is to be attached separately to the two chords of the rib, the loading specified in paragraph 1 of this section should be modified so that approximately 75 per cent of the ultimate load is on the top chord and 50 per cent on the bottom, the total load being 125 per cent of the ultimate load.

.32 PROOF OF TAIL AND CONTROL SURFACES.

1. 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.

.321 1. 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 a certain counter wire will not become slack before the ultimate 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.

TABLE V_c - RIB LOAD POINTS FOR HIGH ANGLE OF ATTACK

PD Classification	(2) Camber	$C_{m_{a.c.}}$	Load Points in Percent Chord															
			1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
(1) 10 & 11	Any Value	(3) 0 to -.019	.5	1.9	3.4	5.2	7.2	9.6	12.4	15.5	19.0	23.4	28.2	33.2	40.3	46.2	72.0	90.0
		-.020 to -.039	.5	2.0	3.5	5.6	8.0	10.5	13.4	16.8	20.8	25.2	29.8	35.3	42.1	50.2	72.0	90.0
		-.040 to -.059	.7	2.0	4.0	6.3	8.8	11.4	14.8	18.5	22.8	27.2	32.7	38.0	44.8	52.8	72.0	90.0
		-.060 to -.079	.8	2.6	4.5	6.7	9.5	12.7	16.2	20.0	24.2	28.8	34.0	40.0	46.7	54.4	72.0	90.0
		-.080 to -.099	.8	2.8	5.0	7.5	10.6	13.7	17.4	21.2	25.7	30.3	35.5	41.5	47.8	55.4	72.0	90.0
		-.10 or Greater	.8	3.0	5.5	8.2	11.4	14.8	18.8	22.7	27.3	32.2	37.5	42.9	49.6	57.5	72.0	90.0

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

TABLE V_d - RIB LOAD POINTS FOR HIGH ANGLE OF ATTACK

PD Classification	Camber	$C_{m_{a.c.}}$	Load Points in Percent Chord															
			1	2	3	4	5'	6	7	8	9	10	11	12	13	14	15	16
12	0.0 to 2.9	0.00 to -.0199	.7	2.3	3.9	5.8	7.9	10.4	13.1	16.3	20.1	24.4	28.9	34.3	41.0	49.0	72.0	90.0
		-.02 to -.0399	.8	2.5	4.5	6.4	8.7	11.8	14.1	17.6	21.3	25.6	30.6	36.2	43.2	51.1	72.0	90.0
	3.0 or Greater	0.00 to -.0199	.8	2.5	4.5	6.5	8.7	11.0	13.5	16.4	19.7	23.6	28.0	33.5	39.7	47.7	72.0	90.0
		-.02 to -.0399	.9	2.8	6.0	7.2	9.6	12.0	14.7	17.9	21.5	25.6	30.1	36.5	41.8	49.7	72.0	90.0

.323 VIBRATION TESTS

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

2. Vibration equipment is available at the Civil Aeronautics Administration offices at LaGuardia Field, Long Island; Kansas City, Missouri, and Santa Monica, California. Loan of this equipment may be obtained by contacting the Regional Manager. It is especially important that the manufacturer pay particular attention to the instructions furnished with the above vibration equipment, in order that satisfactory results may be obtained. However, the manufacturer may use other types of vibration equipment, in which case a report should be submitted containing a complete description of the equipment and sufficient test data to substantiate its accuracy. When a resilient element such as a spring or rubber ball is incorporated in the driving unit of the vibrator, its stiffness should be low relative to the stiffness of the surface being vibrated, in order to avoid misleading results. If the manufacturer desires, arrangements can be made to obtain experienced Civil Aeronautics Administration personnel to supervise the operation of the vibration equipment.

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

4. Location of Vibrator on the Structure.

a. The proper location of the vibrator on the structure is important in obtaining satisfactory results. Suggested vibrator locations for exciting various modes of vibration are given in Fig. 28 .

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

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

(1) The location is of primary importance and should be at a point of large deflection. See Fig. 28

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

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

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

VIBRATION MODES AND VIBRATOR LOCATION

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

ITEM	SURFACE	MODE	DESCRIPTION OF MODE AND SUGGESTED VIBRATOR LOCATION
1	Rudder (Single)	(As a Unit)	Rudder swinging back and forth, under the spring action of the control cables. (Vibrator aft of horn on rudder T.E.)
2	Rudder (Single)	Torsion	Torsional vibration of the rudder, under the spring action of the torque tube (Nodal line extends from trailing edge to torque tube.) (Vibrator near upper end on rudder T.E.)
3	Rudders (Twin Tail)	Sym.	Same as item 1 above, except both rudders swinging inward (or outward) at same time. (Vibrator aft of horn on rudder T.E.)
4	Rudders (Twin Tail)	Unsym.	Same as item 1 above, except both rudders moving to right (or left) together. (Vibrator aft of horn on rudder T.E.)
5	Rudders (Twin Tail)	Torsion	Same as item 2 above.
6	Elevator	Sym.	Both elevators swinging up or down together under the spring action of the control cables (or push pull tubes) (Vibrator at inner end of one elevator T.E. or aft of horn on single elevator.)
7	Elevator	Unsym.	One elevator (or 1/2 of single elevator) moving up, other down at same time under spring action of torque tube. (Nodal line in plane of surface, extending aft from point near center of elevator spar.) (Vibrator at center of semi-span near T.E.)
8	Aileron	Sym.	Each aileron as a unit swinging up and down together, under the spring action of the control cables (or push pull tubes). (Vibrator between center and inner end at aileron T.E.)
9	Aileron	Unsym.	One aileron as a unit moving up, other down at same time, under spring action of control cables. (Vibrator aft of horn at aileron T.E.)
10	Aileron	Torsion	Torsional vibration of aileron, under the spring action of the torque tube (Nodal line extends from trailing edge to torque tube.) (Vibrator near outer end at aileron T.E.)
11	Wing	Sym. Bending	Wing bending as a beam supported at its midpoint (fuselage), or as a braced beam supported at the brace points. (Vibrator near wing tip approximately on elastic axis of wing.)
12	Wing	Unsym. Bending	Same as item 11 above, except that right and left halves of the wing move in opposite directions at the same time. Usually only important in large multi-engine aircraft. (Vibrator near wing tip approximately on elastic axis of wing.)
13	Wing	Sym. Torsion	Torsional vibration of the wing about a spanwise axis. Right and left halves of the wing move in same direction about this axis at the same time. (Vibrator near wing tip, forward or aft of elastic axis of wing.)
14	Wing	Unsym. Torsion	Same as 13 above, except right and left halves of the wing move in opposite directions about a spanwise axis at the same time. (Vibrator near wing tip, forward or aft of elastic axis of wing.)
15	Wing (Biplane only)	Cellule Sym. Torsion	Same as item 13 above, except center of rotation may be located between upper and lower wings. (Vibrator acting in chordwise direction at upper wing outer rear interplane strut attachment.)
16	Wing (Biplane only)	Cellule Unsym. Torsion	Same as item 14 above, except center of rotation may be located between upper and lower wings. (Vibrator acting in chordwise direction at upper wing outer rear interplane strut attachment.)
17	Fuselage	Torsion	Whole tail unit and fuselage vibrating torsionally about longitudinal axis. (See item 23 also) (Vibrator on fin or stabilizer - on hinge line, at tip of surface.)
18	Fuselage	Side Bending	Whole tail unit vibrating in side direction about vertical axis through fuselage forward of tail unit. Usually important only on large airplanes. (Vibrator at tail end of fuselage.)
19	Fuselage	Vertical Bending	Same as item 18 above, except vibrating in vertical direction about horizontal axis through fuselage. Only important in very large aircraft having fuselage cutouts in top and bottom sides. (Vibrator at tail end of fuselage.)
20	Stabilizer	Sym. Bending	Stabilizer bending as a beam supported at its midpoint (fuselage). Usually only important in cantilever tail surface designs. If wire or strut braced with small tip overhang, the mode of vibration may not be such as to permit interaction with the elevator. (Vibrator near outboard end of stabilizer.)
21	Stabilizer	Sym. Torsion	Torsional vibration of the stabilizer. Similar to item 13, for the wing. Usually only important for cantilever and twin tail aircraft. (Vibrator near outer end of stabilizer at L.E.)
22	Stabilizer	Unsym. Torsion	Torsional vibration of stabilizer. Similar to item 14 for the wing. Nodal line at ℄ of stabilizer (fuselage). (Vibrator at leading edge of stabilizer or at trailing edge of elevator. Elevator clamped to stabilizer.)
23	Stabilizer	Rocking about its fuselage attachments	See item 17 also. Stabilizer as a unit rocking about its fuselage attachments. Usually only important for twin tail aircraft. (Vibrator on fin or outboard end of stabilizer on twin tail aircraft, or near outboard end of stabilizer - elevator hinge line on single tail aircraft.)
24	Fin (Single tail)	Bending	Fin bending as a beam fixed at one end. Usually only important in cantilever tail surface designs. (Vibrator near upper end of fin near rudder hinge line.)
25	Fin (Single tail)	Torsion	Torsional vibration of the fin. Similar to item 21 for the stabilizer. (Vibrator on fin L.E. near upper end.)
26	Fin (Twin Tail only)	Bending (Sym. with respect to attaching fin to stabilizer)	Fin bending as a beam fixed at its attachment to stabilizer. (Vibrator near upper, or lower, end of fin near elastic axis of fin.) (Upper and lower portions may have different frequencies.)
27	Fin (Twin Tail only)	Bending (Unsym. with respect to attachment of fin to stabilizer)	Fin bending as a beam fixed at its attachment to stabilizer.
28	Flap	Torsion	Torsional vibration of outboard end of flap. Similar to item 10 above. Assuming irreversible control arm used. (Vibrator near outboard end of flap on T.E.) Only important when flap extends outboard on wing beyond 50% semi-span location.
29	Tabs (Rudder Aileron Elevator)		Note: Effect of vibrator weight on tab frequency should be investigated when vibrator is attached directly to tab.
30	Trim or Balance Tabs	Torsion	Torsional vibration of the tab under the spring action of the torque tube. Node on T.E. (Vibrator on tab near one end.)
31	Servo Tabs	Sym.	Tab as a unit swinging back and forth under the spring action of the control cables. (Vibrator on adjacent tab supporting structure, or on tab itself.)
32	Servo Tabs	Torsion	Same as item 10 above. (Vibrator on tab near one end.)
33	Balance Weights for movable surfaces - mounted on long supports	Bending of support	Support bending as a beam (fixed at one end) in various planes depending on the rigidity. Most important directions are vertical and sideways with relation to the airplane centerline. (Vibrator near balance weight support.)

FIG. 28

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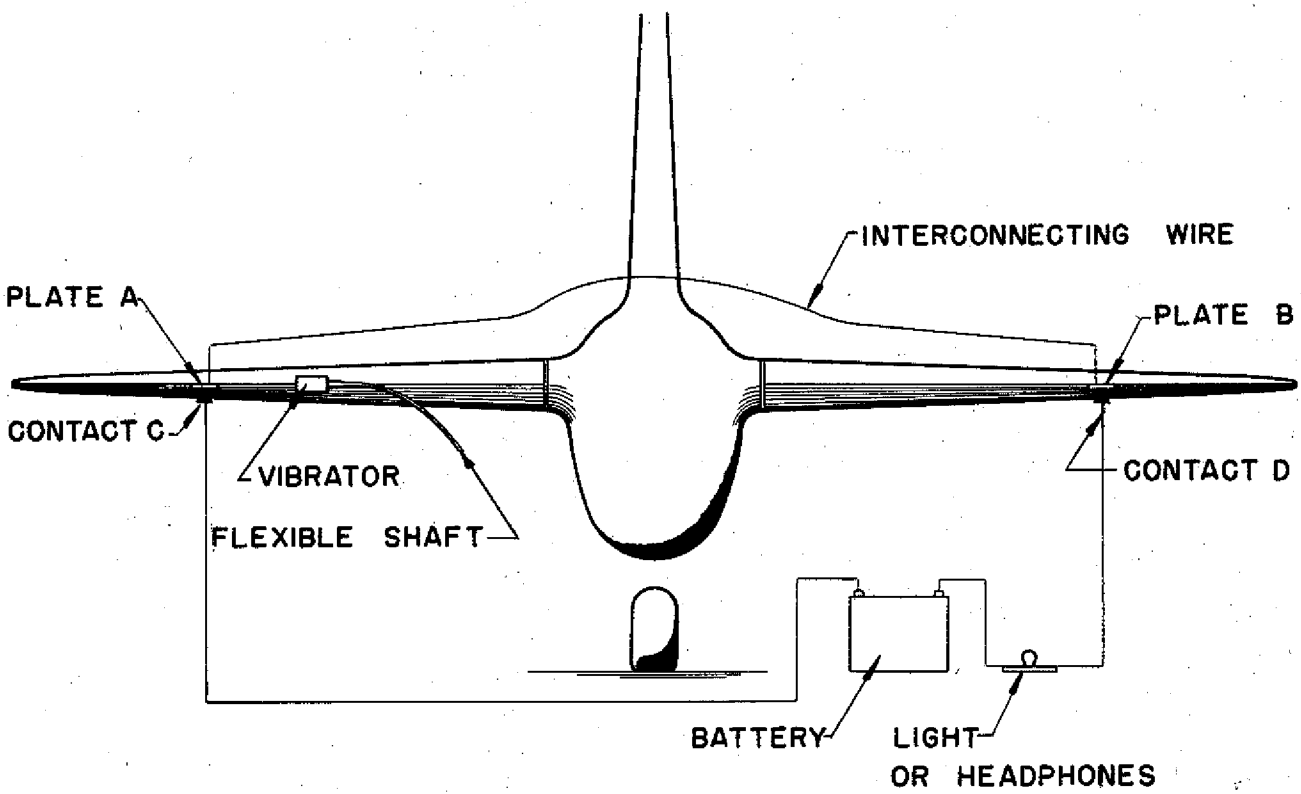
5. Testing. A certain amount of experience is necessary in recognizing the various modes and resonant frequencies. In conducting the tests, the vibrator should be placed on the structure as suggested and then operated at increasing speeds until a response peak is reached (the amplitude of vibration of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

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

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

The phase relationship of vibrating parts may be determined by the method shown in Fig. 28a as applied to the particular case of the elevators. The metal plates A and B, attached to the trailing edges of the elevators and interconnected with a wire, are necessary only in the case of fabric covered surfaces or surfaces which have a poor electrical interconnection. When the parts are vibrating the phase relationship may be determined by manually holding the leads C and D close to the surfaces so that intermittent contact is made during each cycle. If the light flashes or clicks are heard in the headphones at regular intervals (with the contacts in the same side; i.e., upper or lower), the surfaces are vibrating in phase, whereas, if the light does not flash, or no click is heard in the headphones, the surfaces are out of phase. This should be verified by reversing one contact, for example, putting contact D on the upper side.

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



(REF. CAM 04.323-7)

FIG. 28 A TEST SET-UP FOR DETERMINATION OF PHASE RELATIONSHIP

8. A number of frequency trend curves are shown in Figure 28b. These will be expanded as more data become available. The wing and stabilizer data shown on this figure were obtained on unbraced surfaces. It should be appreciated that these trends are approximate and can serve as a rough guide only. Many factors, such as the type of construction involved, etc., will have a marked influence on the actual values which will be obtained for any particular design.

9. Fig. 28 gives a detailed description of the possible modes that may be observed during the tests and includes suggested vibrator locations for each mode. In general, only a fraction of the modes listed will be applicable to any one airplane.

3 PROOF OF CONTROL SYSTEMS.

1. 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.

2. An investigation of the strength of a control system includes that of the various fittings and brackets used for support. In particular, the rigidity of the supporting structure is important especially in aileron, wingflap, and tab control systems.

4 PROOF OF LANDING GEAR.

1. The landing conditions tabulated in Figs. 24 and 25 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 different members. If the design is such that it is obvious that a certain condition cannot be critical for any member, such a condition need not be investigated. It will probably be necessary, however, to determine the loads acting on the fuselage in all conditions, for use in the fuselage analysis.

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

a. 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.
b. Tension loads are positive, compression loads negative.
c. Moments are represented by vectors according to the "right hand" rule.
d. The basic axes also represent positive moment vectors, each axis being the axis of rotation for the corresponding moment. (Note that changing the sign of a moment reverses the direction of the vector.)
e. In writing the equations of equilibrium, all forces are initially assumed to be tension, i.e., positive. The true nature of the forces will be indicated by the sign of the vector obtained in the final solution.
f. Moments can be combined vectorially in exactly the same manner as forces and can also be solved for by the same methods.

NATURAL FREQUENCY TRENDS

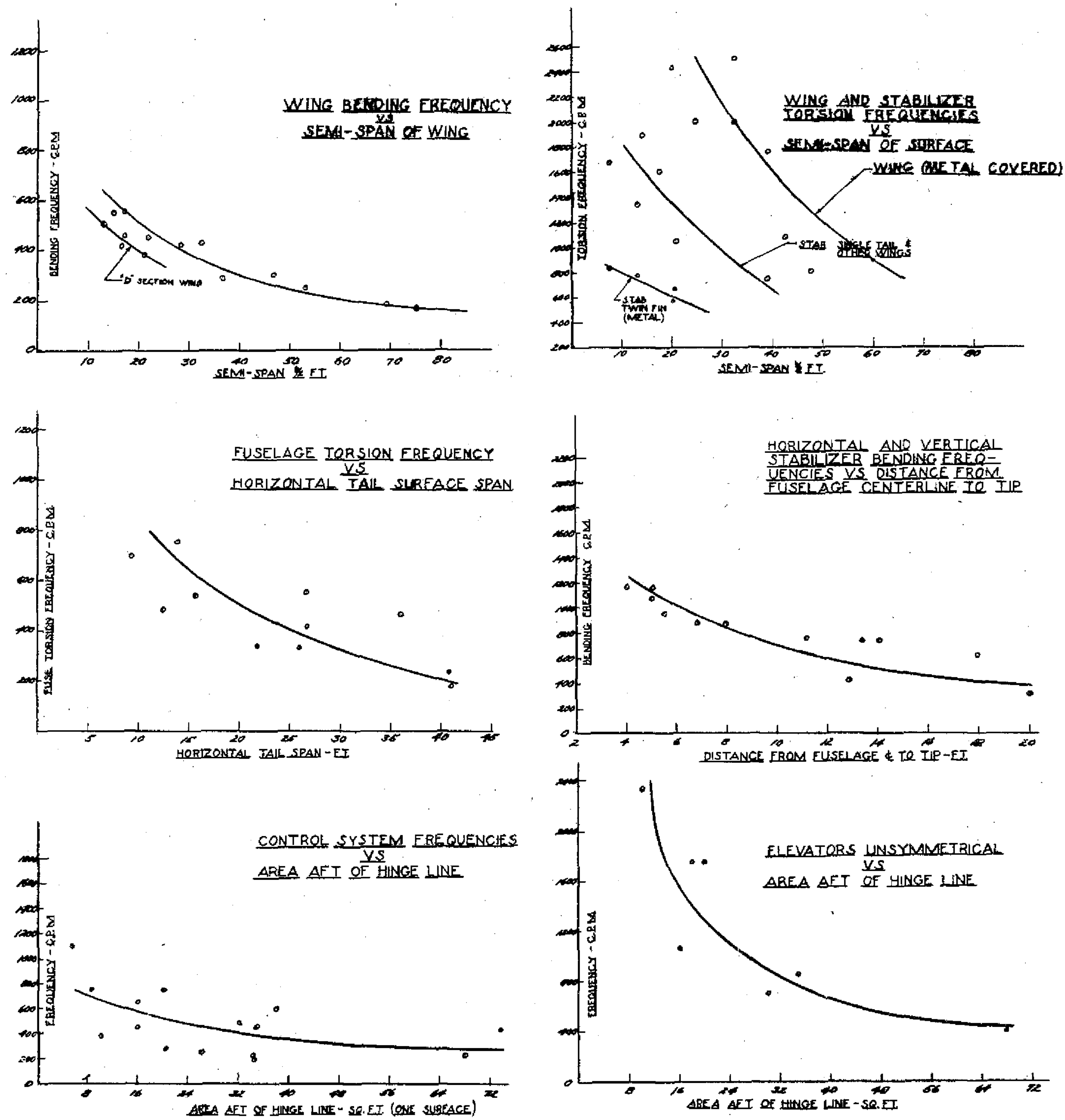


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FIG. 28b

340 ENERGY ABSORPTION TESTS.

A GENERAL.

1. As stated in CAR 04.440 the shock-absorbing system must so limit the acceleration in specified drop tests (CAR 04.2411 and CAR 04.2420) that the ultimate load used in the design of any member is not exceeded. In general this is interpreted to mean that the acceleration recorded in drop tests should not exceed the ultimate load factor for the condition being tested. In infrequent cases the ultimate load factor is exceeded in a drop test but, due to margins of safety, the ultimate strength of any member is not exceeded. In such cases the true margins should be listed in the analysis. Drop tests alone from the required height are not acceptable as proof of strength. Any yielding of structural components in drop tests will be subject to review and further consideration.

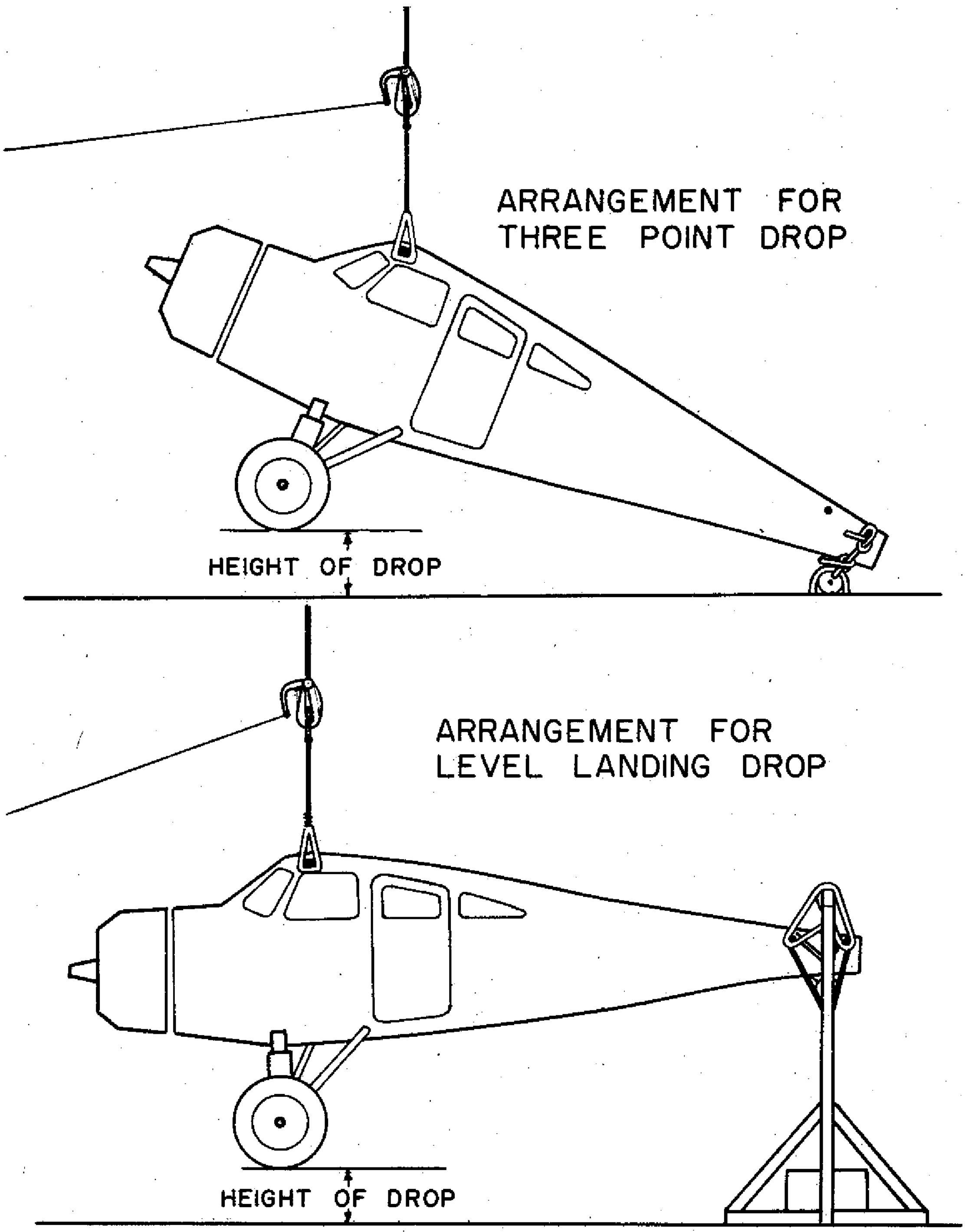
2. Many cases arise which involve approval of a higher gross weight, the necessary greater height of drop, and/or the use of different tires from those used in the original drop test. In some such cases it may be possible to demonstrate compliance with the requirements without an additional drop test. In general, however, time and expense will be saved if such changes are anticipated and substantiated at the time of the original drop test.

3. In the drop test it is acceptable to allow for the effect of wing lift present in the landing maneuver only when such effect is substantiated, i.e.. when a completely rational analysis of the problem is made.

B MAIN GEAR TESTS - FIRST METHOD

1. The first method of testing involves dropping the fuselage or equivalent structure with the complete landing gear attached. A beam with the proper location of landing gear fittings may be considered as equivalent structure. Tests should be made for either the three-point or level landing condition, whichever is critical with respect to energy absorption, i.e., whichever (in the case of conventional gear) involves a smaller component of wheel travel (relative to the airplane) in the direction of the resultant external force. See E below for considerations in the case of nose wheel type gear. However, tests should also be made for the other condition if it involves higher bending loads in the shock absorber than does the critical condition.

2. For the three-point landing test the rear end of the fuselage is held in place on the floor as shown in Fig. 29. For the level landing test the rear end of the fuselage is raised until the center of gravity of the loaded airplane is vertically above the wheel axles, or until the fuselage is inclined at a nose-down angle of 14 degrees, whichever is reached first. The rear end of the fuselage is then held in this position, as shown in Fig. 29. Care should be taken, particularly in the level landing drop test, to restrain the rear end of the fuselage from rising as a result of the impact. When the airplane is in position for the drop it is advisable to place sand bags under the structure near the CG to minimize the damage in case of failure.



(REF. CAM 04.340-B)

FIG. 29 SET-UP FOR LANDING GEAR DROP TEST

3. The accelerations should be obtained by use of a recording accelerometer, a space-time recorder, or other suitable means attached or connected as close to the CG as possible. The NACA has a number of accelerometers which are approved for this purpose and will lend them to manufacturers on request. In this connection it should be noted that when accelerometers are used they should have a very short natural period, i.e., 1/20 second or less. In general the use of a recording device in which a mass travels an appreciable distance will be questioned.

4. The following procedure should be observed in conducting the tests:

a. For tests in the level landing attitude the weight on the main wheels should be the full gross weight of the airplane. Note that this does not require that additional weight be used to duplicate the stress analysis resultant load which includes the vertical and aft components. In the three-point attitude the weight on the main wheels should be the static reaction for this attitude with the full gross weight at its most forward CG location.

b. The tire pressure should be the same as that recommended by the Tire and Rim Association for use in service. Likewise the proper fluid, fluid level and air pressure (if any) of the shock absorber should be used.

c. A hoisting sling with a quick-release mechanism is attached to the fuselage near the center of gravity. By means of this hoist the front end of the structure is raised until the tires are clear of the floor by the desired amount. When using the tape type space-time recorder it is desirable to mark the "static" and "clear" positions on the tape.

d. The floor, or a steel plate placed under the tires, may be greased if desired to prevent the tires from rolling off the rims if there is appreciable side movement of the wheels.

e. The quick-release is operated, allowing the structure to drop freely.

5. It is advisable that the drop height be increased by increments from some low value until the height specified in CAR 04.2411 is attained so that unsatisfactory characteristics can be detected before the gear is overstressed. Note that the specified height is measured from the bottom of the tire to the ground, with the landing gear extended to its extreme unloaded position.

6. The final test should be witnessed by a representative of the Administrator. The manufacturer's report should include, in addition to other data (see CAM 04.032-A), the accelerometer records or exact copies of them, with the magnitude of the maximum acceleration determined and marked thereon. A record of the maximum tire deflection should also be given.

C MAIN GEAR TESTS - SECOND METHOD

1. The second method of testing involves dropping the shock absorption unit, including wheel and tire assembly, in a special test rig. When using this method it is strongly recommended that the actual linkage ratios (wheel travel to shock absorber travel) be duplicated, and that bending in the shock absorber member (if present in service) be simulated in the test. When this is impracticable it will be acceptable to use the "in line" method (wheel, shock-absorber and load in line) outlined below provided that the following points are observed:

   a. Prior to final tests the proposed test procedure should be submitted to the Administrator for ruling as to its acceptability.

   b. Drops should be made from several different heights in order to establish the trend in accelerations.

   c. The "in line" method is not recommended when the values of K (see 2a below) exceeds 1.75.

   d. A margin between the developed acceleration and the ultimate load factor, proportional to the degree of bending present in service and the pertinent value of K, should be shown.

2. The following procedure should be observed in setting up for "in line" drop test:

   a. Determine the value of K (ratio of the static load on the strut to the static load on the tire) for the critical condition being simulated in the test (See B above and E below for considerations involved).

   b. Use a test weight equal to K times the static load on the tire. Of this test weight, the "unsprung" or "semi-sprung" portion of the jig weight, i.e., that portion of the jig weight which moves with the wheel, should be held to the minimum practicable.

   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 approximately 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. It may be possible to obtain the above characteristics by changing the inflation pressure of the original tire and by using stops.

   d. The height of free drop should be 1/K times the height specified.

   e. The foregoing adjustments are necessary in order to reduce to a minimum the errors in impact energy, piston velocity, and shock strut load. Note that such errors increase with an increase in the value of K.

D TESTS OF TAIL WHEELS AND TAIL SKIDS.

1. Tests for the energy absorption capacity of the tail wheel assembly may be made in a manner similar to that used for testing a complete main gear assembly (See CAM 04.340-B), except that the tests need be made only for the three-point condition. The test load may be obtained by loading the fuselage or by concentrating the required mass over the tail wheel.

2. In conducting these tests the front wheels rest on the floor while the tail is raised the required distance (See CAR 04.2411) and dropped. The accelerometer or space-time recorder tape is attached to the structure at a point over the wheel. Drop tests of complete assemblies, or "in-line" drops made in test rigs (See CAM 04.34-C), are equally acceptable.

3. Tests for the energy absorption capacity of tail skids should be conducted in a manner similar to that outlined above for tail wheels .

E TESTS OF NOSE-WHEEL TYPE GEAR

1. In general, the tests of main wheel and nose wheel installations may be made in accordance with the methods outlined in A to C above. The tests of each installation should be made for the most critical (most unfavorable with respect to shock absorption) of the conditions outlined in CAM 04.240-2a through 2e. Each of these conditions is assumed to be produced by the free drop from the height specified in CAR 04.2411. In determining the critical conditions, consideration should be given to the value of the component of wheel travel (relative to the airplane) in the direction of the resultant external force and also to the magnitude of this force. In general, the higher the force and the smaller the travel, the more critical the condition. In cases where question arises as to the applicability of the design conditions used it is advisable to conduct actual landing and taxiing tests with one or more accelerometers installed in the airplane.

2. In all cases the proposed test procedure, together with details of the installation, should be submitted to the Administrator for comment prior to the tests.

F TESTS AT PROVISIONAL WEIGHT

1. When advantage is taken of the provisions of CAR 04.711 in designing the landing gear only for the standard weight, it is necessary to show that the airplane is capable of safely withstanding the ground shock loads incident to taxiing and taking-off at the provisional weight. This can be demonstrated by showing that the accelerations developed in taxiing and taking-off over rough ground (off runway)are such that the limit load for any landing gear member is not exceeded. The accelerations developed in these tests should be obtained by means of a recording accelerometer.

.36 PROOF OF FUSELAGES AND ENGINE MOUNTS

A GENERAL

1. In addition to determining the loads in the main structural members of a fuselage, the local loads imposed by the internal weights which they support 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. Also, whenever critical, control system loads which occur in the specific flight or landing conditions should be combined with the primary loads. The combined stresses should be determined in such cases.

B STRESS ANALYSIS PROCEDURE

1. Weight Distribution. All major items of weight affecting the fuselage should be so distributed to convenient panel points that the true center of gravity of the fuselage and its 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:

a. 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.

b. 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, as the case may be.

c. 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. When a rational analysis is not possible, the division may be estimated.

d. 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.

e. 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.

2. Balancing (Symmetrical Conditions). Methods of balancing the airplane are discussed in CAM 04.218. It will, in general, be satisfactory to apply directly the balancing loads found in the various flight conditions. The acceleration factor applied to each item of mass in the fuselage will be the net acceleration factor as determined from the

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c. 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. When a rational analysis is not possible, the division may be estimated.

d. 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.

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a. 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.

b. 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, as the case may be.

c. 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. When a rational analysis is not possible, the division may be estimated.

d. 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.

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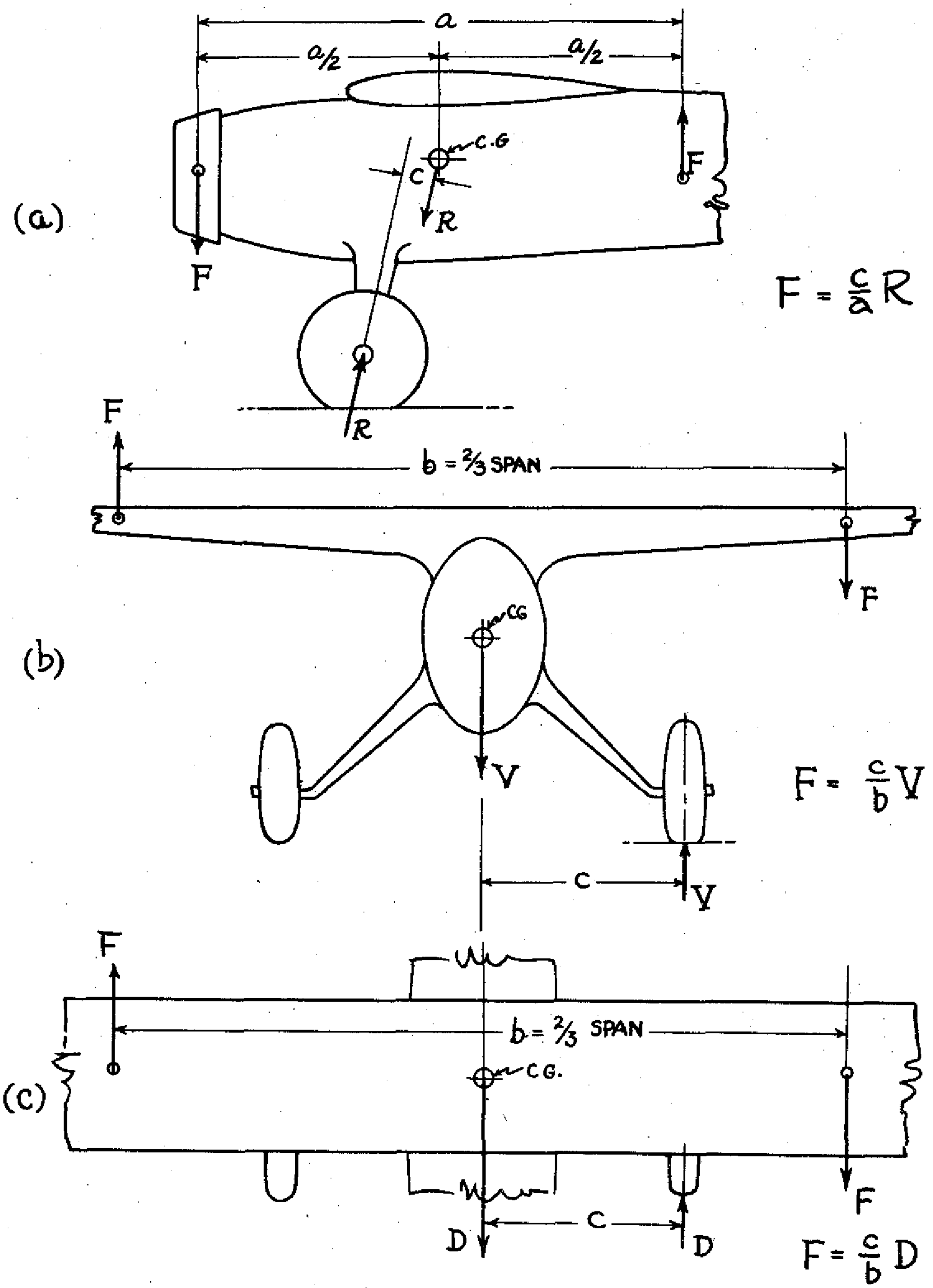
balancing computations. The basic inertia force on any item will be parallel to the resultant external applied force and will not, in general, be perpendicular to the thrust line. In certain cases the chord components of the inertia forces (i.e., the components along the thrust line or fuselage centerline) can conveniently be combined into a single force 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.

3. 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 to 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 the linear acceleration should be determined separately from those produced by angular accelearation. 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 CG of the airplane. It will usually be acceptable, in analyses of this nature, to represent the weights of major items such as wing panels, nacelles, etc., by assumed concentrated masses at the centers of gravity of the respective items. Fig. 30 illustrates approximate methods by which the fuselage can be balanced for a typical unsymmetrical landing condition (one-wheel landing).

a. Fig. 30a shows a level landing condition in which the resultant load does not pass through the center of gravity. In such a case it will generally be acceptable 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. Fig. 30b indicates an acceptable method of balancing externally applied rolling moments about thelongitudinal axis. The forces resisting angular acceleration are assumed to be applied by the wing. The arbitrary location shown is based on the fact 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 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.



(REF CAM 04.36-B3)

FIG. 30 METHODS OF BALANCING FUSELAGE FOR UNSYMMETRICAL LOADS

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c. Fig. 30c 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 CAR 04. 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 for a side load acting at the tail.

d. It should be noted that the balancing couples shown on Fig. 30 will act in addition to the inertia loads due to linear acceleration. For instance, in Fig. 30b the load V shown as a reaction at the CG 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.

C SPECIAL ANALYSIS METHODS

1. 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:

a. 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.

b. 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 or vertical trusses, or taken as 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.)

c. The diagonals of the rearmost bulkheads, i.e., the bulkheads through which the torque is transmitted to the wing, and of 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.

d. 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.

2. Engine Torque. In investigating the conditions involving engine torque, the following points apply:

a. The basic torque may be computed by the following formula:

$T = 63{,}000\ P/N$, where
T = torque in inch pounds,
P = horsepower of engine,
N = propeller speed in revolutions per minute.

b. 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 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.

c. 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. When 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.)

d. 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.

e. 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.

D ANALYSIS OF STRESSED-SKIN 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 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.
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 pertinent test data. Whenever possible it is desirable to test the entire fuselage for bending and torsion, but tests of certain component parts may be acceptable in conjunction with a stress analysis. As this subject is now being investigated by the NACA, the latest information should be obtained from that organization before the stress analysis or test methods are decided upon.

.37 PROOF OF FITTINGS AND PARTS

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

2. The additional ultimate factor of safety of 1.20 for fittings (CAR Table 04-7) is to account for various factors such as stress concentration, eccentricity, uneven load distribution, and similar features which tend to increase the probability of failure of a fitting. As noted in the Table, this factor may be covered by several other factors so that when the ultimate 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 above the value used for the primary members.

.4 DETAIL DESIGN AND CONSTRUCTION

.400 MATERIALS AND WORKMANSHIP

1. Materials and processes conforming to the specifications of the Army, Navy, S.A.E. or other responsible agencies are satisfactory. It is important that minimum specification values of strength properties given in ANC-5 be used rather than "typical" or "average" values.

2. Tolerances should be closely held in order that the assumed or tested structure is accurately reproduced. Metal sheet and tubing gages usually conform to well established specifications. Tolerances on machined parts are based on general practice and will vary from about ± .015 inch to values necessary tossecure interchangeability of mating parts. Tolerances on sheared and nibbled parts are usually ± 1/32 inch. Minus tolerances on section dimensions of wood structural members such as spars should not exceed 1/64 inch in the fully seasoned condition unless justified by check of margins. Plus tolerances are limited by assembly considerations.

3. Long assemblies such as spars with a large number of rivets will "grow" slightly as the riveting progresses. End fittings should therefore be jig installed as a last operation. A similar procedure is followed with welded assemblies. Heat treating of long welded structures results in shrinkage and in extreme cases allowances for this must be made.

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"4. In regard to wood construction, Sitka spruce has been the most extensively used species in the fabrication of primary structural members due to its high strength-weight ratio, its excellent over-all handling qualities and its uniformity in texture and properties. Several other species, however, are also suitable for such use and may be substituted for spruce, in some cases directly and in other cases, with only minor design revisions. It is assumed that spruce will, in general, be used as a basis for design, hence Figure 30A (on the reverse side of this sheet) has been set up using spruce as a standard material. Figure 30A includes comparative information on strength properties, various limiting factors on the selection of wood of aircraft quality, remarks on the characteristics of each wood listed and general notes on wood defects. This table will be revised or expanded as the need arises."

.4010 GLUING

1. High grade casein, animal, and synthetic resin glues are satisfactory. Details of composition and methods are given in Appendix IV herein. It should be noted that condition of the surface, moisture content of the wood, gluing pressure, and protective coatings as well as other factors play an important part in the making of acceptable joints.

SELECTION AND PROPERTIES OF AIRCRAFT WOOD

FIG. 30A

SPECIES OF WOOD	STRENGTH PROPERTIES AS COMPARED TO SPRUCE	PERMISSIBLE RANGE OF MOISTURE CONTENT %	SPECIFIC GRAVITY (WHEN OVEN DRY) Average	SPECIFIC GRAVITY (WHEN OVEN DRY) Minimum	POUNDS PER CU. FT. AT 15% MOISTURE CONTENT	MAX PERMISSIBLE GRAIN DEVIATION (SLOPE OF GRAIN)	MINIMUM NO OF ANNUAL RINGS PER INCH	REMARKS
1	2	3	4	5	6	7	8	9
SPRUCE (PICEA) SITKA (P. SITCHENSIS) RED (P. RUBRA) WHITE (P. GLAUCA)	100%	8-12	.40	.36 (See note 1)	27	1-15 (See note 1)	6	Excellent for all uses. Considered as standard for this table.
DOUGLAS FIR (PSEUDOTSUGA TAXIFOLIA)	EXCEED SPRUCE	8-12	.51	.45 (See remarks)	34	1-20	8	May be used as substitute for spruce in same sizes or in slightly reduced sizes providing reductions are substantiated. If used as direct substitute for spruce in same sizes, a reduction in the minimum specific gravity (column 5) to .38 is permissible. Difficult to work with hand tools. Some tendency to split and splinter during fabrication and considerable more care in manufacture is necessary. Large solid pieces should be avoided due to inspection difficulties. Gluing satisfactory.
NOBLE FIR (ABIES NOBILES)	SLIGHTLY EXCEED SPRUCE EXCEPT 8% DEFICIENT IN SHEAR	8-12	.40	.36	27	1-20		Satisfactory characteristics with respect to workability, warping and splitting. May be used as direct substitute for spruce in same sizes providing shear does not become critical. Hardness somewhat less than spruce. Gluing satisfactory.
WESTERN HEMLOCK (TSUGA HETEROPHYLLA)	SLIGHTLY EXCEED SPRUCE	8-12	.44	.40	29	1-20	8	Less uniform in texture than spruce. May be used as a direct substitute for spruce. Upland growth superior to lowland growth. Gluing satisfactory.
PINE, NORTHERN WHITE (PINUS STROBUS)	PROPERTIES BETWEEN 85% and 96% THOSE of SPRUCE	8-12	.38	.34	26	1-20		Excellent working qualities and uniform in properties but somewhat low in hardness and shock resisting capacity. Cannot be used as substitute for spruce without increase in sizes to compensate for lesser strength. Gluing satisfactory.
WHITE CEDAR, PORT ORFORD (CHARAECY-PARIS LAWSONIANA)	EXCEED SPRUCE	8-12	.44	.40	30	1-20	8	May be used as substitute for spruce in same sizes or in slightly reduced sizes providing reductions are substantiated. Easy to work with hand tools. Gluing difficult but satisfactory joints can be obtained if suitable precautions are taken.
POPLAR, YELLOW (LIRIODENDROW TULIPIFERA)	SLIGHTLY LESS THAN SPRUCE EXCEPT IN COMPRESSION (crushing) AND SHEAR	8-12	.45	.38	28	1-20	8	Excellent working qualities. Should not be used as a direct substitute for spruce without carefully accounting for slightly reduced strength properties. Somewhat low in shock resisting capacity. Gluing satisfactory.

NOTES

1. GRAIN DEVIATION. It will be noted that various specifications for aircraft lumber call for a maximum permissible grain deviation of 1 inch in 20 inches, with the exception of spruce where 1 in 15 is specified. When the spruce lumber specification was originated, it was anticipated that the proportion of the material having a slope no greater than 1 to 20 would be sufficient to provide such a slope in highly stressed members and that material having a slope between 1 to 20 and 1 to 15 would be used in other parts. Spruce having a slope of 1 to 20 or less should, therefore, be used wherever possible, however if it becomes necessary to utilize spruce having a grain slope between 1 to 20 and 1 to 15 in highly stressed members, only material having a specific gravity (when oven dry) of 0.40 or greater should be used.
2. DRYING METHODS. Woods used in aircraft construction may be seasoned by air-drying, kiln-drying or a combination of both. Some artificial drying is usually necessary to reduce the moisture content to the 8% to 12% range of values specified in the table. Kiln-drying has the advantage that atmospheric conditions are controlled, permitting the drying of wood without development of checks and other defects. Such defects are much more prevalent in air dried lumber. Kiln-drying schedules may be obtained from U. S. Army Specification No. 82-13, "Kiln Drying Process for Aircraft Lumber"; Navy Department General Specification for Inspection of Material, Appendix IV, "Lumber and Timber"; or Department of Agriculture Forest Products Laboratory Report No. 1360, "Aircraft Kiln Schedule". Western Hemlook should be kiln dried in accordance with Table 1 of those references.
3. SAWING METHODS. In all species, quarter sawed lumber is definitely preferable to flat grain stock for aircraft use. Quarter sawed stock (also referred to as edge-grain, rift-grain and rift-sawn) is defined as stock in which the rings are at an angle of 45° to 90° to the surface, and flat-grain stock (also referred to as plain-sawed or slash-grain) is stock in which the rings are at an angle of 0° to 45° to the surface. Quarter-sawed stock has less tendency to check and is less subject to bowing flatwise and to cupping, since it tends to shrink or swell equally with changes in moisture content. Resistance to indentation from tightening fittings, etc., is more uniform over an edge-grain surface. Experience with Douglas Fir in aircraft construction indicates that it should be quarter sawed in all cases.
4. DEFECTS PERMITTED.
(a) CROSS GRAIN. Spiral grain, diagonal grain or a combination of the two is acceptable providing the grain does not diverge from the longitudinal axis of the material more than specified in column 7. A check of all four faces of the board is necessary to determine the amount of divergence. The direction of free flowing ink will frequently assist in determining grain direction.
(b) WAVY, CURLY, AND INTERLOCKED GRAIN. Acceptable if local irregularities do not exceed limitations specified for spiral and diagonal grain.
(c) HARD KNOTS. Sound hard knots up to 3/8 inch in maximum diameter acceptable providing: (a) they are not in projecting portions of I-beams, along the edges of rectangular or beveled unrouted beams or along the edges of flanges of box beams (except in lowly stressed portions); (b) they do not cause grain divergence at the edges of the board or in the flanges of a beam more than specified in column 7; and (c) they are in the center third of the beam and are not closer than 20" to another knot or other defect (pertains to 3/8" knots - smaller knots may be proportionately closer). Knots greater than 1/4 inch should be used with caution.
(d) PIN KNOT CLUSTERS. Small clusters acceptable providing they produce only a small effect on grain direction.
(e) PITCH POCKETS. Acceptable in center portion of a beam providing they are at least 14 inches apart when they lie in the same growth ring and do not exceed 1 1/2 inches length by 1/8 width by 1/8 inch depth and providing they are not along the projecting portions of I-beams, along the edges of rectangular or beveled unrouted beams or along the edges of the flanges of box beams.
(f) MINERAL STREAKS. Acceptable providing careful inspection fails to reveal any decay.
5. DEFECTS NOT PERMITTED.
(a) CROSS GRAIN. Not acceptable unless within limitations noted in 4(a).
(b) WAVY, CURLY, AND INTERLOCKED GRAIN. Not acceptable unless within limitations noted in 4(b).
(c) HARD KNOTS. Not acceptable unless within limitations noted in 4(c).
(d) PIN KNOT CLUSTERS. Not acceptable if they produce large effect on grain direction.
(e) SPIKE KNOTS. These are knots running completely through the depth of a beam perpendicular to the annual rings and appear most frequently in quarter-sawed lumber. Wood containing this defect should be rejected.
(f) PITCH POCKETS. Not acceptable unless within limitations noted in 4(e).
(g) MINERAL STREAKS. Not acceptable if accompanied by decay (see 4(f)).
(h) CHECKS, SHAKES, AND SPLITS. Checks are longitudinal cracks extending, in general, across the annual rings. Shakes are longitudinal cracks usually between two annual rings. Splits are longitudinal cracks induced by artificially induced stress. Wood containing these defects should be rejected.
(i) COMPRESSION WOOD. This defect is very detrimental to strength and is difficult to readily recognize. It is characterized by high specific gravity, has the appearance of an excessive growth of summer wood, and in most species shows but little contrast in color between spring wood and summer wood. In doubtful cases the material should be rejected or samples should be subjected to a toughness machine test to establish the quality of the wood. All material containing compression wood should be rejected.
(j) COMPRESSION FAILURES. This defect is caused from the wood being overstressed in compression due to natural forces during the growth of the tree, felling trees on rough or irregular ground or rough handling of logs or lumber. Compression failures are characterized by a buckling of the fibers that appears as streaks on the surface of the piece substantially at right angles to the grain, and vary from pronounced failures to very fine hair lines that require close inspection to detect. Wood containing obvious failures should be rejected. In doubtful cases the wood should be rejected or further inspections in the form of microscopic examination or toughness tests made, the latter means being the more reliable.
(k) DECAY. All stains and discolorations should be examined carefully to determine whether or not they are harmless stains or preliminary or advanced decay. All pieces should be free from rot, dote, red heart, purple heart and all other forms of decay.

.4011 TORCH WELDING

1. Acceptable practices and further references are discussed in Appendix IV herein.

.4012 ELECTRIC WELDING

1. When arc welding is used the information needed for approval may be met by specifications or reports covering the following;

a. The type of equipment to be used and the proposed scope of application of the process.

b. The proposed minimum requirements established for welders, covering qualifying tests, experience, etc. Reference to Air Corps Specification 20013-A "Welding Procedure for Certification of Welders", if this specification is used, is sufficient in this connection.

c. General procedure covering polarity, arc length, allowable voltage variation, electrode type and material, and identification of each welder's work.

d. Detail procedure for each combination of metals covering size and material of electrode, amperage and voltage for various gages of material.

e. The method of control including test and inspection procedure, etc. In this connection, sketches of the proposed standard test samples, a sample test report sheet, and a statement concerning the frequency of sample tests, should be submitted. Use of the Specification noted in b above is considered sufficient in this connection.

f. Drawings of parts to be welded.

2. When spot and/or seam welding are employed the information required for approval is similar to that required for the approval of arc welding, except that greater importance is attached to the equipment control means and the detail design of the pertinent joint than to the requirements for welders.

3. When the experience of a manufacturer and the reliability of the product has been demonstrated by him to be satisfactory, a blanket approval may be granted for his use of the process, i.e., he need not obtain approval of each subsequent specific application.

4014 PROTECTION

1. Paints, varnish, plating and other coatings should be adequate for the most severe service expected. Information on the subject of protection is available from paint and varnish manufacturers as well as from metal and alloy producers. Reference may also be made to Appendix IV herein. Expensive changes dictated by service experience will be avoided if the question of protection is considered in the initial design stages. In addition to surface protection it is essential that moisture-trapping pockets and closed non-ventilated compartments be avoided. This is particularly true with light alloy and plywood structures. Drain holes should be provided at low points.

2. Two methods of specifying protective coatings are in general use. In one the various operations or code symbols therefor are listed on the pertinent detail or assembly while in the other method a specification listing the operation and the numbers or classes of the parts to be so treated is prepared. The latter is more flexible when various agencies are being dealt with. Data submitted to the Administrator need cover only the minimum protection to be employed.

.4015 INSPECTION

1. Points most frequently in need of inspection are main fittings, control linkages, cables at pulleys and at fairleads and all moving parts and locations where wear is likely to occur or where lubrication is required. This includes all points where bolts or pins are installed as bearing surfaces which are subject to any movement and wear. Satisfactory inspection and servicing of these and other points can only be carried out if the size and location of inspection openings are such as to give adequate accessibility. Particular attention should be given to providing openings making it possible to inspect for rust or corrosion where dust, sand or moisture is likely to accumulate. Careful attention should be given to the tail section of the fuselage in this regard.

.402 JOINTS, FITTINGS AND CONNECTING PARTS

1. These parts continue to be the most critical structural elements. No specific rules can be laid down but some of the more important considerations follow. The type of fitting is mainly dependent on the magnitude of the loads involved and the nature of the parts being connected. The material should be chosen after consideration of such factors as corrosion, fatigue, bulk, weight and production ease. It should be possible to inspect, service and replace each vital fitting. Points sometimes over-looked in the detail design of fittings include:

a. Stress concentration, either from section changes or from welding or heat treating effects.
b. Adequate allowance for flexibility of parts being joinad.
c. Specifying proper surface condition, i.e., a rough turning job on a highly stressed part invites cracking and failure.

2. In the design of fittings at the end of wood spars there is a tendency to crowd bolts too close to the spar end in order to secure a more compact fitting. This sometimes results in a shear failure of the wood along the grain, even though the design load in the tension direction is samll. To reduce the possibility of such failures bolt spacings and end margins should be in accordance with Fig. 2-4 of ANC-5.

3. In using extruded sections it should be borne in mind that the nature of the extruding operation produces in effect a longitudinal grain structure. Fittings therefore should be designed to avoid critical "cross-grain" loading.

4. Fitting drawings should include tolerances for dimensions of critical sections, such as lugs, in order to maintain the required strength properties.

5. Some examples and discussion of good and bad fitting practice are given in Appendix IV herein.

Fig. 31 SUGGESTED CASTING PRACTICE
Ref. CAM 04.4023

ALLOY	MINIMUM FILLET RADIUS (3)	MINIMUM SECTION (3) (Webs, etc.)	MAXIMUM RATIO OF ADJACENT SECTIONS (4)	REMARKS (1)(2)
ALUMINUM - Alcoa 12, 43, etc., and equivalent	1/8"	1/8"		Used where strength is not primary consideration. Alcoa 12 (SAE No.33) should not be used where subject to shock or impact, due to its low elongation (2%). Alcoa 43 (SAE No.35) and 356 alloys which have high silicon content are used where leak-proof or complicated castings are required.
ALUMINUM - (High-Strength) Alcoa 195, 220 etc., and equivalent	3/16"	5/32 - 3/16" (1/8" if structurally unimportant)	3:1	Most aluminum alloy structural castings are made of the 195 or equivalent material. The 220 alloy is superior for shock and impact loading but castings should be simple due to the difficulty in securing satisfactory complex castings.
BRASS, BRONZE	1/8"	1/8"		Red brass such as SAE No. 40 or Federal Specification QQ-B-691, grade 2, is used in fuel and oil line fittings. Phosphor Bronze (SAE No.64 and No.65 or Federal Specification QQ-B-691 grade 6) is used for anti-friction installations such as bushings, nuts, gears and worm wheels. Manganese and aluminum bronzes (SAE No.43 and No.68 or Federal Specifications QQ-B-726 and QQ-B-691) are used where maximum strength and hardness are desired.
MAGNESIUM	1/8" (50% greater than aluminum preferred)	5/32"		Not recommended for use at elevated temperatures (limit approximately 400°F) or in exposed locations on seaplanes. Particular care should be observed in protecting against corrosion and electrolytic action.
STEEL	1/4" (1/2" preferred)	1/4"	5:2	Used primarily for heavily loaded parts such as in landing gear of large aircraft. Alloys used include chrome-molybdenum, nickel and manganese. When using high ultimate tensile-strengths the effect of the corresponding low elongation should be considered.

(1) For allowable stresses see ANC-5 "Strength of Aircraft Elements".
(2) For additional factor of safety see CAR 04, Table 04-7. When using this factor the 50% stress reduction noted in ANC-5 may be disregarded.
(3) Larger values should be used where possible.
(4) Highly dependent on other factors.

FIG. 31 SUGGESTED CASTING PRACTICE

4020 BOLTS, PINS AND SCREWS

1. Approved locking devices include cotter pins, safety wire, peening, and, with certain restrictions, Elastic Stop nuts and Dardelet Threaded parts.

2. Restrictions on the use of Elastic Stop Nuts are as follows:

a. Two types of nuts, i.e. (1) fiber insert and (2) metallic insert (Hytemp), are approved subject to the following general restrictions:

(1) They should be made to conform to Army or Navy material specifications.

(2) They should not be used at joints which subject the bolt or nut to rotation

(3) Bolts must be of such length that completely formed thread extends through the nut.

(4) They should be called out on the pertinent drawings submitted to the Administrator.

(5) Nuts of the fiber insert type should not be used where subject to temperatures in excess of 250°F.

(6) Nuts of the metallic type (Hytemp) should not be used for primary structural connections in the following specific applications:

(a) They should not be used to attach wing panels, fins and stabilizers.

(b) They should not be used in the control system, including surfaces, hinges and bracket attachments thereof.

(c) They should not be used to attach exhaust manifolds and similar items where the temperature may exceed 600°F.

3. Restrictions on the use of Dardelet Threaded parts follow:

a. The parts must be manufactured by a licensee of Dardelet Threadlock Corporation under the terms of its license agreement. (Note this covers manufacturing considerations peculiar to this design.)

b. They should be made to conform to Army or Navy material specifications.

c. They should not be used at joints which subject the bolt or nut to rotation.

d. Bolts must be of such length that completely formed thread extends through the nut.

e. They should be called out on the pertinent drawings submitted to the Administration.

.4023 CASTINGS

1. Castings should be obtained from a reliable source with experience on similar type castings. Such castings should incorporate generous fillet radii, ample draft, and gradual changes of section. Sound castings can only be secured by proper consideration of and allowance for the flow of molten metal in the mold. Casting drawings should be "load marked", i.e., the direction and approximate magnitude of the design loads should be shown. It is then possible for the foundry to cast the densest and soundest metal at the critical sections. Finished surfaces should end in radii at inside corners to prevent stress concentration. Some of the more important design and drafting considerations are given in Fig.31. It should be emphasized that these are not given as requirements but merely as values and points found acceptable in general practice. Reference should be made to trade literature of the various metal and alloy producers for additional information.

2. As with other aircraft parts, the acceptance of castings for primary structure is predicated upon thorough and adequate inspection. It is customary to test and section or to X-ray the first castings of a new part in order to be certain of good design and satisfactory foundry technique. Production runs may be inspected visually in conjunction with occasional tests for verification. Hardness testing of the casting and physicaltests of test coupons cast with the part are also used. Steel castings with smooth surfaces may be inspected by magnafluxing. X-raying provides an excellent means of thoroughly inspecting castings if the results are properly interpreted, i.e., by an expert.

.403 TIE RODS AND WIRES

1. When unswaged threaded-end tie rods are used, particular attention must be paid to the end connections to insure proper alignment. The wires should be so carried through sleeves or fittings that any bending is limited to the unthreaded portion of the rod. Where this is not done, even small bending stresses may soon cause fatigue failure at the thread roots. High margins should be incorporated since practically all working from tension loads, with attendant stress concentration, will occur in the threaded portion. Swaged tie rods are considered much more satisfactory and may be no more costly in quantities. A satisfactory locking means should be used. Check nuts have been found acceptable for this purpose, when employed on terminals not subjected to appreciable vibration.

.404 FLUTTER PREVENTION MEASURES

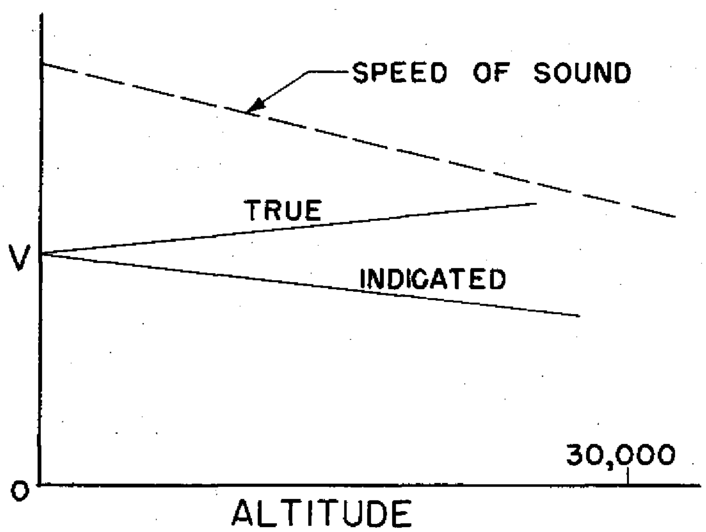
A GENERAL

1. Flutter is a violent self-induced vibration of a body resulting from a coupling of aerodynamic, elastic and inertia forces acting upon the body. For detailed information on flutter theory and its application, reference should be made to one or more of the following:

Theodoresen and Garrick - "Mechanism of Flutter" NACA TR 685
Kassner-Fingado - "The Two Dimensional Problem of Wing Vibration" Translation - Journal Royal Aero Society, October 1937
Küssner - "Status of Wing Flutter" Translation - NACA TM 782, January 1936
Lombard - "Conditions For The Occurrence of Flutter" California Institute of Technology Thesis (1939)

Reference to other work will be found in the bibliographies contained in the above. The study of flutter is passing through a period of rapid development and it appears that a better and more accurate understanding of the inter-relation of rigidity, mass properties and frequencies and their effect upon flutter will soon emerge.

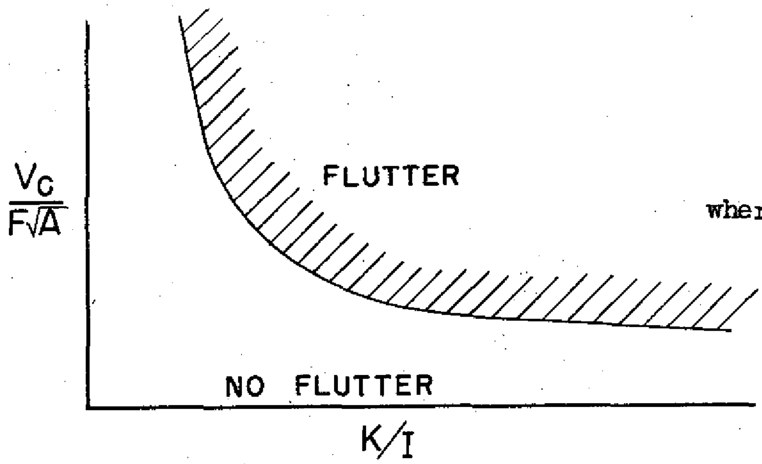
2. Flutter theory is in general based upon true velocity and sea level atmospheric conditions. For this as well as other reasons an ample margin, particularly on high performance aircraft, should be maintained between the computed flutter speed and the actual dive speed attained in testing. The trend of critical flutter speed with altitude may be expressed thus:



but values for specific cases will be dependent upon wing weight and other factors.

3. For wings, the use of the fundamental theory of Theodorsen has been simplified for certain cases by the work of Bergen and Arnold (given at the Institute of the Aeronautical Sciences meeting at Los Angeles in June 1940) who treat the special case of wing bending-aileron by a graphical method, and Wylie (unpublished). In addition, for civil aircraft, conventional size and performance, the use of a suitable wing torsional rigidity criterion, together with proper observance of other measures, provides adequate insurance against flutter.

4. For control surfaces, recent unpublished Air Corps studies (Dent and Smilg) indicate that a relationship:



where V_C = Flutter Speed

F = Freq. Fixed Surface

A = Area Mov. Surface

K/I = Balance Coefficient of Movable Surface

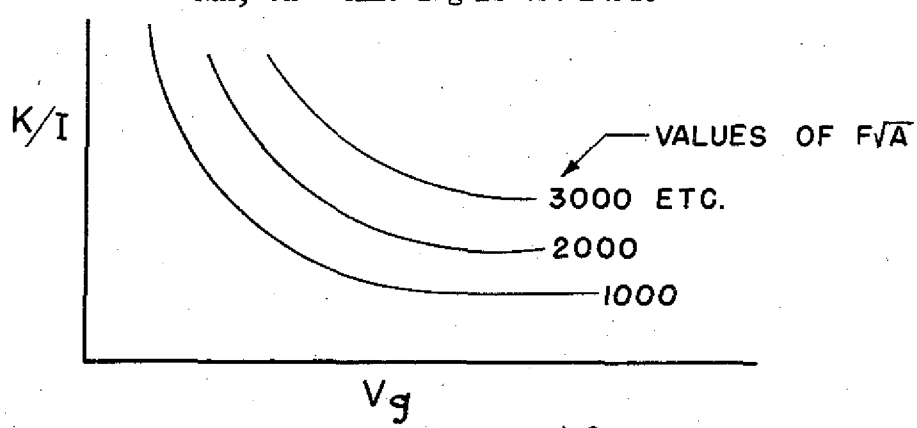
holds considerable promise, and they have tentatively established a number of check points on the curve. The basic similarity of the above curve to the usual Küssner formula of:

$$V_C = \frac{F\,C}{K},$$

where
F = Critical Frequency
C = Wing Chord
K = Reduced Frequency Coefficient (Dependent on particular characteristics of the airplane)

is apparent, since both show an increase in flutter speed with an increase in frequency or size of the airplane, other factors remaining constant (both C and $\sqrt{A}$ being dimensionally length units).

5. By replotting the Air Corps data on the scales used in Fig. 32A, the following is obtained:



If the results of further study and experience warrant, suitable modifications to the K/I - Vg relation shown on Figures 32A and 32B will be made.

6. In general, the various limiting values given hereafter to rigidity, mass balance and frequency ratio, and the practices on detail design should be closely observed. As indicated above, however, rapid progress is being made toward a better understanding of the problem, and for new aircraft, therefore, it is recommended that an outline of the proposed flutter prevention measures be submitted to the Administration for examination and comment as early in the development as possible. The Flutter Control Data Form No. ACA-719 (Table Vb) used in the final vibration testing was evolved for the dual purpose of simplifying submittal of data and of facilitating study by the National Advisory Committee for Aeronautics and other interested government agencies through a more rapid collection of information.

B RIGIDITY

1. This factor is of first importance, since coupling (and consequently flutter) can only occur through deflection. However, increases in aircraft size, a trend toward thinner airfoils, prevalence of discontinuities and cutouts, and weight limitations make necessary the establishment of minimum acceptable rigidities. Rigidity may be represented in terms of frequencies, and often is in flutter theory.

2. Wings. The torsional rigidity of wings is highly important. This should be investigated by means of a wing torsion test (CAM 04.31E) unless other adequate data are submitted. Figure 32, a development of an earlier curve of the same number, but based upon considerably more information including certain Navy data, indicates values of C_{TR} which have been found satisfactory for conventional designs. If the test is conducted on a fabric covered wing with taut fabric, an allowance of approximately 10% (dependent somewhat on the size of the wing) should be made for the effect of fabric aging. Since the actual torsional deflection of the wing will depend upon the moment coefficient of the airfoil employed, it is advisable to introduce the additional criterion that the maximum torsional deflection under the limit load critical for torsion not exceed 3°.

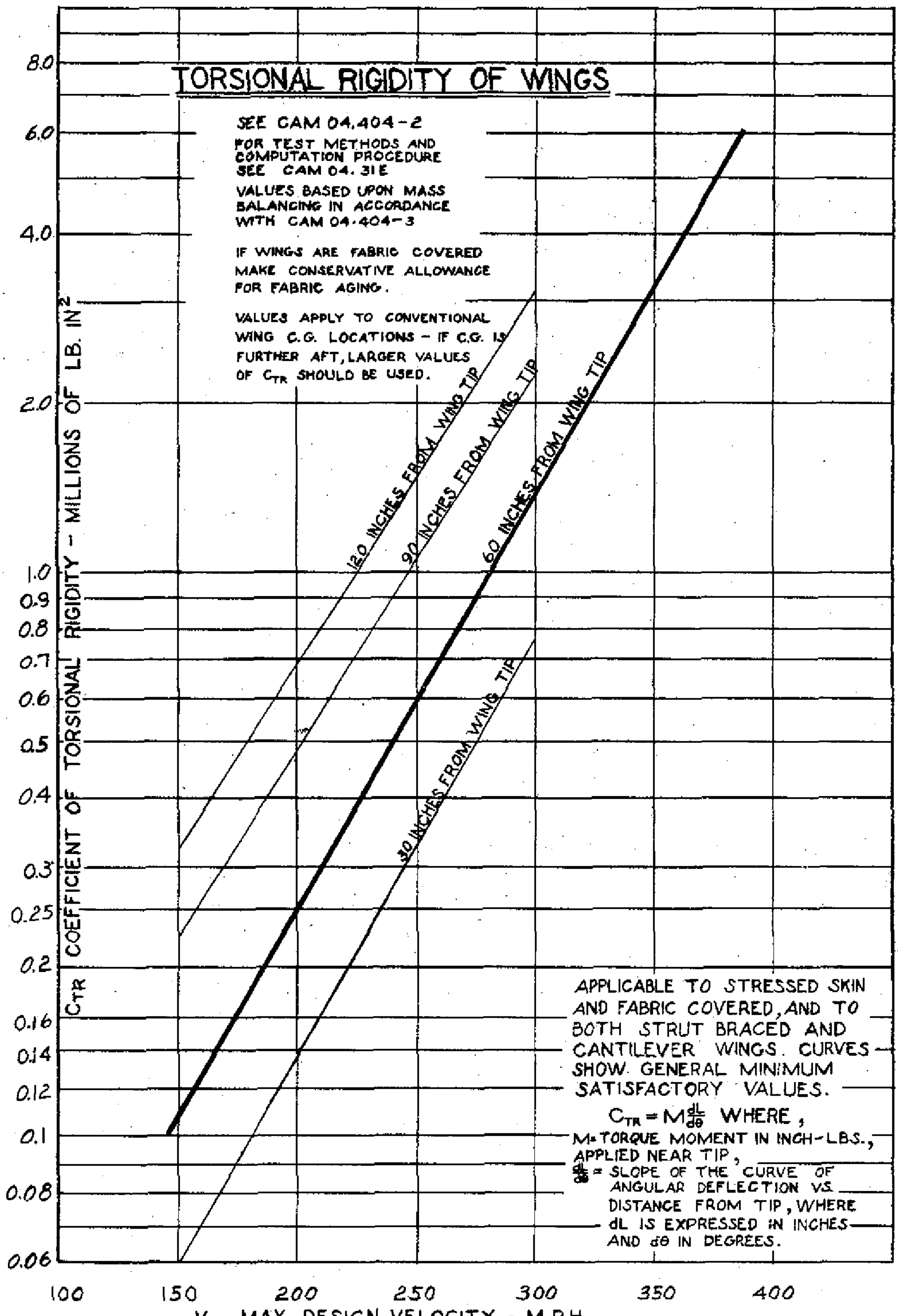


FIG. 32

REVISED 9-40

C MASS BALANCING

1. For methods of computing static and dynamic balance values see CAM 04.424. (The weight and static hinge moment, or c.g. location, of finished movable surfaces should be checked to insure that the computed values are not unconservative.) See paragraph F DETAIL DESIGN below, for notes on the installation of balance weights. Note that the specified dynamic balance coefficient values may in some cases be influenced by the frequency ratio (see Fig. 32C).

2. Compliance with the following dynamic balance coefficients and static balance conditions should be shown unless other equally effective steps to prevent control surface flutter are shown to have been taken:

a. Ailerons. When Vg is in excess of 150 mph the dynamic balance coefficient as computed about the aileron hinge axis and the longitudinal axis of the airplane should not be greater than the following value

$$K/I = 1.6\ (3 - Vg/100) \quad \text{(See Fig. 32A)}$$

except that it need not be less than zero. Ailerons on internally braced wings, or on airplanes with Vg in excess of 200 mph should be completely statically balanced about their hinge line when in the neutral position. Special consideration will be given to lesser static and dynamic balance when the aileron control system is substantially irreversible.

b. Rudders. When Vg is in excess of 150 mph, the dynamic balance coefficient of the rudder(s), as computed about the rudder hinge axis and the axis of torsional vibration of the fuselage, should not be greater than the value given in paragraph a. above, except that it need not be less than zero. When rudders are not in the plane of symmetry they should be completely statically balanced (zero unbalance).

c. Elevators. When Vg is in excess of 150 mph, the dynamic balance coefficient of each separate elevator (for each half of a continuous elevator), as computed about the elevator hinge axis and the centerline of the intersection of the stabilizer and the plane of symmetry, should not be greater than the following value

$$K/I = 3.0 - Vg/250 \quad \text{(See Fig. 32B)}$$

When the rudder(s) has (have) complete dynamic balance about a conservatively chosen axis, a special ruling may

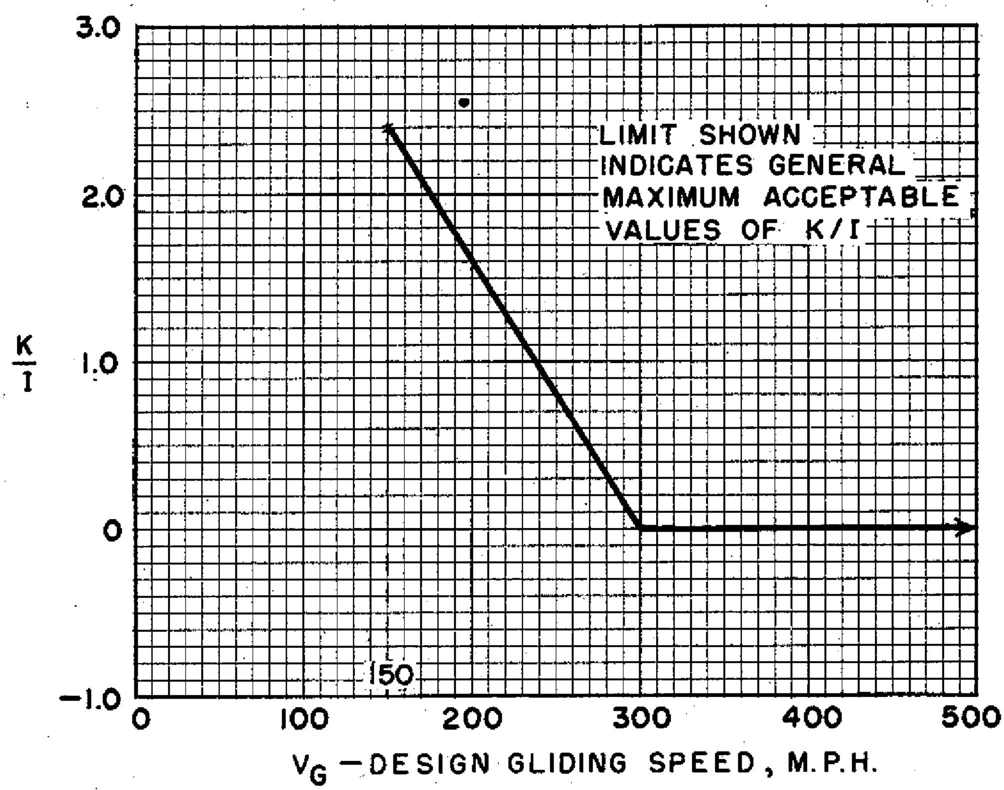


FIG.32A AILERON & RUDDER DYNAMIC BALANCE

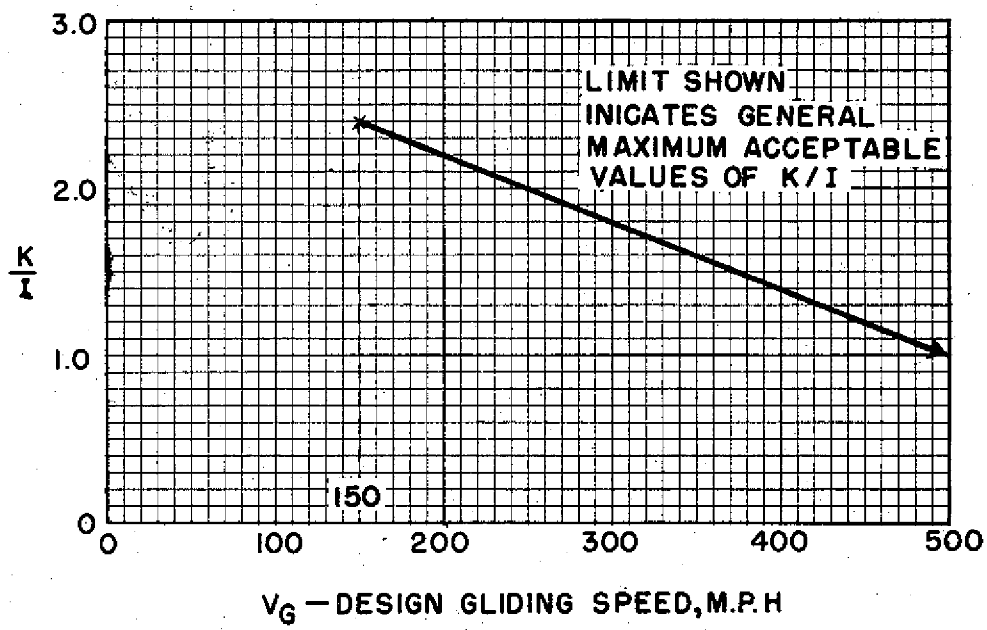


FIG.32B ELEVATOR DYNAMIC BALANCE

be obtained from the Administration regarding the elevator dynamic balance if the coefficient is greater than above specified. This ruling will be dependent on the general design of the entire tail unit.

d. Tabs. Trim and balancing tabs should be statically balanced about their hinge axes unless an irreversible non-flexible tab control system is used. The balancing of control tabs will depend on the particular installation involved and special rulings should be obtained from the Administration in such cases.

D FREQUENCY RATIO

1. In accordance with general practice, in this discussion frequency ratio is defined as the frequency of the movable surface divided by the frequency of the fixed surface (or other element to which the movable surface is attached), and for a single airfoil as the frequency in bending divided by the frequency in torsion, thus:

$$\frac{F\text{ movable surface}}{F\text{ fixed surface}}\text{, and}$$

$$\frac{F\text{ bending}}{F\text{ torsion}}$$

Tests conducted by the Air Corps with flutter models have indicated that, when the frequency ratio involving a movable surface is greater than 1.0 the possibility of flutter in this mode is much reduced as compared with frequency ratios less than 1.0. A study of previous vibration test data and the service records of the aircraft tested, together with Air Corps test data, have indicated the desirability of using the frequency ratio as an additional limitation on the curves of the dynamic balance coefficient (K/I) versus design gliding speed (Vg), as shown in Fig. 32A for the aileron and rudder and in Fig. 32B for the elevator.

2. This limitation is expressed as a curve in Fig. 32C. The shaded portion marked "Approval dependent upon special considerations" is considered an undesirable range and approval is subject to consideration of the modes involved, actual values of frequencies, speed of the aircraft, value of K/I, etc. For this reason it is advantageous to submit to the Administration as early as possible in the development of a new model an outline of the proposed flutter prevention measures.

301230 O—41——10

n. Balance Weight and Arm Assembly
Surface mode (see below)

Note: The critical balance weight and arm frequency will usually be bending in a plane normal to the hinge line of the surface. The surface mode would be the one likely to couple with the above—such as movable surface (sym.). It is desirable to have the above frequency ratio substantially greater than 1.0. However, the balance weight arm bending frequency should also be checked in a plane parallel to the hinge line of the surface. This may be critical for the elevator balance weight in a fuselage side bending mode, etc.

The following special cases should also be considered for large aircraft:

o. Aileron (Unsym)
Wing Bending (Unsym)

p. Aileron (Unsym)
Wing Torsion (Unsym)

q. Wing Bending (Unsym)
Wing Torsion (Unsym)

r.* Rudder (Torsion)
Fin Rocking about Stabilizer Attachments

Note: This would apply to outboard vertical surfaces disposed both above and below the stabilizer and is somewhat analogous to an elevator unsymmetrical—fuselage torsion case.

s.* Stabilizer Torsion
Fuselage Torsion

* Only for rudders not in the plane of symmetry.

In general the natural frequency of a tab having an irreversible control mechanism should not be less than 1500 c.p.m. In the case of a servo tab with a frequency ratio less than 1.2, complete dynamic balance should be had; i.e., $K/I = 0$.

E DETAIL DESIGN

1. Service troubles and accident records reveal that particular attention should be paid to items such as the following:

a. The trailing edges of movable surfaces should be rugged to reduce the possibility of additional weight being added during field repairs with an adverse effect on the mass balance characteristics.
b. Tab mechanisms should be simple and rugged to avoid improper assembly, or the possibility of play developing due, for example, to open end (i.e., magneto type) ball thrust bearings being inserted backwards.
c. Provide adequate "carry through" structure to insure rigidity.

2. A rugged aileron hinge bracket is of little merit unless the rear spar to which it attaches is well restrained against "rolling". Likewise rugged cabane members with good angular relations will fail to restrain the wing cellule if anchored into eccentric apex joints.

3. The interconnection between elevators should be rigid and rugged, in order to maintain a satisfactory margin of safety against an elevator unsymmetrical (torsion) mode of vibration. Butt welded joints should preferably be reinforced with gussets.

4. The general principles of flutter prevention should be observed on all airplanes. This applies particularly to the design and installation of control surfaces and control systems and includes such desirable features as structural stiffness, reduction of play in hinges and control system joints, rigid interconnections between ailerons and between elevators, complete static and dynamic balance of control surfaces and high damping. For fixed surfaces, such as wings, stabilizers, and fins, it is desirable to keep the center of gravity location of the surface as far forward as possible. Features tending to create aerodynamic disturbances, such as sharp leading edges on movable surfaces, should be avoided. These principles apply also to wing flaps and particularly to control surface tabs which are relatively powerful, and correspondingly more dangerous if not properly designed. In the design of control surfaces, dynamic balance should be achieved, as far as practicable, by distributing the structural material in such a way (element by element, spanwise) that a uniform condition of static balance will result without adding large amounts of lead. If possible, the use of large concentrated weights should be avoided, since fatigue failures may result in the supporting arms and attachments. When concentrated weights are used to achieve the required degree of dynamic and static balance, it is a good rule that the number of weights used be at least equal to the number of hinges. The attachment of each weight should be sufficiently strong and rigid that its frequency is above that

of the surface proper. Where weights are riveted into the leading edge of a surface no difficulty should be encountered on this point, but if an arm is used to support the mass, a design factor several times the Condition I load factor may be necessary for the arm and its attachment to the surface.

5. It should be realized that various forms of flutter are possible and that there usually exists for each type of flutter a critical speed at which it will begin. This critical speed will be raised by an improvement in the antiflutter characteristics of the particular portion of the airplane involved and may even be eliminated entirely in some cases.

04.41 Detail Design of Wings.

1. The general considerations of good detail design, which have been previously covered, are of particular importance in connection with wing structures since these structures are involved in carrying some of the heaviest loadings present in the airplane structure. Fittings and joints in wings often represent some of the most critical design problems and they must be carefully proportioned so that they can pick up loads in a gradual and progressive manner and redistribute those loads to other portions of the structure in a similar manner. This requires that special attention be paid to minimizing stress concentrations by avoiding too rapid changes in cross sections, and to providing ample material to handle stress concentrations and shock loadings which cannot be avoided.

"2. The above principles are of special importance in connection with wing carry-through structures which serve to carry landing loads to the fuselage. In such cases allowance should be made for the fact that landing loads are of the impact type. This necessitates the provision of adequate bearing areas for all attachments which carry such loads into the basic structure. Recommendations in this connection are given in ANC-5, Tables 4-2 and 4-3, and in ANC-5 - 5.501.

"3. In addition to the problem of locally introducing these loads into the structure, provision must be made for distributing these loads throughout the structure. Bulkheads for this purpose in stressed-skin wings must have ample rigidity and an allowance must be made for the fact that it requires some distance before these loads are completely distributed into the structure in a manner approaching that indicated by simple beam theory.

"4. The above recommendations apply with equal weight when a landing gear is attached directly to a wooden spar. In some respects they apply even more since experience has shown that wood is much more subject to stress concentration effects than has been appreciated in the past. Due allowance should be made for stress concentrations due to holes through the spar, and when such holes pierce both the spar cap strip and filler block (when such are used) the filler block should extend a sufficient distance away from this part so that it has an ample opportunity to distribute the loads. In addition, filler blocks should be generously tapered in order to avoid rapid changes in the effective cross-section of the spar. In general, it is good practice to distribute the landing loads into wooden spars by means of a number of small bolts than by a few large bolts. When it is difficult to obtain an adequate attachment of the gear to the spar by means of bolts alone (i.e., when the bolt area required would cut-out too much of the spar), it is advisable to incorporate a hardwood block under the spar to assist in distributing the landing gear load."

.4110 WING BEAM JOINTS

1. See CAM 04.402.

.415 FABRIC COVERING

A. COTTON FABRIC, REINFORCING TAPE, SURFACE TAPE AND LACING CORD

1. **CAA grade A fabric.** Cotton fabric used on aircraft with wing loadings of 11 lbs. per sq. ft. or over, or design gliding speeds of 240 m.p.h. or over (See Fig. 32D) should conform with or exceed the following specification. (Note: AN-CCC-C-399-1 exceeds this spec.)

a. The cloth should be made from single ply or 2-ply, combed cotton yarn and should be of plain weave. The yarn should be mercerized under tension or the cloth should be piece mercerized or similarly processed to remove the wax coating from the cotton fibers for the purpose of increasing the dope absorptive capacity of the material.

b. The selvage edges should be flat woven with no greater tension than the body of the cloth.

c. Finishing should consist of washing, framing and calendering. The calendering should be sufficient to lay any nap present and to provide a smooth even surface.

d. The cloth should not contain over 2.5 percent sizing, finishing and other non-fibrous materials and should be chemically neutral.

e. The number of threads per inch should be between 80 and 84 in both warp and filling.

f. The breaking strength should not be less than 80 lbs. per inch in both warp and filling as determined by the strip method of testing. This method is outlined in Federal Specification No. CCC-T-191a and consists briefly of the following:

(1) 5 warp and 5 filling specimens at least 6 inches long and 1.25 inches wide taken from at least one-tenth of the width of the material away from the selvage should be prepared for test by raveling to one inch in width, taking from each side approximately the same number of threads.

(2) The specimen should be placed in the testing machine with the jaws 3 inches apart at the start of the test.

(3) The breaking strength is the average of the results obtained by breaking 5 specimens. If a specimen slips in the jaws, breaks in the jaws, breaks at the edges of the jaws, or breaks prematurely for any other reason attributable to faulty operation, the result may be disregarded, another specimen taken, and the result of this break included in the average.

g. The elongation should not be greater than 13 percent in the warp and 11 percent in the filling under 70 lbs. tension load during the strip test. The percentage should be based on the average of the five specimens.

h. Fabrics meeting the above limitations may vary considerably in doping characteristics. Therefore, the airplane manufacturer should demonstrate that his doping equipment and production procedure and technique will result in adequate dope penetration and adhesion in the case of the particular type of fabric he employs.

2. CAA lightweight fabric. Cotton fabric used on aircraft with wing loadings up to and including 8 lbs. per sq. ft., or design gliding speeds up to and including 150 m.p.h. (see Fig. 32D) and having rib spacings in accordance with Figure 33 should conform with or exceed the following specification.

a. Same as 1a.

b. Same as 1b.

c. Same as 1c.

d. Same as 1d. A de-sizing operation may be necessary to reduce the sizing content to the 2.5 percent value specified.

e. The number of threads per inch should not exceed 110 in both warp and filling.

f. The breaking strength should not be less than 50 lbs. per inch in both warp and filling as determined by the strip method of testing. This test method is described in 1f above.

g. The elongation should not be greater than 13 percent in the warp and 11 percent in the filling under 44 lbs. tension load during the strip test. The percentage should be based on the average of the five specimens.

h. Same as 1h.

Note: Fabric for aircraft having wing loadings and speeds substantially lower than noted above or fabric that does not meet the above specification in all respects but has been proven by extensive satisfactory service experience to be suitable for aircraft use will be subject to special consideration. In no case, however, should the sizing content exceed 2.5 percent.

3. A straight line variation between the physical properties of the fabrics described in paragraphs 1 and 2 should be assumed in selecting fabric for aircraft which have wing loadings or design gliding speeds between those noted in paragraphs 1 and 2. For example, fabric having a breaking strength of 60 lbs. per inch and a thread count of 90 is superior to lightweight fabric by 77% as regards thread count but only by 33% as regards strength. Thus the fabric is suitable for all aircraft having wing loadings of 8 + .33 (11-8) = 9 lbs. per sq. ft., or less, or design gliding speeds of 150 + .33 (240-150) = 180 mph, or less, as shown by the broken line in Fig. 32D.

4. Reinforcing tape used in connection with the fabric described in paragraph 1 should conform with or exceed the following specification:

a. The tape should be made from combed cotton yarn, should be unbleached and should not contain more than 3.5 percent sizing, finishing and other non-fibrous materials.

b. The breaking strength should not be less than indicated in the following table when tested in accordance with Federal Specification No. CCC-T-191a, using the average breaking strength of 5 full-width specimens.

Width - inches	Breaking strength - lbs.
1/4	80
3/8	120
1/2	150
5/8	170
3/4	200
1	250

5. Reinforcing tape used in connection with the fabric described in paragraph 2 should conform with or exceed the following specification:

a. Same as 4a.

b. The breaking strength should not be less than one-half of the values listed in 4b when tested in accordance with Federal Specification No. CCC-T-191a, using the average breaking strength of 5 full-width specimens.

6. A straight line variation between the physical properties of the reinforcing tapes described in paragraphs 4 and 5 should be assumed in selecting reinforcing tape for use in connection with fabrics determined by paragraph 3.

7. Surface tape should have approximately the same properties as the fabric with which it is used and should have pinked, scalloped or straight edges.

8. Lacing cord should be of high quality linen or cotton cord, should have a strength of at least 80 lbs. when tested double, and should be lightly waxed before using.

B. OTHER COVERING MATERIALS.

Fabric, reinforcing tape, surface tape, and lacing cord made from materials other than cotton (or linen in the case of lacing cord) will be subject to special consideration. In addition to showing compliance with the pertinent parts of the specifications listed in A above, it will usually be necessary that a certain amount of satisfactory service experience on experimental aircraft covered with these materials be accumulated before final approval may be granted.

C. TECHNIQUE OF COVERING.

1. Seams. All seams should be plain lap, folded fel or French fel seams (see Fig. 32E) machine stitched except as indicated below. Eight to ten stitches per inch should be used. The row of stitches nearest each folded edge of each seam should be approximately 1/16 inch from the edge of the fold and the rows should be 1/4 to 3/8 inches apart.

a. All seams should be parallel to the line of flight except as noted in b below. Seams should preferably not cover a rib or be placed so that the lacing will be through or over the seam. In the case of tapered wings or control surfaces, the seams should be disposed so as to cross the fewest number of ribs consistent with efficient cutting of the pattern.

b. The only seam extending spanwise of the wing or control surface, whether hand or machine sewn, should be at the trailing edge, except that in the case of tapered wings or control surfaces additional seams may be made in the tapered portion at the leading edge.

2. Covering. Either the envelope method or blanket method of covering is acceptable.

a. The envelope method of covering is accomplished by sewing together widths of fabric cut to specific dimensions and machine sewn to form an envelope which can be drawn over the frame. The trailing and outer edges of the covering should be machine sewn unless the frame is not favorably shaped for such sewing, in which case the fabric should be joined by hand sewing as described below for blanket covering.

b. The blanket method of covering is accomplished by sewing together widths of fabric of sufficient lengths to form a blanket covering for all surfaces of the frame. The trailing and outer edges of the covering should be joined by using a plain overthrow or baseball stitch, except that on airplanes with design gliding speeds of 180 m.p.h. or less the blanket may be lapped at least one inch and doped to the trailing and outer edges of wing and control surface structures and to fuselage structures . In fabricating both the envelope and blanket coverings, the fabric should be cut in sufficiently great lengths to pass completely around the frame, starting from and returning to the trailing edge.

c. Hand sewing or tacking should begin at a point where machine sewing stops and should continue to a point where machine sewing or uncut fabric is reached. Hand sewing should be locked at intervals of six inches, and the seams should be properly finished with a lock stitch and a knot. Where tacks are used, they should be not more than 1.25 inches apart.

d. An adequate number of drain grommets properly located to insure complete drainage and ventilation of the wing or control surface should be installed.

3. Attachment of Fabric to Structure. Fabric is usually attached to the structure by means of lacing cord. Other means of attachment such as self-tapping screws and wire and strip should give comparable support. (In questionable cases, sketches and tests or test proposals should be submitted for rulings by the Administrator.) The following factors should be considered when attaching fabric to the structure by means of lacing cord.

a. Care should be taken to insure that all sharp edges against which the lacing cord may bear are adequately protected by commercial tape, or its equivalent, in order to prevent abrasion of the cord.

b. Reinforcing tape of at least the width of the rib cap strips should be placed under all lacing. The tape under moderate tension should be tacked or otherwise attached at the trailing edge of the ribs, brought completely around the wing or control surface and attached again at the trailing edge. In the case of wide cap strips, two widths of reinforcing tape may be used. In the case of wings or control surfaces with plywood or metal sheet from the nose to the front spar, the reinforcing tape need only extend from the trailing edge to the front spar.

c. Stitch spacing should not exceed those shown in Figure 33 which has been derived from extensive experience.

d. A slip knot for tightening should be used at the first point of lacing. The cord, which should be lightly waxed before using, should then be carried to the next point and secured by a seine knot or other equally satisfactory knot and this process continued until the lacing is complete. The cord should be secured at the finish of the lacing by tacking or by a double or lock knot.

e. All seams, leading edges, trailing edges, outer edges and ribs should be covered with suitable width surface tape.

4. Dope and Other Protective Coatings. The number and type of such coatings are usually based upon such factors as the service expected, degree of finish desired, and cost. In general, the dope manufacturers' recommended doping procedures should be followed. Precaution should be taken not to sand heavily over surface tape and spars in order not to damage the stitching cords and fabric.

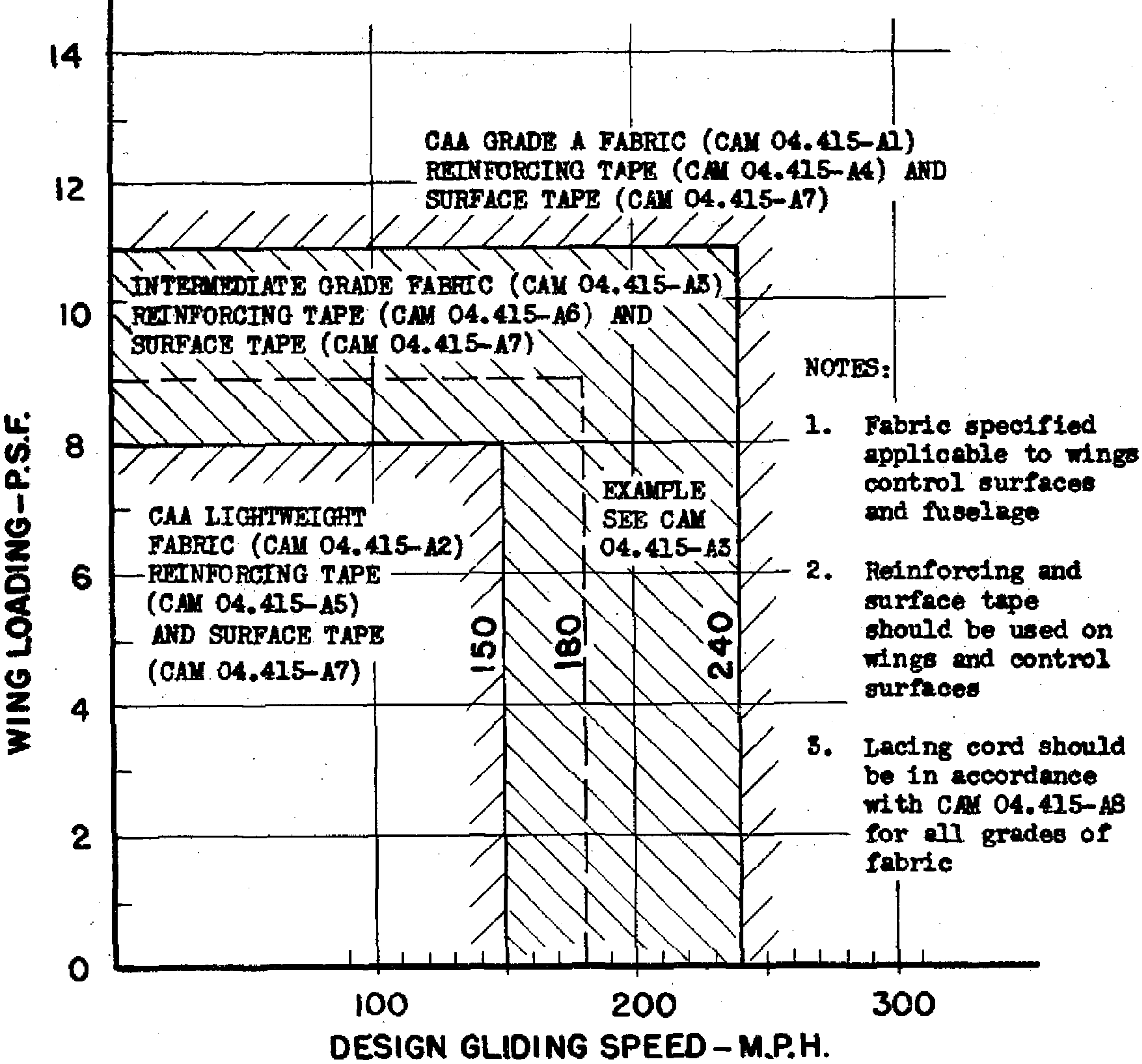


FIG. 32D SELECTION OF FABRIC

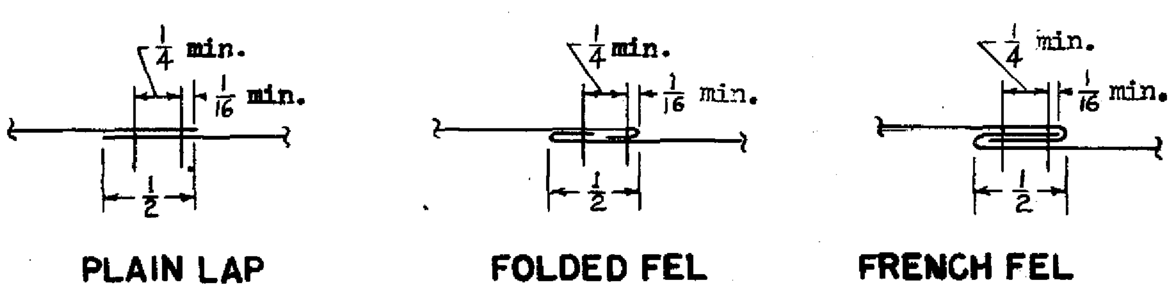


FIG. 32E FABRIC STITCHING SEAMS

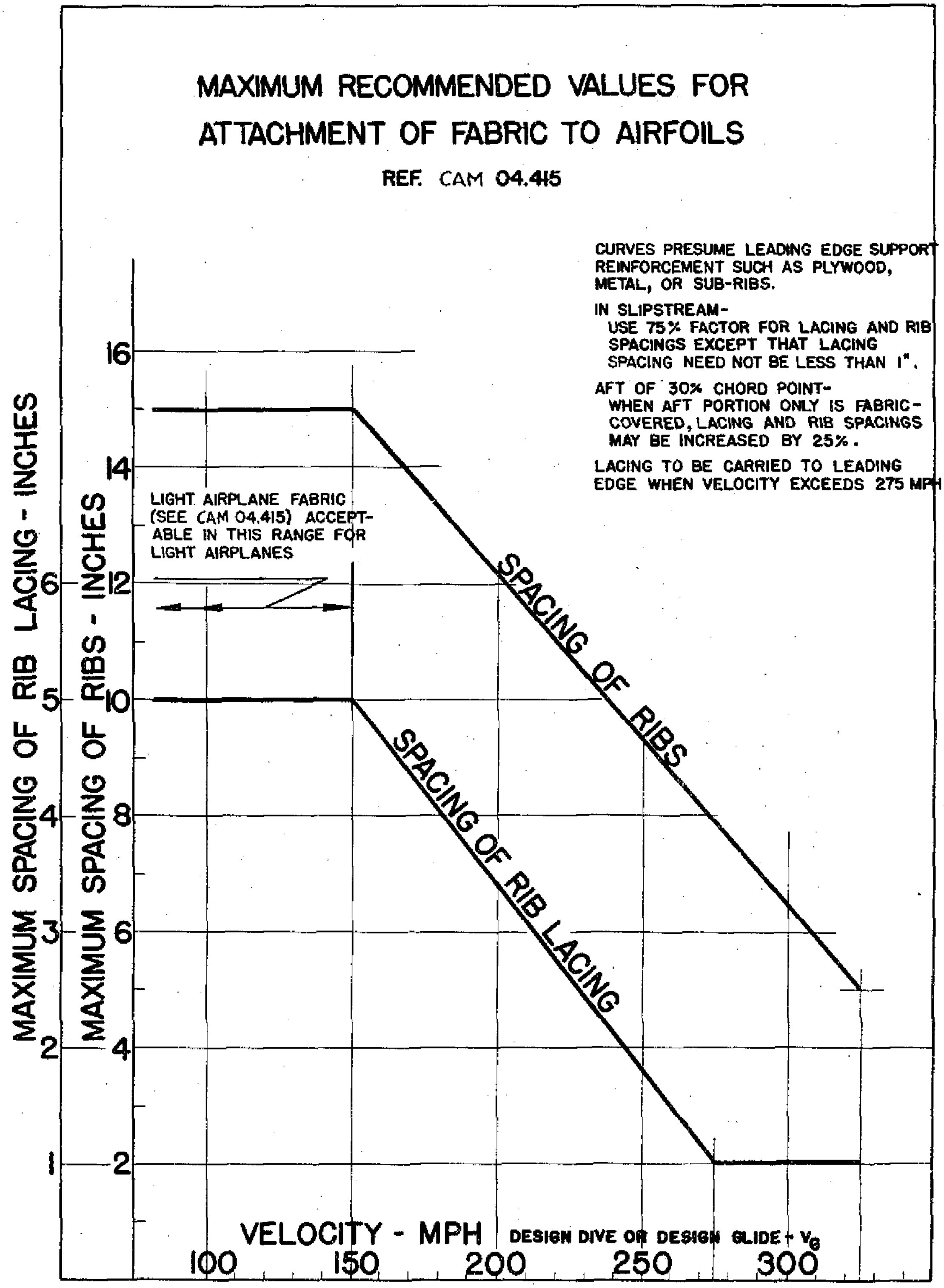


FIG. 33 FABRIC ATTACHMENT

.416 METAL-COVERED WINGS

1. The covering should be sufficiently strong and adequately supported to withstand critical air loads and handling without injury or undesirable deformations. Deflections or deformations at low load factors which may result in fatigue failures should be avoided. In general, skin which shows deformations commonly known as "oil-canning" under static conditions is considered unsatisfactory.

2. In an attempt to establish an empirical method of predicting panel sizes which will be free from unsatisfactory "oil-canning", Fig. 34 has been included as a proposal. In this case the skin thickness and unsupported panel width have been considered the main variables. Other important variables include stress (if appreciable) carried by skin, airspeed, wing loading, and workmanship. Comments and data on this subject are solicited.

.42 DETAIL DESIGN OF TAIL AND CONTROL SURFACES

1. It is very important that control surfaces have sufficient torsional rigidity. No specific limits of permissible maximum deflection of the surface alone are offered, since these may vary widely with the type, size and construction of the surface. However, the behavior of the surface during proof tests should be closely observed. In addition the effect of the control system "stretch" on the total surface deflection under limit maneuvering loads should be considered from the standpoint of "surface usefulness", as described in CAM 04.43-11.

2. Clearances, both linear and angular, should be sufficient to prevent jamming due to deflections or to wedging by foreign objects. It is common practice in the design stage to incorporate an angular clearance of 5 degrees beyond the full travel limit. Surfaces and their bracing should have sufficient ground clearance to avoid damage in operation.

3. External wire bracing on tails is subject to vibration and the design of the wire assembly and end connections should be such as to withstand this condition. Swaged tie rods are recommended, except that for use on light aircraft unswaged rod is acceptable if the points covered in CAM 04.403 are followed. Leading edges and struts should have adequate strength to withstand handling loads if handles or grips are not provided.

4. Direct welding of control horns to torque tubes (without the use of a sleeve) should be done only when a large excess of strength is indicated.

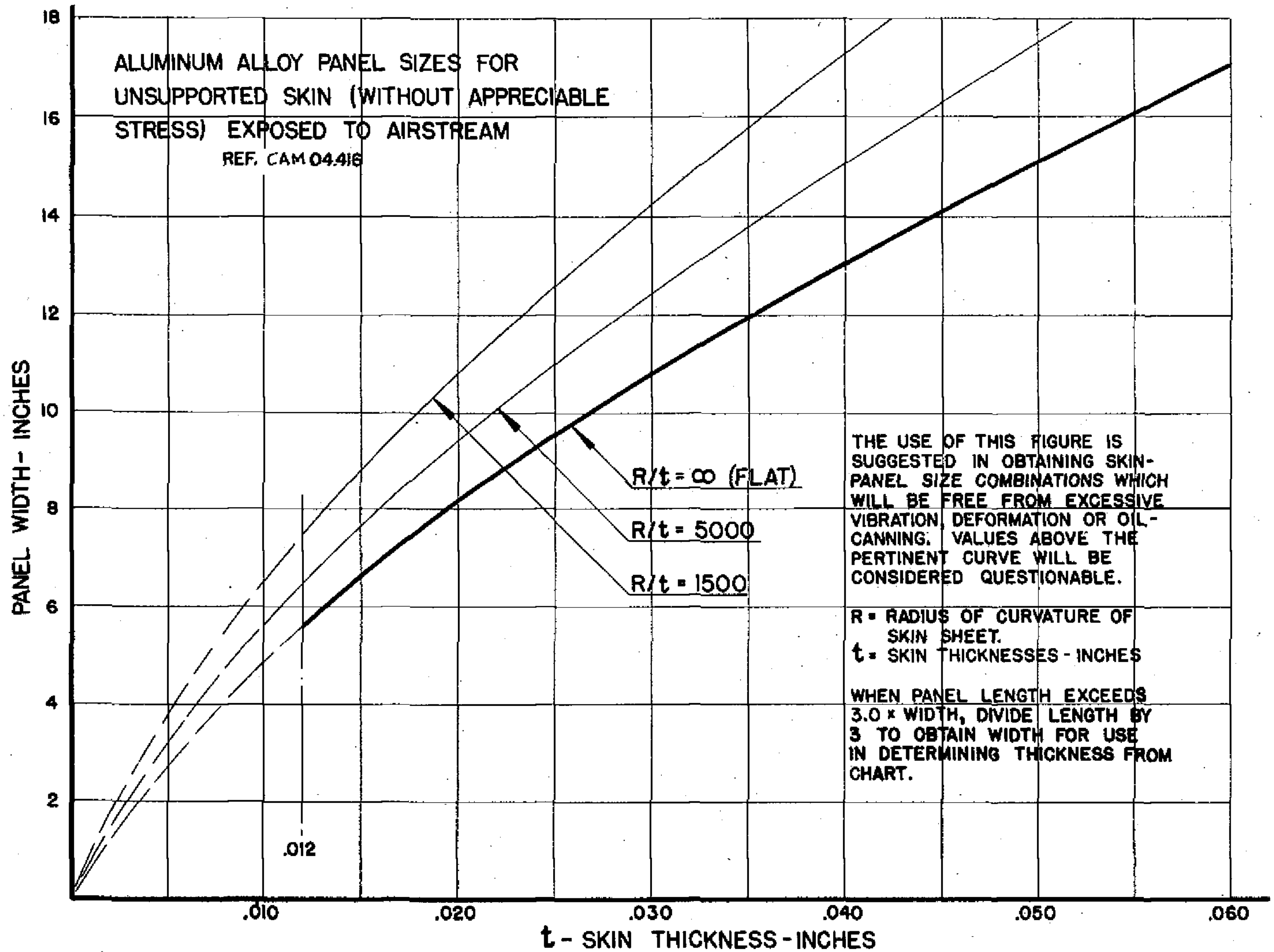


FIG. 34 ALUMINUM ALLOY PANEL SIZES

.421 STOPS

1. Stops are specifically required in the case of adjustable stabilizers and elevator trailing edge tab systems (CAR 04.421 and 04.4210). Some form of stop should, however, be employed at all surfaces in order to avoid interferences and possible damage to the parts concerned and to limit the travel to the approved range. (See also CAM 04.431.)

422 HINGES

1. The following points have been found of importance in connection with hinges:

a. Provision for lubrication should be made if self-lubricated or sealed bearings are not used.
b. The effects of deflection of the surfaces, such as in bending, should be allowed for, particularly with respect to misalignment of the hinges. This may also influence spacing of the hinges.
c. Sufficient restraint should be provided in one or more brackets to withstand forces parallel to the hingecenterline. Rudders, for instance, may be subjected to high vertical accelerations in ground operation.
d. Hinges welded to elevator torque tubes or similar components may prove difficult to align unless kept reasonably short and welded in place in accurate jigs.
e. Piano type hinges are acceptable with certain restrictions. In general only the "closed" type should be used, i.e., the hinge leaf should fold back under the attachment means. The attachment should be made with some means other than wood screws, and this attachment should be as close as possible to the hinge line to reduce flexibility. Piano hinges should not be used at points of high loading, such as opposite control horns, unless the reaction is satisfactorily distributed. Due to the difficulty in inspecting or replacing a worn hinge wire, it is better to use several short lengths than one long hinge.

423 ELEVATORS

1. When dihedral is incorporated in the horizontal tail the universal connection between the elevator sections should be rugged to conform with CAR 04.423.

.424 DYNAMIC AND STATIC BALANCE OF CONTROL SURFACES

1. 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 as though rigidly connected with, and a part of, the fixed surface to which it is attached. As the types of flutter likely to be encountered in aircraft structures involve 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 subjected.

2. Static Balance. Complete static balance of a movable control surface is obtained when the CG of the movable structure is located on the hinge line; i.e., zero unbalance hinge moment, or in a plane through the hinge line and normal to the median plane of the surface. The following points should be noted in connection with statically balanced surfaces:

a. When a surface is in complete static balance the numerical value of the product of inertia (K) is the same for any set of parallel oscillation axes. However, the sign of the product of inertia (K) will depend on the location of the oscillation axis with respect to the center of pressure (CP) of the surface.
b. It should be noted that when each section of a surface perpendicular to its hinge axis is statically balanced, the surface will be in complete dynamic balance for oscillation about any axis perpendicular to the hinge axis; i.e., $K/I = 0$.
c. When the surface is statically balanced it will have some dynamic unbalance with respect to oscillations about an axis parallel to the hinge axis; i.e., $K/I = 1.0$.

3. Balance Coefficients. The dynamic balance coefficient, K/I, is a measure of the dynamic balance condition of a control surface. A zero coefficient corresponds to complete dynamic balance for any given set of axes; i.e., perpendicular, parallel, or at an angle to each other. Positive and negative coefficients correspond to dynamic unbalance or over-balance, respectively. This coefficient is non-dimensional and consists of a fraction whose numerator is the resultant weight product of inertia of the control surface including balance weights (about the hinge

and oscillation axes) and whose denominator is equal to the weight moment of inertia of the control surface (including balance weights) about the hinge axis. The coefficient K/I may be said to represent:

$$\frac{\text{Exciting Torque}}{\text{Resisting Torque}}$$

and is therefore more rational than the coefficient C_B which is:

$$\frac{\text{Exciting Torque}}{\text{Weight x Area}}$$

Both are non-dimensional and will yield comparable results for conventional surfaces, but only K/I may be considered to properly apply to other surfaces. It should be pointed out, however, that when K/I is used, variations with different aspect ratio of the control surface may arise, particularly for the perpendicular axes case. This does not occur with C_B.

4. Product of Inertia with Respect to Two Axes that are Mutually Perpendicular. In computing the dynamic balance coefficient, K/I, of a control surface for axes that are mutually perpendicular (within 15°), the following procedure may be used: Referring to Figure 35:

a. Assume X-axis coincident with the assumed (or known) oscillation axis. Positive direction from the Y-axis is aft of control surface hinge axis, and negative forward of hinge.
b. Assume Y-axis coincident with the control surface hinge axis. Positive direction from X-axis is taken on the same side of the X-axis as is the center of pressure (CP) of the maneuvering load on the surface (see CAR 04, Figures 04-5 and 04-7 for the maneuvering load distribution). It should be noted that it is unnecessary to compute the position of the CP for these purposes, if the side of the X-axis on which it lies may obviously be determined by inspection.
c. After the reference axes have been established, the surface should be divided into relatively small parts and the weight of each such part (w) and the perpendicular distance from its CG to each axis (x to Y-Y axis and y to X-X axis) should be determined and tabulated. (see typical table, figure 35A). The weights and distances should be accurately determined. The weights and CG locations of doped fabric and trailing edge material are sometimes underestimated with a resulting serious unbalance condition, and a larger value of K/I. In addition changes in service may tend to increase the

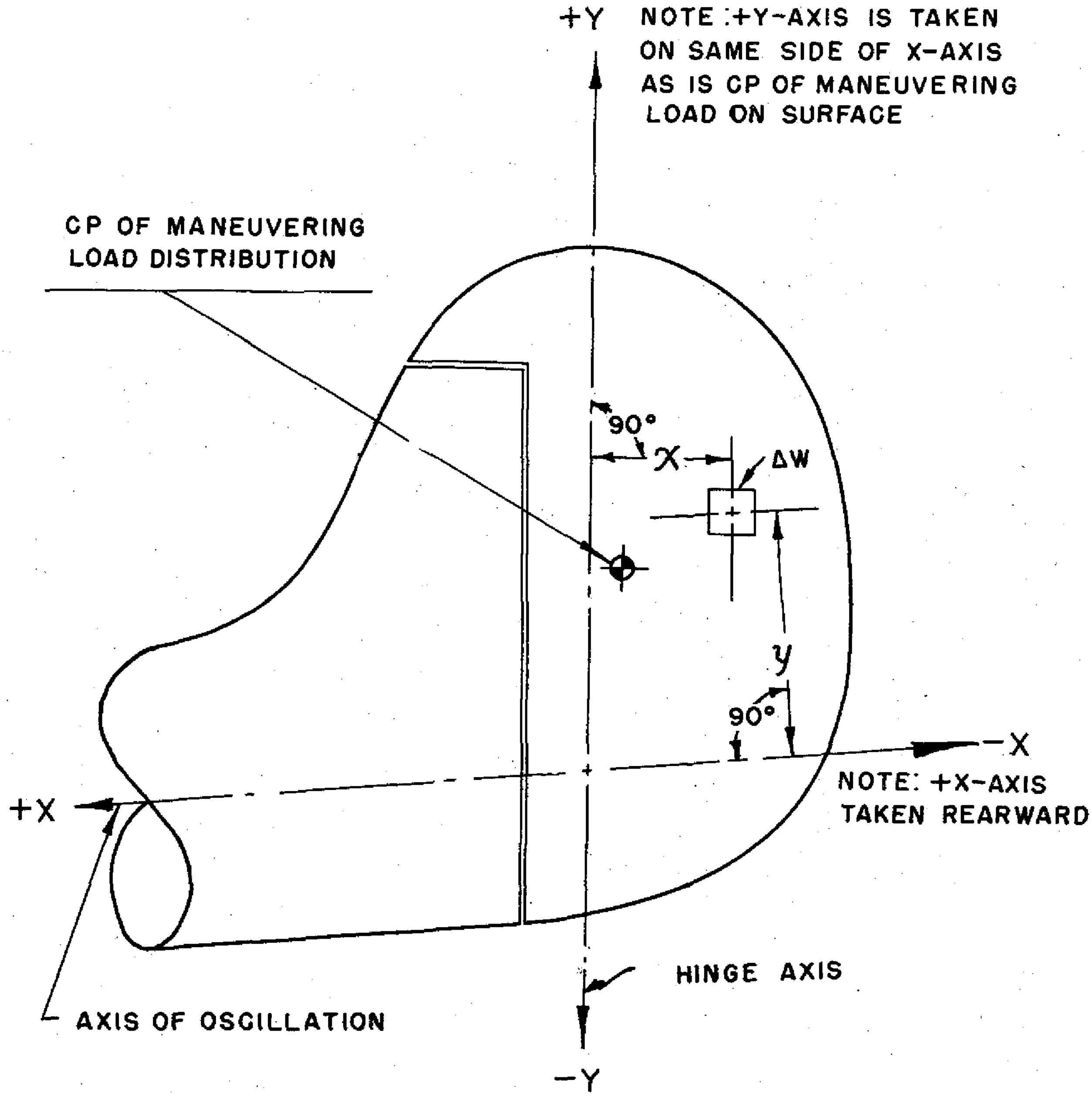


FIG. 35 DYNAMIC BALANCING OF CONTROL SURFACES
(REF. CAM 04.424-3)

TYPICAL TABLE

for

MASS BALANCE COMPUTATIONS

MODEL:__________ SURFACE:__________ DATE:__________

Item No.	Part No.	Description	Weight w lbs.	Dist. from hinge ℄ x inches	x^2	Moment = wx − inch-lbs.	+ inch-lbs.	$I_{y-y}=wx^2$ lb.-ins.^{2}	Dist. from oscil- lation axis = y In.	$K = wxy$ −	+
			(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)
1 2 etc.											
		TOTALS	Σ			$-\Sigma$	$+\Sigma$	Σ		$-\Sigma$	$+\Sigma$

STATIC UNBALANCE = ALGEBRAIC SUM OF Σ(Col. 4) and Σ(Col. 5) =

$$K/I_{y-y} = \frac{\Sigma(\text{Col. } 8) + \Sigma(\text{Col. } 9)}{\Sigma(\text{Col. } 6)} =$$

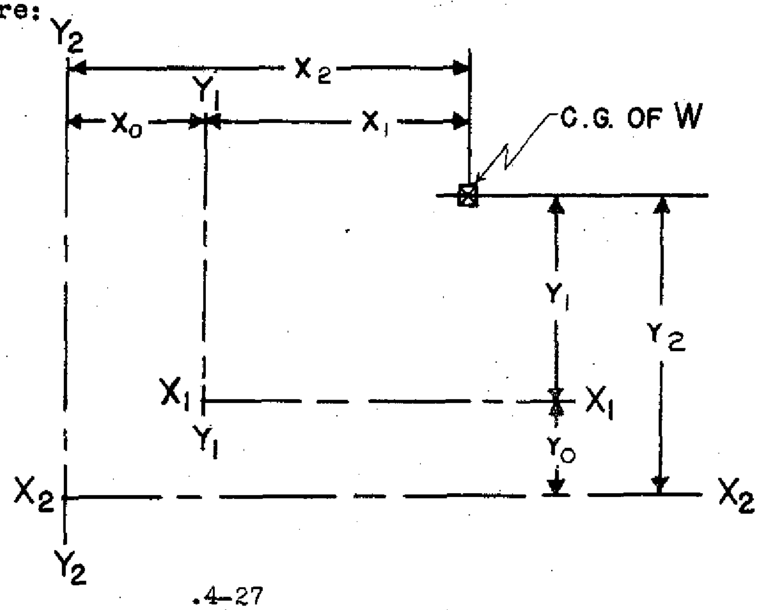
FIG. 35A

unbalance. Referring to Figure 35 of CAM 04, the product of inertia of the item of weight, w, is equal to wxy. The product of inertia, K, of the complete surface is the sum of the individual products of inertia of each part. Hence, K = Σ wxy. The weights should be expressed in pounds and the distance in inches. K is then in lbs.-inches^2.

d. <u>Moment of Inertia of Control Surface about the Hinge Axis</u>. The moment of inertia (I_{y-y}) of the control surface about its hinge axis may be computed from the data found for computing K (in paragraph c, above). I for a small part of the weight, w, is equal to wx^2, when x is the perpendicular distance from its CG to the hinge axis. Hence I_{y-y} is equal to the sum of the individual moments of inertia of each part and is equal to $\Sigma\, w\, x^2$. The weight should be in pounds and the distance in inches, so that I_{y-y} will be in lbs.-inches^2. It should be noted that the correct value of I_{y-y} will only be obtained, if the weight items are broken down into a sufficient number of small parts, <u>especially in the chordwise direction</u>. This is particularly important for such items as fabric covering and tape, dope, metal skin, trailing edge tabs, tab operating mechanism, etc., <u>unless</u> the moment of inertia is first obtained about a parallel axis through the CG of the larger concentrated weight, w, and then transferred to the hinge axis; i.e., $I_{y-y} = I_{CG} + w\, d^2$ where d is the perpendicular distance in inches between the CG and the hinge axis.

e. The dynamic balance coefficient is then equal to K/I for the X and Y axes assumed.

f. It is sometimes found necessary to calculate the product of inertia (K_2) with respect to one set of axes (X_2 and Y_2) given the product of inertia (K_1) with respect to another set of axes (X_1 and Y_1) lying in the same plane. Referring to the figure:



$$K_2 = K_1 + x_o\, y_1\, W + y_o\, x_1\, W + x_o\, y_o\, W$$

Where W = Total weight in lbs. and x_1 and y_1 are the coordinates (in inches) of the CG with reference to the X_1 and Y_1 axes respectively, and x_o and y_o are the distances (in inches) between the X axes and Y axes respectively.

It should be pointed out that in the case of statically balanced control surfaces (zero unbalance), the product of inertia (K) is independent of the true location of the axis of oscillation (X) but not of its direction.

5. Product of Inertia with Respect to Two Axes that are not Mutually Perpendicular. This case might occur for some wing bending versus aileron mode of vibration, with, for example, the relations shown in Figure 35B. As shown in ACIC #711, the product of inertia for the inclined axes (O-O and Y-Y) can be obtained from the perpendicular axes (X-X and Y-Y) value by the use of the following equation:

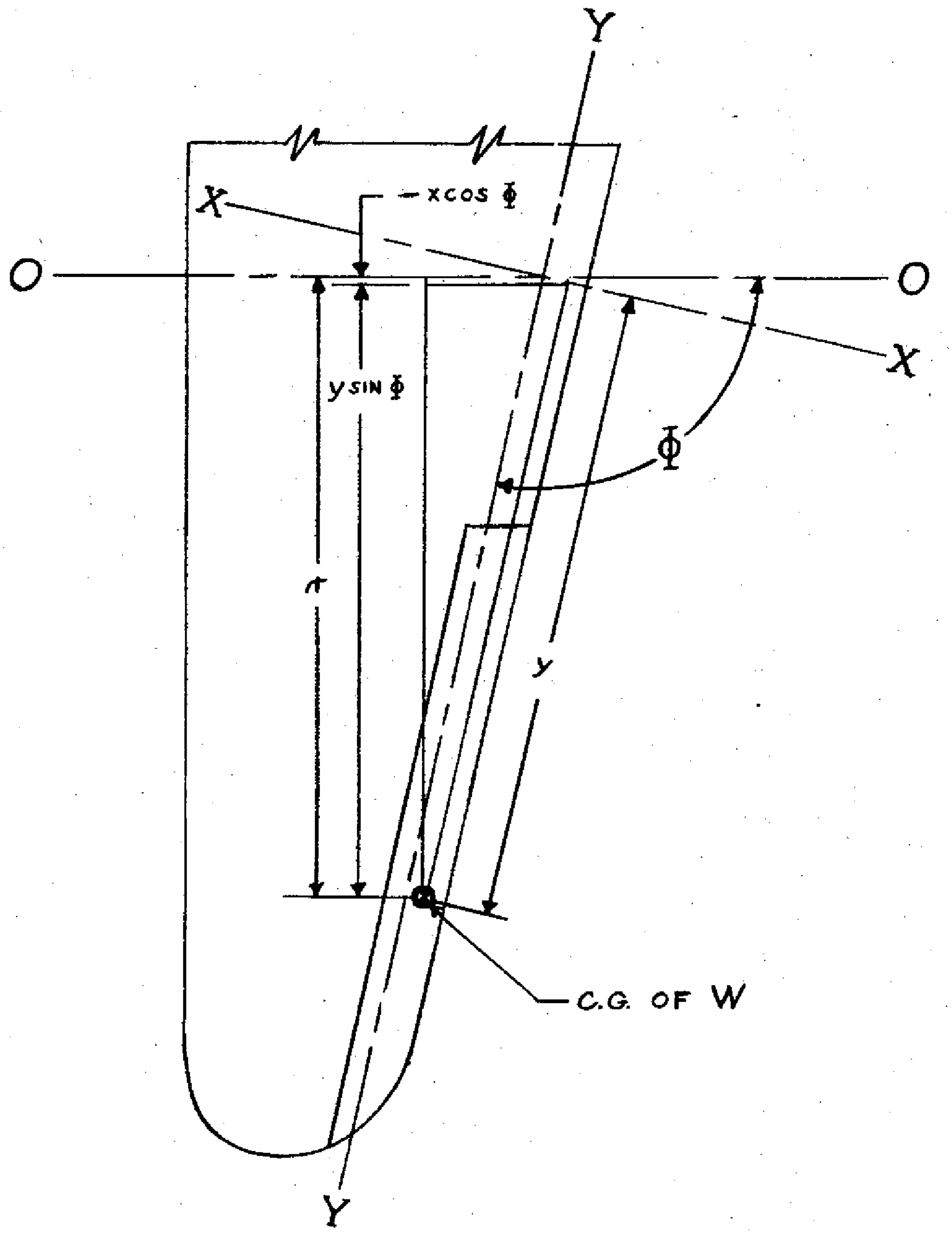
$$K_{oy} = K_{xy} \sin \phi - I_{y-y} \cos \phi$$

If ϕ is taken as the angle between the hinge axis and the axis of oscillation in that quadrant where the center of gravity of the surface is located, neglecting the inclination of the axes will be conservative if ϕ is acute; if ϕ is obtuse the result may be unconservative, especially if K is small compared with I.

6. Product of Inertia with Respect to Two Axes that are Parallel and in whose Plane the Control Surface CG is Located. This case may be of importance in some of the wing torsion versus aileron and fuselage bending versus rudder or elevator modes of vibration. Using the same nomenclature as in the previous cases where Y-Y is the hinge line of the control surface and X-X is the axis of oscillation of the body as shown in Figure 35C which represents a fuselage side bending versus rudder mode of vibration, then

$$K_{xy} = x_o\, x_1\, W + I_{y-y}$$

where:
- x_o is the distance between the two parallel axes in inches.
- x_1 is the distance of the CG of the control surface (including balance weights) from the hinge line in inches (aft of hinge is positive and forward is negative).
- W is the weight of the control surface (including balance weights) in lbs.
- I_{y-y} is the moment of inertia of the control surface (including balance weights) about the hinge line in lbs.-inches^2.



INCLINATION OF THE AXIS FOR A TYPICAL WING-BENDING VERSUS AILERON MODE OF VIBRATION.

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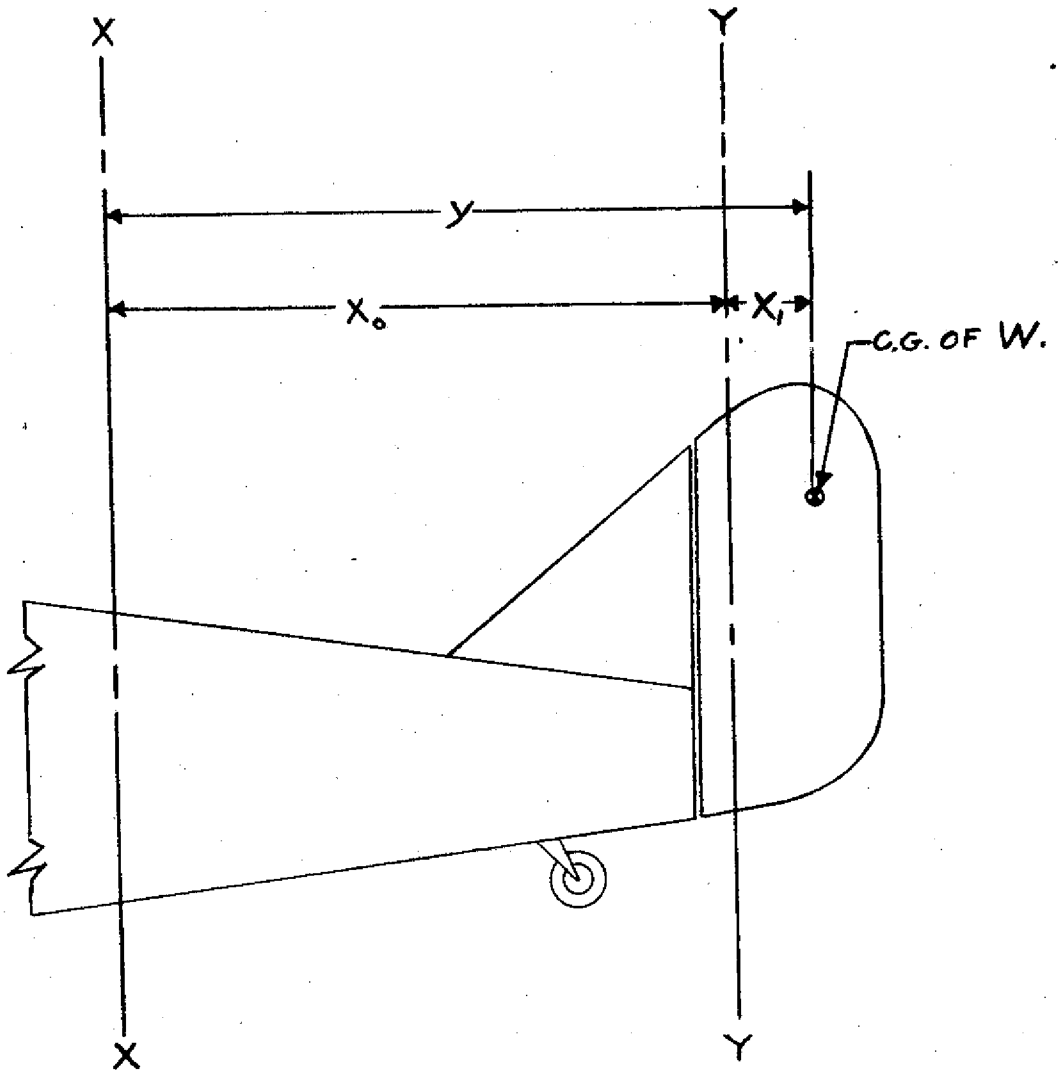
FIG. 35B



PARALLEL AXES FOR A TYPICAL FUSELAGE-BENDING VERSUS RUDDER MODE OF VIBRATION.

FIG. 35C

It is thus obvious that to make K equal to zero for this mode of vibration x_1 must be negative; that is, the center of gravity of the control surface must lie forward of the hinge line.

7. The special case of parallel axes wherein the control surface CG is located outside of the plane of the axes, may in most cases be resolved into the above parallel axes case (6) by projecting the X-axis to the plane through the control surface hinge resulting in a new X'-axis and resolving the CG reaction into components perpendicular and parallel to this plane. This may only be done when it can be shown that the CG of the control surface falls in the new plane through the X' and Y-axes which will be found true for most rudders and elevators. However, for the aileron, as shown in Figure 35D, where the hinge axis is usually near the bottom of the surface resulting in the CG being above an X'-Y plane, it will be necessary to consider the components of the CG, since an appreciable unbalance may be present even with a statically balanced aileron, for the true oscillation mode involving rotation about the X-axis.

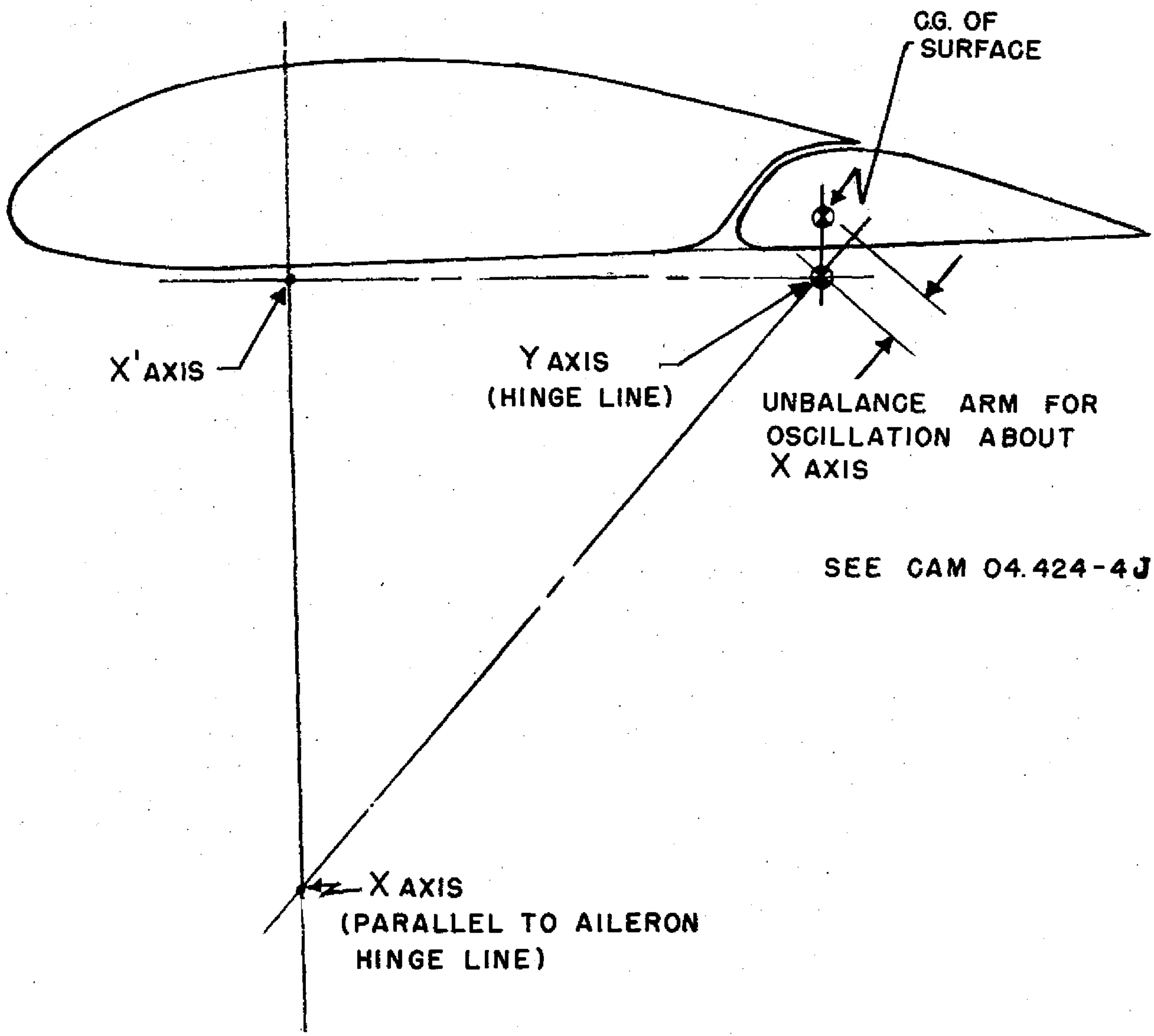
.425 WING FLAPS

1. In addition to the usual air loads, flaps may be subjected to high local loadings from impact of water when the airplane is operated from wet fields, or when used on seaplanes. This is particularly true of low-wing installations.

2. Ground clearance of the flaps should be considered in the initial design stages, 12 inches being a reasonable minimum. Since flap travel may be varied before final approval in order to secure the desired flight-path, trim, or landing characteristics, the maximum expected travel should be used when determining clearance.

.426 TABS

1. Minimum deflections and play are of first importance in the installation of these surfaces. Strength of the surface and anchorage should be sufficient to prevent damage or misalignment from handling. This is particularly true of thin sheet tabs which are set by bending to the proper position. See also CAM 04.424.



PARALLEL AXES WITH THE CONTROL SURFACE
OUTSIDE THE PLANE OF THE AXES

FIG. 35D

.43 DETAIL DESIGN OF CONTROL SYSTEMS

1. General. The movements of horns, cables and other components with respect to each other should be such that there is no excessive change in system tension throughout the range. Adjustable stabilizer-elevator combinations, in particular, should be checked for this condition. Pulley guards should be close fitting to prevent jamming from slack cables since wide temperature variations will cause rigging loads to vary appreciably. The design of the pulley brackets should be such that the pulley lies in the plane determined by the cable. Allowable tolerances in manufacturing should not permit the cable to rub against the pulley flanges.

2. Travel. The travel of the primary control elements is generally dependent on the size of the aircraft. Stick travel at the grip may vary from 18" x 18" total to much smaller values for light aircraft. Angular travel of the control wheel from neutral may vary correspondingly from 270° to 90°. A usual value of pedal travel is 6" total. There is a trend toward adjustment for variations in stature of the pilot, either in the seat or at the controls.

3. Positioning. In the layout and positioning of a control consideration should be given to its relative importance and to its convenient placement for the usual sequence of operations. Thus for landing, it is desirable that throttle, propeller pitch control, flap control, and brakes be operable without changing hands on the wheel or stick. Likewise secondary controls such as fuel valves, extinguishers, and flares should be so located that the possibility of accidental or mistaken operation is remote.

4. Centering Characteristics. A point sometimes overlooked is the effect of the weight of a control member or of a pilot's arm or leg on the centering characteristics of the control. For instance, resting the hand on a stick grip in which the fore and aft axis is not directly below the grip will tend to apply aileron. Likewise rudder pedals on which the whole foot is rested and which have their hinge line below the pedal will tend to move away from center.

5. Cables. Control cables should be of the 6 x 19 or 7 x 19 extra flexible type, except that 6 x 7 or 7 x 7 flexible cable is acceptable in the 3/32 inch diameter size and smaller provided that particular care is taken to prevent wear.

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"Cable smaller than 1/8 inch diameter should not be used in primary control systems, except that smaller sizes may be used for tab control systems where, in the event of cable failure, it is demonstrated that the airplane can be safely controlled in flight and landing operations." (See the following paragraph 7 regarding the use of fairleads.)

"For properties of control cable see Table 4-14 of ANC-5 and for cable terminal efficiencies see 4.530 of ANC-5."

End splices should be made by an approved tuck method such as that of the Army and Navy, except that standard wrapped and soldered splices are acceptable for cable less than 3/32 inches in diameter. Approved swaged-type terminals are also acceptable. It should be remembered that cable sizes are governed by considerations of control system deflection as well as by strength requirements, particularly when long cables are used.

6. Spring type connecting links for chains have been found to be not entirely satisfactory in service. It is advisable that a more reliable means, such as peening or cotter pins, be employed.

7. Fairleads should be used to prevent cables, chains and links from chafing or slapping against parts of the aircraft, but should not be used to replace pulleys as a direction-changing means. However, where the cable load is small, and the location is open to easy visual inspection, direction changes (through fairleads) not exceeding 3° are satisfactory in primary control systems except when 3/32 inch diameter cable is used. (See paragraph 5.) A somewhat greater value may be used in secondary control systems. Because of its corrosive action on cables, rawhide should not be used for fairleads or chafing strips.

8. When using extreme values of differential motion in the aileron control system or a high degree of aerodynamic balance of the ailerons, the friction in the system must be kept low, otherwise the ailerons will not return to neutral and the lateral stability characteristics will be adversely affected. This is particularly true when the ailerons are depressed as part of a flap system, in which case there may even be definite overbalance effects.

9. Adjustable stabilizer controls should be free from "creeping" tendencies. When adjustment is secured by means of a screw or worm, the lead angle should not exceed 4° unless additional friction, a detent, or equivalent means is used. In general, some form of irreversible mechanism should be incorporated in the system, particularly if the stabilizer is hinged near the trailing edge.

10. Dual control systems should be checked for the effects of opposite loads on the wheel or stick. This may be critical for some members such as aileron bell crank mountings in an "open" system, i.e., no return except through the balance cable between the ailerons. In addition, the deflections resulting from this long load path may slack off the direct connection sufficiently to cause jamming of cables or chains unless smooth close-fitting guards and fairleads are used.

11. It is essential that control systems, when subjected to proof and operation tests, indicate no signs of excessive deflection or permanent set. In order to insure that the surfaces to which the control system attaches will retain their effectiveness in flight, the deflection in the system should be restricted to a reasonable limit. As a guide for conventional control systems, the average angular deflection of the surface, when both the control system and surface are subjected to limit loads as computed for the maneuvering condition neglecting the minimum limit control force but including tab effects, should not exceed approximately one-half of the angular throw from neutral to the extreme position. (See CAM 04.42-1)

12. It is essential that when a nose wheel steering system is interconnected with the flight controls care be taken to prevent excessive loads from the nose wheel overstressing the flight control system. This objective may be attained by springs, a weak link, or equivalent means incorporated in the nose wheel portion of the control system.

.431 STOPS

1. Although the location of stops within the control system is not specified, they should preferably be located close to the operating force in order to avoid a "springy" control. As noted in CAM 04.421, additional stops may in some cases be needed at the surfaces. Stops should be adjustable where production tolerances are such as to result in appreciable variation in range of motion.

.432 JOINTS

1. Bolts, straight pins, taper pins, studs, and other fastening means should be secured with approved locking devices. (See CAM 04.4020) Rivets should not be subjected to appreciable tension loads.

2. The assembly of universal and ball and socket joints should be insured by positive locking means, rather than by springs. In addition the angular travel of such joints should be limited by system stops rather than by accidental interferences which may induce extremely high stresses in the joints.

3. Woodruff keys should not be used in tubing unless provision is made against the key dropping through an oversize or worn seat.

.434 FLAP CONTROLS

1. Undesirable flight characteristics, such as loss of lift and consequent settling, may result from too rapid operation of flaps which give appreciable lift. When the prime function of the flap is to act as a brake, however, slow operation is not so important. When flaps extend over a large portion of the span the control and means of interconnection should be such as to insure that the flaps on both sides function simultaneously.

.435 TAB CONTROLS

1. In addition to the air loads, consideration should be given in the design to the lapping effect of dust and grease on fine threads, deflections of the tab due to the small effective arm of the horn or equivalent member, and vibration common to the trailing edge portion of most movable surfaces.

2. It is advisable to avoid a tab control with small travel because of the resulting abrupt action of the tab.

.437 SINGLE-CABLE CONTROLS

1. Single cable controls refer to those systems which do not have a positive return for the surface or device being controlled. Rudder control systems without a balance cable at the pedals are considered satisfactory if some means such as a spring is used to maintain cable tension and to hold the pedals in the proper position. It should be noted that it is not the intent of the specified requirement to require a duplication of cables performing the same function.

13. Power Boost Controls. Such controls should exhibit control force versus surface deflection curves which are smooth and free from discontinuities. (See also CAR 04.75120.) Consideration should be given in the design and in test to the effects of the temperature variation to be expected in operation, in order to avoid the possibility of jamming or excessive lag. Small changes in valve adjustments or other settings should not result in marked changes in operating or control characteristics.

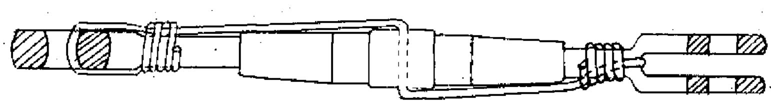
-14 Installation of Turnbuckles. Fork ends of turnbuckles should not be attached directly to control surface horns or to bellcrank arms unless a positive means (such as the use of shackles, universal joints, etc.) is used to prevent binding of turnbuckles relative to the horns or bellcrank arms which may be caused by excessive tightening of the attaching bolts, or unless it can be shown that the turnbuckles have adequate strength assuming one end fixed and the design cable loads pulling off the other end at 5° to the turnbuckle axis. Care should be taken to insure that there is no interference between the horns or bellcrank arms and the fork ends of turnbuckles, throughout the range of motion of the control surfaces.

-15 Safetying of Turnbuckles. All turnbuckles should be safetied with wire as indicated below. After safetying the turnbuckle, no more than three threads should be exposed, and the ends of each safety wire should be securely fastened by at least four wraps.

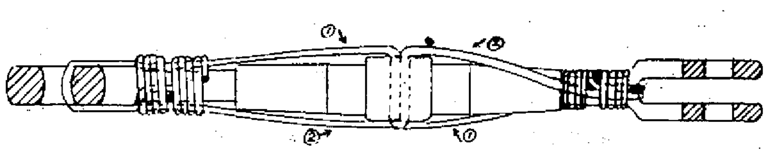
Turnbuckle Strength (AN Std.)	Type of Wrap	Diam. of Wire	Material (Annealed Condition)
800	Single	.040	Copper, Brass, Galv. Steel*
1600	Single	.040	Copper, Brass, Galv. Steel*
2100	Single	.040	Stainless Steel
2100	Double	.040	Copper, Brass, Galv. Steel*
3200 and greater	Single	.072	Copper, Brass, Galv. Steel*
	Double	.051	Copper, Brass, Galv. Steel*
	Double	.040	Stainless Steel

* Galvanized or tinned soft iron or tinned steel wires are also acceptable.

SINGLE WRAP



DOUBLE WRAP



Notes: 1. Wire 1 is passed through the turnbuckle hole as shown, the two wire ends are passed through the right and left hand ends of the turnbuckle and are then bent back along the barrel of the turnbuckle.

2. Wire 2 is installed and wrapped (these wraps are next to the ends of the turnbuckle).

3. The two loose ends of wire 1 are then wrapped.

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.44 DETAIL DESIGN OF LANDING GEAR

1. The wheel travel should be ample for the service and requirements involved. The geometric arrangement of members in the landing gear should be such that the wheel travel in the direction of the resultant external force will be adequate. See CAM 04.440-1. Extremely high heat treats, particularly when combined with thin sections, are usually sources of trouble in service. An ultimate strength of 180,000 pounds per square inch may be regarded as a usual upper limit, except in special cases. To prevent binding and scoring in shock absorbers it is desirable to keep bending deflections, and bearing stresses at pistons, packing glands and bearings, at low values.

2. In general the purpose of unconventional gear is to facilitate landing under unfavorable conditions. In order to realize this purpose it is advisable that the energy absorption capacity be in excess of that needed for conventional gear.

.440 SHOCK ABSORPTION

1. In order to obtain adequate energy absorption without exceeding the specified load factors it is essential to provide sufficient wheel travel. Neglecting the effect of tire and structural deflection, it may be shown that:

$$t = \frac{h}{n\eta - 1}$$

t = component of wheel travel in the direction of the resultant external force.

h = specified height of drop,

n = load factor, and

η = absorber efficiency.

Thus when a certain height of drop h must be met without exceeding a load factor n, the recommended minimum wheel travel for any absorber efficiency may be computed. While absorber efficiencies as high as .85 have been developed, it should be noted that such shock absorbers tend to give bouncing and undesirable taxiing characteristics. This may be obviated by ample travel in combination with an absorber which does not develop high loads in the first part of travel but rather "builds-up" gradually to a peak load only when near the fully deflected position. In such cases, an efficiency of .60 to .70 may be expected. The effect of the tire in altering the above relationship will in general not be large because, while it provides additional energy absorption, its deflection increases the energy to be dissipated. Structural deflections, while not usually of importance, may in some cases appreciably reduce load factors.

.443 A wheel appended to a previously approved tail skid installation will not be classed as a "landing gear wheel". See CAM 04.060-2 for an acceptable procedure of use in making such a change.

.444 RETRACTING MECHANISM

1. The requirement of a visual position indicating means may usually be met by mechanically or electrically operated indicators. When windows or other openings are so placed that it is possible for the pilot to note directly the position of the wheels, a separate visual indicator is not required. In such cases, however, it is essential that illumination be provided for night operation. When it is necessary for latches to operate before the gear will carry landing loads, lights or other means should be used to indicate completion of this operation. In the case of amphibians the requirement in CAR 04.444 regarding aural indicators does not apply. With this type of airplane it is usually more important to guard against the possibility of alighting on the water with the wheels down.

2. In the design of retracting systems, the source of most service troubles lies in such items as latches (particularly if spring loaded), limit switches, valves, cable installations, universal joints, and indicating systems. The effects of mud, water, ice and extreme temperature variations should be studied.

3. In manually operated systems it is desirable that the crank or lever forces not exceed 15 to 20 pounds. Further, about sixty 12 inch strokes per minute is a practical maximum. Hence the total work input for operation varies with the time. To keep this at a reasonably low value, it is therefore important that losses be kept small. With larger and heavier gear the use of a bungee may be necessary.

.4440 The usual reduction ratios of screw and nut, and of worm and worm wheel combinations, are considered to provide irreversibility. Detents or other means should be provided however if there is appreciable creeping. Some types of swinging arms which move slightly past dead center to a position against a stop are also acceptable, but the effect of bouncing on landing should be considered.

.45 HULLS AND FLOATS

1. General practice in the design and construction of floats and hulls is well established. Rivet spacing for watertight joints is substantially closer than required for structural strength. The same applies to spacing of spot welds. Drain holes should be positioned at stringers, transverse frames, and other members so that water will drain to the low point without being trapped in pockets at inaccessible points. Adequate inspection openings should be provided. When the bottom is curved in transverse section there may be high loads acting inward at the chine between frames due to the tension in the bottom plating.

2. Due to the severe nature of the loads imposed by water operation, consideration should be given to the effect of sharp impacts and racking loads. Particular attention should be paid to fittings, and, in twin float seaplanes, to trusses and members carrying unsymmetrical loads.

.450 BUOYANCY (MAIN SEAPLANE FLOATS)

1. It should be noted that Canadian requirements specify that twin-float seaplanes shall have at least 100 per cent reserve buoyancy in the floats. See also CAM 04.451.

.451 BUOYANCY (BOAT SEAPLANES)

1. Any of the methods common to naval architecture may be used to demonstrate compliance with buoyancy requirements. Bulkheads should be watertight at least 18 inches above the water line being considered. Acceptable substitutes for watertight doors in bulkheads are sills or sections which may be slid or set into place. These should likewise extend at least 18 inches above the waterline considered, and should be quickly installable. Bulkheads should possess ample strength to withstand hydrostatic loads with some reserve for surges. Cables in the hull should not be carried below the waterline due to the impracticability of sealing at watertight bulkheads. Watertight closed compartments should be vented to a point well above the waterline and consideration should be given to air pressure variation at the venting point.

.452 WATER STABILITY

1. The methods employed in naval architecture may be used to demonstrate compliance with the stability requirements. In some cases this compliance has been shown by assymetric loading of the aircraft on the water. Computations are acceptable but with certain types of seaplanes, such as those incorporating seawings, the use of metacentric height as a criterion becomes meaningless due to variation with list and loading. Recourse must then be made to methods such as Bonjean curves or the homogenous mass method to demonstrate the existence of adequate righting moments. For a further discussion of methods see texts such as "The Naval Construction" by Simpson, "Theoretical Naval Architecture" by Attwood, and "Engineering Aerodynamics" by Diehl. Note that the Canadian requirements for twin float seaplanes specify that the metacentric height shall not be less than the following values:

Transverse metacentric height = $4\sqrt[3]{D}$ ft, and

Longitudinal metacentric height = $6\sqrt[3]{D}$ ft, where

D = total displacement of the seaplane in cubic feet.

.4632 OPERATION LIMITATIONS AND INFORMATION

Satisfactory means of informing operating personnel of necessary operation limitations and information are outlined below:

1. Instrument marking should be used for:

   a. Airspeed not to be exceeded in glide or dive.
   b. Airspeed not to be exceeded in level flight or climb.
   c. Airspeed not to be exceeded with flaps extended.
   d. R.P.M. not to be exceeded in take-off.
   e. R.P.M. not to be exceeded in climb.
   f. R.P.M. not to be exceeded in all other operations.
   g. Manifold pressure not to be exceeded in take-off.
   h. Manifold pressure not to be exceeded in climb.
   i. Manifold pressure not to be exceeded in all other operations.

   When airspeed indicators, tachometers, and manifold pressure gauges are so marked, the coloring outlined below should be used:

   a, d, g - to be marked in "red".
   b, e, h - to be marked in "yellow".
   c, f, i - to be marked in "green".

2. Acceptable methods of marking include:

   a. Pointers, adjustable on the ground only.
   b. Sectors or lines properly marked and outlined on the face of the dial, under the glass.
   c. Lines painted on the glass face of the instrument when a or b above is impracticable and when the glass is adequately secured against rotation. Such lines should be painted over a suitably etched or scratched line on the glass itself. This etching or scratching is considered advisable for more serviceable markings.

3. When considered necessary by the Authority, operating information and limitations such as the following should be included in a manual, or its equivalent, which must be carried in the pilot's compartment and be accessible at all times:

   a. Emergency ceiling and conditions under which it may be obtained.
   b. All other information or limitations considered necessary by the Administration to properly inform operating personnel of the conditions necessary for operation in compliance with the Civil Air Regulations.

APPENDIX I

AN INTERPRETATION OF CAR 04.003 FOR THE CASE OF LARGE AIRPLANES

A GENERAL

1. Since, as stated in CAM 04.00 , the present CAR 04 requirements are based largely on experience with airplanes weighing less than 30,000 pounds, it is realized that certain of these requirements cannot logically be applied to larger and larger aircraft without involving either the danger of inadequate rules or the disadvantage of too severe requirements. It is therefore essential that, during the initial stages of the design of such airplanes, the designer contact the Administrator for special rulings which will be made for the particular design involved. It is likewise essential that very close cooperation be maintained between the designer and the Administrator throughout the design period and until the completion of the airplane.

2. Although it is impossible to anticipate all of the new airworthiness problems involved in the design of large aircraft, the modifications to CAR 04 which are outlined in the following sections are considered to be generally applicable to such aircraft. If cases arise in which there is doubt as to their applicability to a particular project, the designer is of course at liberty to employ alternative modifications, provided that such modifications are substantiated. This appendix will be revised from time to time as new modifications are adopted.

B STRUCTURAL LOADING CONDITIONS

1. Design Gliding Speed. (See CAR 04.211). A V_g of less than 1.25 V_L is in general believed inadequate. This factor may, however, be reduced if it is shown that the resulting placard maximum speed suffices for all the contingencies which may arise in operations. It is suggested that a polar diagram be plotted, showing the flight paths, indicated air speeds, and rates of descent, with zero thrust and with cruising power. This will assist in determining the adequacy of the design gliding speed proposed.

2. Maneuvering Load Factors. (See CAR 04.2120). Although large airplanes are generally less maneuverable than smaller ones, they are also, in many cases, less controllable after a maneuver has been begun, either advertently or inadvertently. Pending the development of more rational maneuvering load factor criteria for such airplanes, it is believed that the minimum limit maneuvering load factors of + 2.67 and - 1.333 should be used at all speeds up to V_g.

3. Gust load factors. (See CAR and CAM 04.2121.) Positive and negative values of U of 30 feet per second (limit) should be used in Condition I (CAR 04.2131) and Condition II (CAR 04.2132). The resulting gust load factors should also be used for Condition III and IV respectively.

4. Horizontal Tail Surfaces. (See CAR 04.221). A 30 foot gust should be used for the design of the horizontal surfaces at V_L. The effects of downwash on the horizontal tail may be allowed for. More definite information on this can probably be obtained from the NACA. The question of maneuvering loads is difficult to decide at present. The existing requirements may be satisfactory, but should not be relied on as final. A rational study of the specific case involved, based on the maximum deflection likely to be used at Vp, may lead to more applicable normal force coefficients than those specified by CAR 04.2211.

5. Ailerons. (See CAR 04.223). It is suggested that the maximum deflection likely to be used at Vp be taken as a criterion for aileron design loads. This will involve an investigation of aileron loadings based on normal force coefficients and pressure distribution data.

6. Wing Flaps. (See CAR 04.211, 04.214, and 04.244). The present requirements for flap design speeds can probably be lowered to 1.67 V_{sf} (placard 1.5 V_{sf}) provided that gust velocities of + 30 and - 30 feet per second are used in Conditions VII (CAR 04.2141) and VIII (CAR 04.2142) respectively. If partial deflections are to be used at higher speeds an additional investigation is necessary.

7. Loads on Sea Wings. No strength requirements have been formulated for sea wings. The suitability of such installations will be determined by operating tests. It should be borne in mind, however, that water is approximately 800 times as dense as air and that sea wings and floats are therefore subjected to very high loads and pressures when they encounter waves in landing or on take-off. The manufacturer proposing to use sea wings should substantiate the loading conditions chosen for their design.

C PROOF OF STRUCTURE

1. Effects of Size. It appears that existing airplane structures have just about reached the limit of safe extrapolation from previously approved structures and that further increase in size introduces an element of uncertainty difficult to remove. In view of the serious nature of this situation it is suggested that designers prepare a comprehensive outline of the general methods of strength analysis to be used on wings, fuselages and hulls, and of the specimen tests which will be made to supplement the analysis. This material should be submitted to the Administrator as early in the design stages as is practicable. It is apparent that a thorough study of this situation is necessary if the Administrator is to avoid requiring high margins of safety which will impair the efficiency of the airplane. Otherwise it may be necessary to conduct destruction tests of complete components.

2. Wing Analysis. In preparing the program mentioned in 1 above, the following points should be considered:

a. Determination of the magnitude and distribution of stresses due to bending and torsion.

b. Determination of allowable compressive loads in wing covering.

c. Allowable shear loads in webs.

d. Combined loadings.

e. Specimen tests, panel tests, and partial wing tests.

f. Ultimate factors of safety. These may be increased over the present required values if there appears to be uncertainty as to the reliability of the strength analysis and test methods).

3. Fuselage and Hull Analysis. A program such as outlined for wings in item 2 above should be submitted. In particular, information should be included as to the strength of main and intermediate frames; the rigidity of intermediate frames and their adequacy in regard to the prevention of general instability; the strength of the side covering in shear; the strength of vertical and longitudinal stiffeners as affected by diagonal tension fields; the effectiveness of the covering in compression, and the effects of cutouts and discontinuities.

D DETAIL DESIGN

1. Flutter Prevention. Before the design has progressed very far, the Administrator should be informed as to all design features and precautions to be used to prevent flutter. Unusually large cantilever spans, and outboard vertical tail surfaces, may necessitate special precautions.

2. Control Systems. If a power control system is used, it will probably be required that certain minimum maneuvers can be performed after the power source has failed.

3. Exits. In view of the large size of the compartments, it is felt that consideration should be given to supplying emergency exits on each side and at the top of each major compartment.

APPENDIX II

(Sample Weight and Balance Report)

NAME OF MANUFACTURER

REPORT NO. ___

WEIGHTS AND BALANCE
OF MODEL _______.

SERIAL NO. ______

IDENTIFICATION MARK ______

Date _______________.

Prepared By ____________________
Checked By ____________________
Witnessed By ____________________
(Signature of Civil
Aeronautics Administration
Representative)

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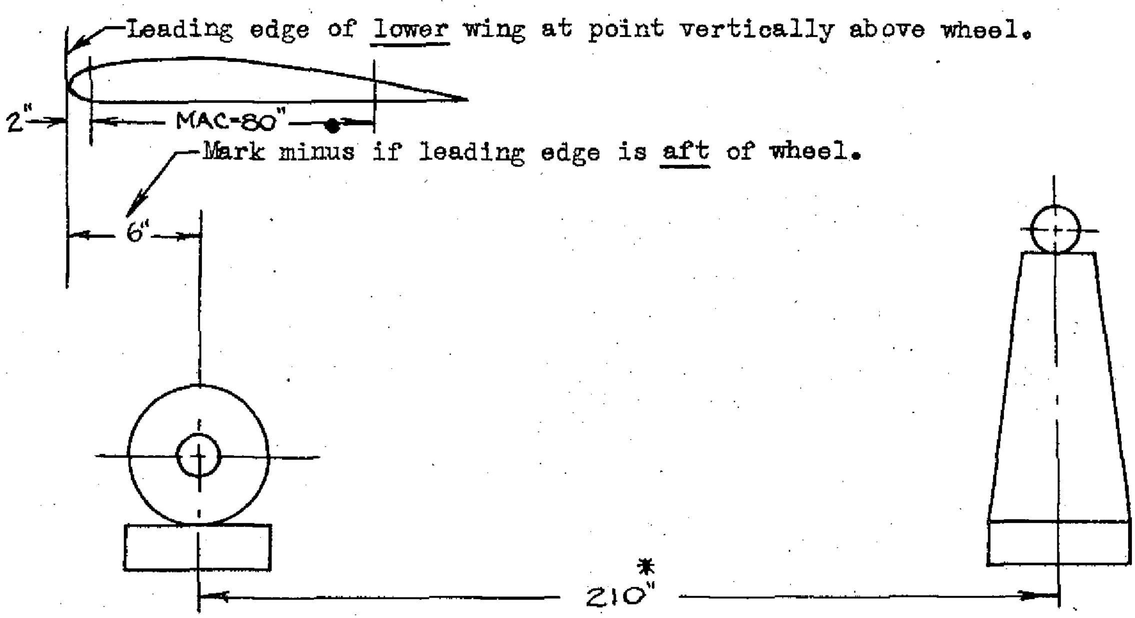
Report No.________

Section 1. Aircraft Empty Weight

(A) Empty weight as weighed (in level landing position**)



	Scale Reading	Tare	Net
Left Wheel	1020	15	1005 lbs.
Right Wheel	1010	15	995 lbs.
Tail Wheel	400	150	250 lbs.
		Total	2250 lbs.

Total net empty weight includes residual oil. The oil tank was filled and the system drained before weighing. 5 gallons of oil were drained from the system.

C.G. Empty (as weighed) is aft of wheel centerlines $\frac{250 \times 210}{2250} =$ 23.3"

C.G. Empty (as weighed) is aft of lower wing leading edge 23.3 + 6 = 29.3"

Lower wing leading edge is aft of datum 100.0"

C.G. Empty (as weighed) is aft of datum 100 + 29.3 = 129.3"

Datum to M.A.C leading edge = 102" (See page 2 of Report 981)

* Measured along floor with aid of a plumb-bob.

** Level by means provided in accordance with CAR 04.91.

(B) Empty weight as weighed includes the following:

(1) Class I Equipment*

Item No.***	Name	Weight**
10	Starter	21
11	Battery	40
12	Heater	2
13	Ventilator	4
14	Generator	20
15	Position lights	--
16	8.50-10 wheels (Mfr. and model) and 8.50-10 6-ply tires	Weights not required except when optional wheels are used.
17	10 1/2 in. streamline tail wheel	
6	Instruments not required (list)	

(2) Items for which approval as Class II or Class III OPTIONAL equipment is desired (and test equipment):

		Weight** (Net increase)	Hor. Arm from Datum	Hor. Moment
7	Wheel streamlines	24	71	1704
19	Flares (Type)	17	175	2975
4	Adj. metal prop. 70 lbs. (Class I prop. is wood 46 lbs.)	24	13	312
6	Optional instruments not required (list)	15	60	900
20	Optional fuel capacity 70 gals. (2 tanks at 35 gals.) (Class I capacity includes 2 tanks at 25 gals. 33 lbs.)	15	90	1350
21	Radio			
	Receiver (Type) and antenna	30	60	1800
	Shielding (Type)	10	16	160
	Bonding	10	50	500
5	Ballast container and straps, etc.	20	138	2760
	Total optional	165		12461

(3)

	Weight	Arm	Moment
Empty weight as weighed	2250	129.3	290925
Optional Equipment	-165		-12461
Basic empty weight	2085	X_E	278464

$X_E = \frac{278464}{2085}$ = Distance from datum to C.G. of airplane empty with all Class I items only.

* "Class I Equipment" (See CAM 04.0322). List all such items even though weights are not included for some.

** All weights of equipment are installation weights. When weight listed is net increase over Class I equipment, list weights for both as noted for propeller and fuel tanks above (items 4 and 20).

*** Item Numbers to correspond with numbers used in Balance Diagram.

Section 2 - Most Forward C.G. Load Condition

(A) Loading as actually flown:

Item No.	Name	Weight	Hor. Arm	Hor. Moment
	Empty weight as weighed	2250	129.3	290925
1	Oil 5 gals.	38	51	1938
2	Fuel 20 gals.	120	90	10800
3	Pilot + parachute	225*	90	20250
4	Propeller (If other than noted in Section 1(B))			
5a	Ballast (incl. containers, straps, etc.)	100	60	6000
	Totals	2733	120.8	329913

Datum to M A C leading edge 102 102

Per cent of M A C 18.8 ÷ 80(MAC) = 23.5%

Inches aft of leading edge of wing 120.8 - 100 = 20.8 in.

(B) Loading substantiated by 2(A):

	Basic empty weight	2085	X_E	278464
1.	Oil 5 gals.	38	51	1938
2.	Fuel 70 gals.**	420	90	37800
3.	Pilot***	170	90	15300
3.	Passengers (in front seat)	170	90	15300
3.	Parachutes in front seats (2 at 20 lbs.)	40	90	3600
4.	Propeller (heaviest to be used)(70-46)	24	13	312
6.	Optional instruments	15	60	900
7.	Wheel streamlines	24	71	1704
21.	Radio equipment forward of most forward C.G. limit Plus other items of optional equipment critical for most forward C.G. load condition for which approval as Class III equipment is desired.			
	Totals	W_F	X_F****	M_F

NOTES ARE PERTINENT TO BOTH SECTION 2 AND 3.

* Actual weight of pilot and parachute shall be used in Sections 2(A) and 3(A) instead of standard weight of 190 lbs. (170 + 20).

** Fuel substantiated shall be as follows: (See CAR 04.7211)
(a) 1 gal. for every 12 MAXIMUM EXCEPT TAKE-OFF horsepower when minimum fuel is critical.
(b) Full tanks when maximum fuel is critical.

*** When controls are arranged in tandem and the aircraft can be flown from either position, Section 2(B) will include the pilot in the front cockpit. Similarly, Section 3(B) for the most rearward C.G. condition will include the pilot in the rear cockpit. (Otherwise the airplane must be placarded accordingly).

**** Shall not exceed limits in 2(A) and 3(A).

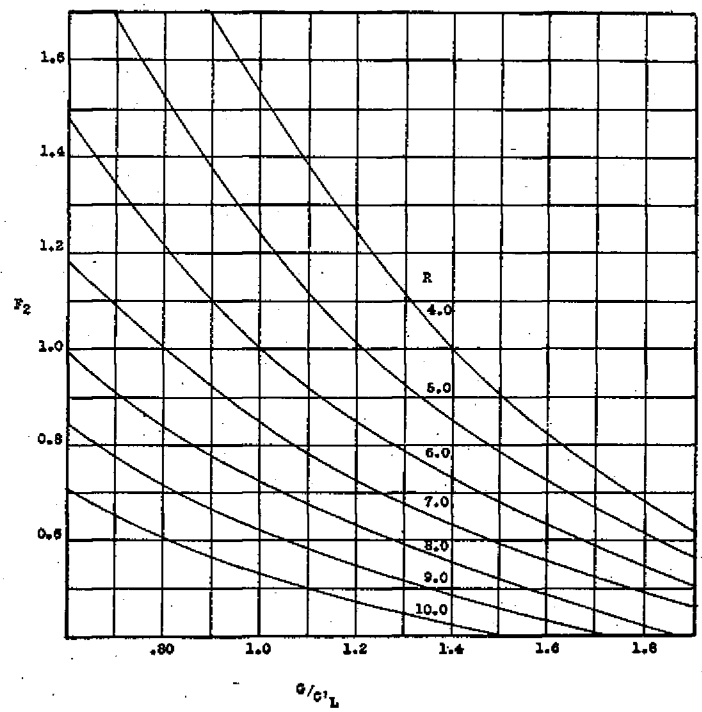


FIG III-4

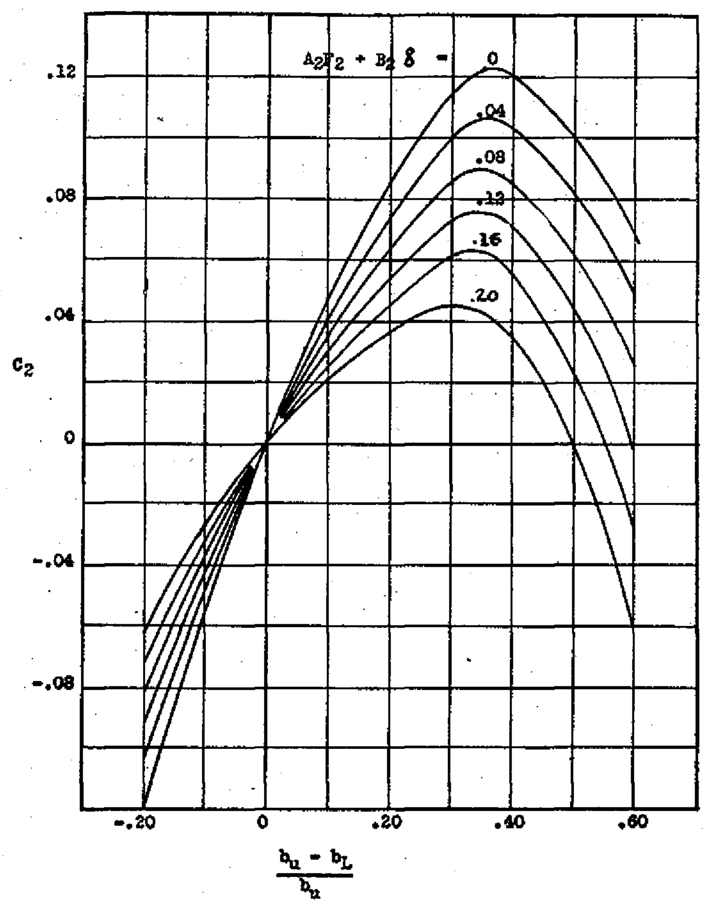
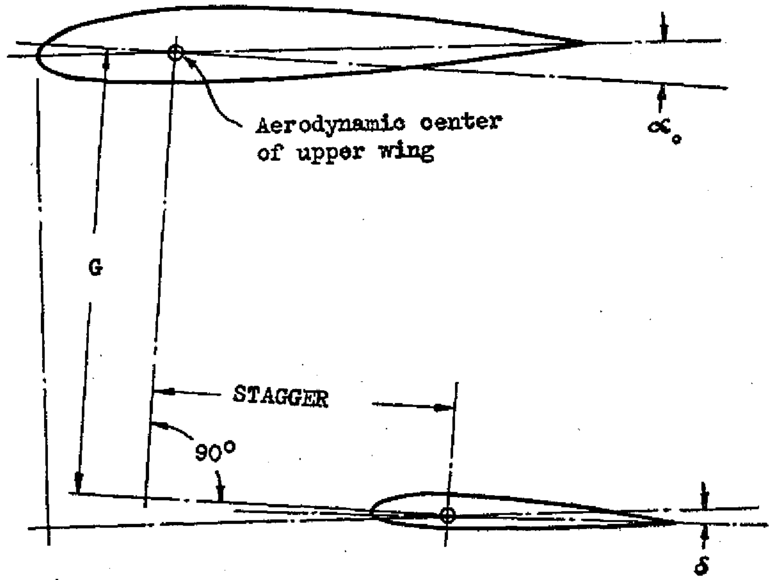


FIG III-5



α_o is weighted average angle of zero lift

FIG. III-6

APPENDIX IV

(To be supplied)

APPENDIX V

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

1. SUMMARY

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

2. BASIC CONSIDERATIONS

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

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

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

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

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

* See Section 8 for nomenclature.

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

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

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

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

The computation of c_{la_1} and c_{lb} is conveniently adapted to a tabular form, the use of which is described in the following section:

3. USE OF TABULAR FORM

The tabular form for computing the values of c_{la_1} and c_{lb} is shown as Table 1. Briefly, the use of this Table consists of entering the basic geometrical data required in columns 1, 2, 3, and 16; entering the "multiplier" in column 4; entering the basic aerodynamic data required in columns 6 and 17; and then proceeding with the routine computations as indicated in the body of the Table. The value of c_{la_1} is then obtained as column 15. The value of c_{lb} is obtained as column 23, in case high lift devices are not used, and as column 26 if such devices are used. The procedure for using this Table will now be outlined:

Column 1

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

Column (2)

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

Column (3)

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

Column (4)

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

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

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

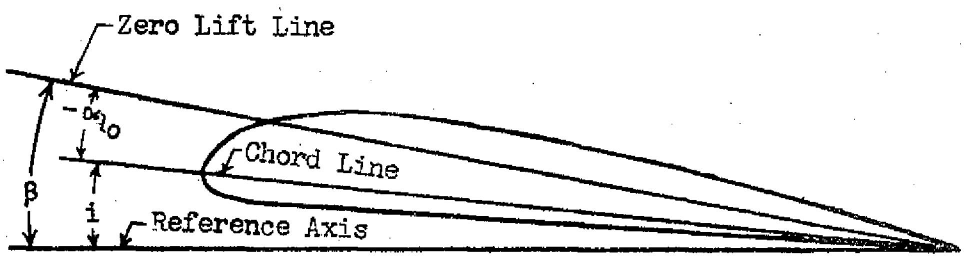
(See also Fig. 3 for an example of this procedure.)

Column (6)

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

Column (16)

Enter here the angle of incidence, i. This is the angle between the chord line (the line used as datum for airfoil ordinates and angles) and the reference axis. The reference axis can be chosen as any convenient axis in the plane of symmetry, such as, the fuselage axis or the chord line of the root chord. Care should be taken in using the proper sign for i, positive values being as measured in the sketch below. (The sign of α_{l_0} is shown as negative in the sketch to agree with NACA airfoil data where the reference line for angles of attack is always the chord line. Therefore, considering only the geometry of the particular sketch, β is obviously $i + \alpha_{l_0}$ or using the sign and expression for β given on the sketch, $\beta = i - (-\alpha_{l_0}) = i + \alpha_{l_0}$).



$$\beta = i - \alpha_{l_0}$$

Column (17)

Enter here the angle of attack for zero lift, α_{l_0} for the pertinent airfoil section (or airfoil-flap combination), <u>taking care to use the proper sign</u>. Data for this purpose can be obtained from standard NACA reports. Computations can now proceed in accordance with the instructions on Table 1. In cases where high lift devices are not used, the final values of $c_{l_{a_1}}$ and c_{l_b} for design purposes are given in Columns (15) and (23), respectively.

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

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

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

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

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

5. COMPARISON EXAMPLES

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

Example #1

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

Example #2

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

Example #3

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

Example #4

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

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

6. CONCLUSIONS

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

1. The aspect ratio is within the normal range of values (say from 5 to 12), and,

2. The wing has reasonably rounded tips, if the taper ratio is greater than 0.5. (This restriction as to rounded tips does not apply for taper ratios less than .5.)

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

7. REFERENCES

1. Schrenk, O.: A Simple Approximation Method for Obtaining the Spanwise Lift Distribution. T. M. 948, N. A. C. A., 1940.

2. Pearson, H.A.: Span Load Distribution for Tapered Wings with Partial-Span Flaps. T. R. 585, N. A. C. A., 1937

3. Sherman, Albert: A Simple Method of Obtaining Span Load Distribution. T. N. 732, N. A. C. A., 1939.

4. ANC 1 (1): Spanwise Air-Load Distribution. 1938.

8. NOMENCLATURE

S Wing area, square inches

b Span, inches

c Chord, inches

$\bar{c}$ Average chord, inches ($= S/b$)

y Distance of a particular station from centerline of wing, inches

C_L Wing lift coefficient

$c_{l_{a_1}}$ Additional lift coefficient for a section when wing $C_L = 1.0$

c_{l_a} Additional lift coefficient for a section ($= C_L c_{l_{a_1}}$)

c_{l_b} Basic lift coefficient for a section due to aerodynamic twist, when wing is operating at zero lift

c_l Section lift coefficient ($= c_{l_a} + c_{l_b}$)

α_{l_o} Angle of attack of a section for zero lift, degrees

i Angle between the chord line and the reference axis, degrees (see sketch on page 6)

β Angle between the zero lift line and the reference axis, degrees (see sketch on page 6; note that $\beta = i - \alpha_{l_o}$)

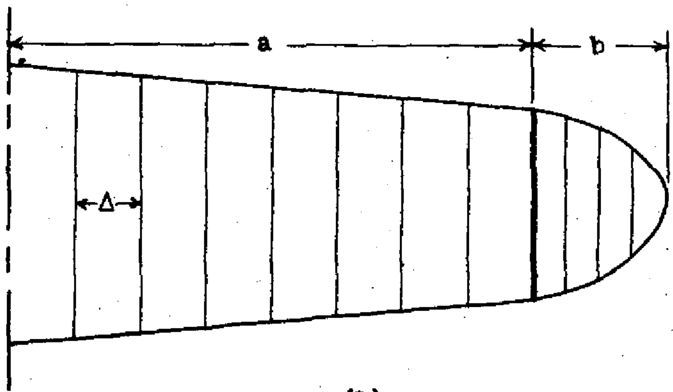
α_R Angle between reference axis and the wind direction, degrees (positive when the reference axis is so inclined to the wind direction as to produce positive lift, assuming (for this purpose only) that the reference axis acts as a zero lift chord line on airfoil section)

α_{R_o} Angle between the wind direction and the reference axis when the wing is operating at zero lift, degrees

α_{a_s} Angle between the zero lift line of a section and the wind direction, degrees ($\alpha_{a_s} = \alpha_R + \beta$)

a Lift curve clope, C_L/degree

a_o Section lift curve slipe, c_l/degree (slope of graph of c_l vs. α)

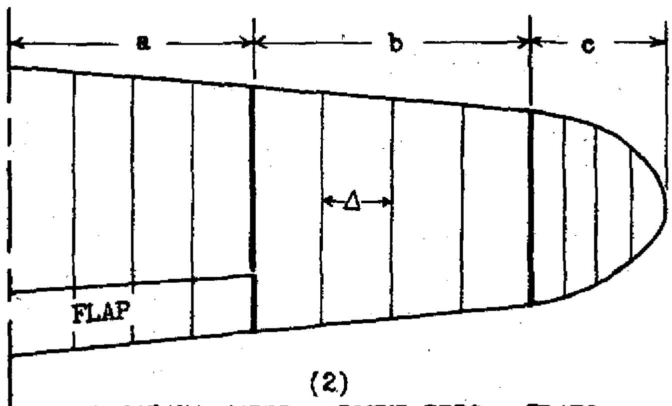


SUGGESTED DIVISIONS

a = 8 even parts

b = 4 even parts

(1)
STRAIGHT TAPER - ROUND TIPS

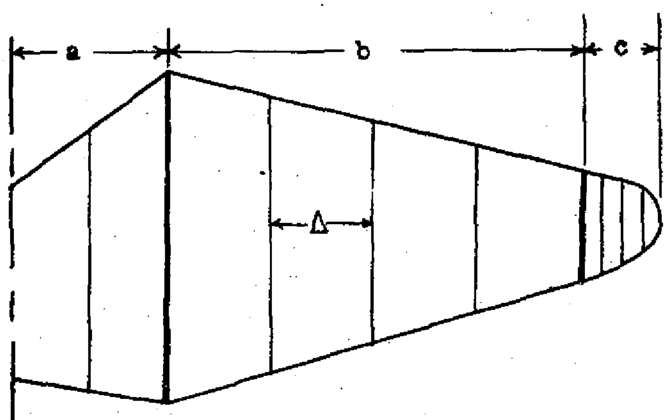


a = 4 even parts

b = 4 even parts

c = 4 even parts

(2)
STRAIGHT TAPER - ROUND TIPS - FLAPS



a = 2 even parts

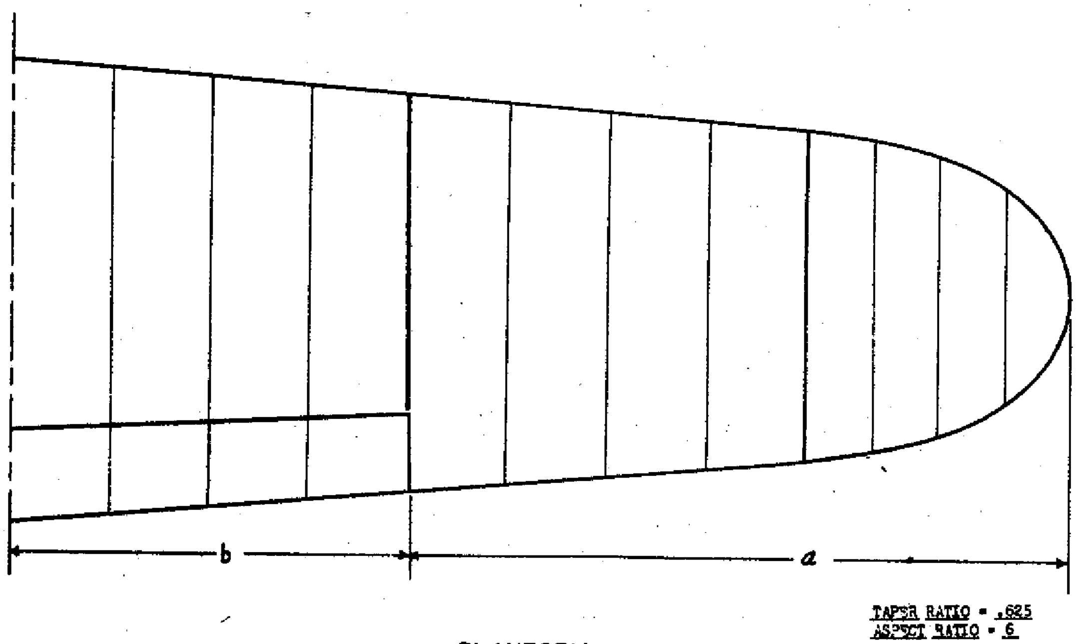
b = 4 even parts

c = 4 even parts

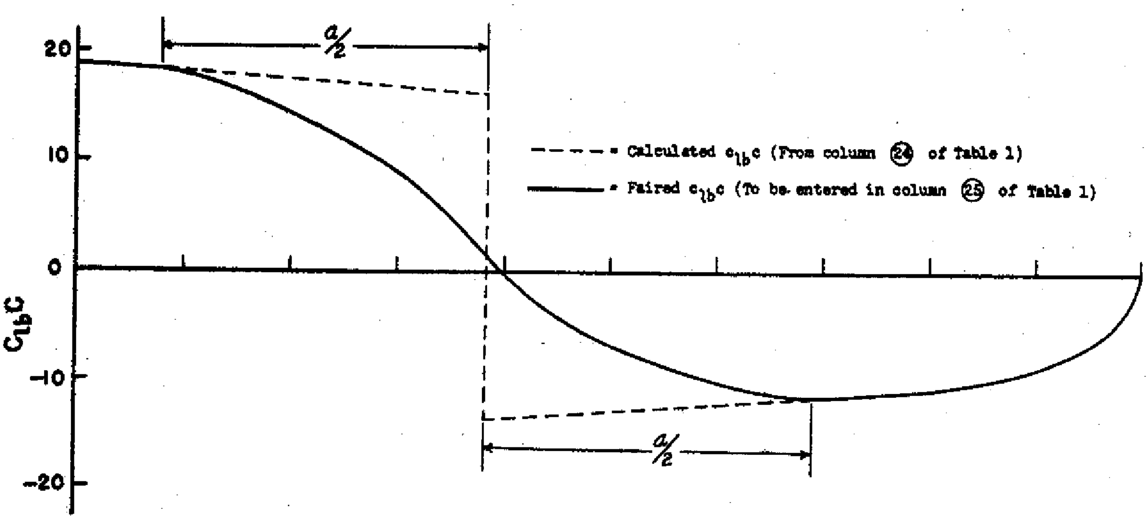
(3)
REVERSE TAPER - ROUND TIPS

EXAMPLES OF WING PLANFORM DIVISION

FIG. 1



PLANFORM



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

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

FIG. 2

SPANWISE AIR-LOAD DISTRIBUTION

Section	1	2	3	4	5	6	7	8	9	10	11	12	13
	$y/_{b/2}$	c - ins.	A - ins.	Multi-plier	2 x 3 x 4	a_o C_L per degree	*(1) 5 x 6	*(2) $6/\bar{a}_o$	$\frac{4\bar{c}}{\pi}/2$	1^2	1 - 10	$\sqrt{11}$	9 x 12
Section From ℄ of Airplane to End of Flap	.000	102.00	23.04	.333	783.36	.100		1.000	.996	.000	1.000	1.000	.996
	.0960	98.70	23.04	1.333	3032.06	.100		1.000	1.030	.009	.991	.995	1.025
	.1920	94.69	23.04	.667	1454.44	.100		1.000	1.079	.037	.963	.981	1.054
	.2880	91.02	23.04	1.333	2796.13	.100		1.000	1.112	.083	.917	.957	1.069
	.3840	87.80	23.04	.333	674.31	.100		1.000	1.158	.147	.853	.924	1.070
Section From End of Flap to Beginning of Tip	.3840	87.80	21.96	.333	642.70	.100		1.000	1.158	.147	.853	.924	1.070
	.4755	83.95	21.96	1.333	2458.06	.100		1.000	1.211	.226	.779	.880	1.066
	.5670	80.40	21.96	.667	1177.06	.100		1.000	1.265	.321	.679	.824	1.042
	.6585	76.87	21.96	1.333	2250.75	.100		1.000	1.323	.434	.566	.752	.995
	.7500	73.38	21.96	.333	537.14	.100		1.000	1.386	.563	.437	.661	.916
Section From Beginning of Tip to Extreme Tip of Wing	.7500	73.38	15.00	.333	366.90	.100		1.000	1.386	.563	.437	.661	.916
	.8125	69.18	15.00	1.333	1383.60	.100		1.000	1.470	.660	.340	.583	.857
	.8750	62.27	15.00	.667	622.70	.100		1.000	1.633	.766	.234	.484	.790
	.9375	49.51	15.00	1.333	990.20	.100		1.000	2.054	.879	.121	.348	.715
	1.000	.000	15.00	.333	.000	.100		1.000	—	1.000	.000	.000	—

Σ 5 = 19,169.41 Σ 7 =

Section	14	15 Final c_{la_1}	16	17	18	19	20	21	22	23 Computed c_{lb}	24 Computed $c_{lb}c$	25 Faired $c_{lb}c$	26 Final c_{lb}
	8 + 13	14/2	*(3) i	α_{lo}	16 - 17	*(4) 5 x 18	*(5) 7 x 18	6/2	18 + α_{R_0}	*(6) 21 x 22	2 x 23	*(7) $c_{l_b}c$	25/2
Section From ℄ of Airplane to End of Flap	1.997	.998	.000	-8.00	8.00	6266.88		.050	3.20	.185	18.87	18.87	.185
	2.025	1.012	.000	-8.00	8.00	24256.48		.050	3.20	.185	16.20	18.00	.182
	2.054	1.027	.000	-8.00	8.00	11635.52		.050	3.20	.185	17.51	14.70	.155
	2.069	1.034	.000	-8.00	8.00	22,369.04		.050	3.70	.185	16.83	9.70	.107
	2.070	1.035	.000	-8.00	8.00	5,394.48		.050	3.70	.185	16.24	1.25	.019
Section From End of Flap to Beginning of Tip	2.070	1.035	.000	-1.20	1.20	771.24		.050	-3.10	-.155	-13.60	1.25	.014
	2.066	1.033	.000	-1.20	1.20	2949.67		.050	-3.10	-.155	-11.01	-5.15	-.068
	2.042	1.021	.000	-1.20	1.20	1412.47		.050	-3.10	-.155	-12.46	-9.25	-.115
	1.995	.997	.000	-1.20	1.20	2700.90		.050	-3.10	-.155	11.91	11.40	-.148
	1.916	.958	.000	-1.20	1.20	644.57		.050	-3.10	-.155	-11.37	-11.37	-.155
Section From Beginning of Tip to Extreme Tip of Wing	1.916	.958	.000	-1.20	1.20	440.28		.050	-3.10	-.155	-11.37	-11.37	-.155
	1.857	.928	.000	-1.20	1.20	1660.32		.050	-3.10	-.155	-10.72	-10.72	-.155
	1.790	.895	.000	-1.20	1.20	747.24		.050	-3.10	-.155	-9.65	-9.65	-.155
	1.715	.857	.000	-1.20	1.20	1188.24		.050	-3.10	-.155	-7.67	-7.67	-.155
	—	—	.000	-1.20	1.20	.000		.050	-3.10	-.155	.000	.000	.000

Σ 19 = 82,437.33 Σ 20 =

*NOTES:

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

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

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

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

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

(6) Values taken from Column 25 are final values of c_{lb} when no high-lift device is operating.

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

CALCULATIONS FOR CONSTANTS

b/2 = 240 ins.

S (total wing area) = $2\Sigma 5 = 2 \times 19{,}169.41$ 38,338.82 sq. ins.

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

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

$\bar{a}_o = \frac{\Sigma 7}{\bar{c}\, b/2} =$ C_L per degree

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

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

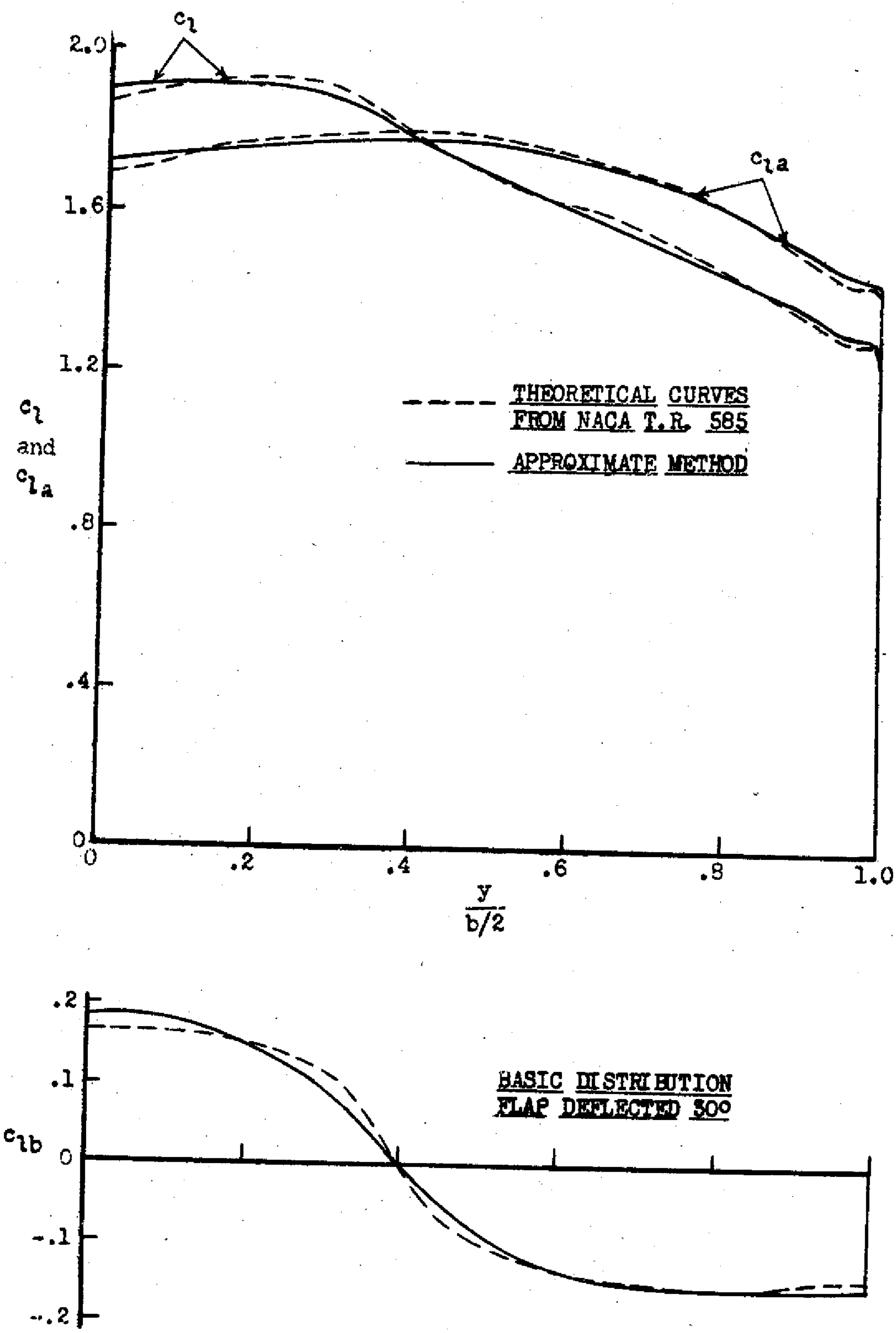
TABULATION FOR WING SHOWN IN FIG. 2

FIG. 3

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

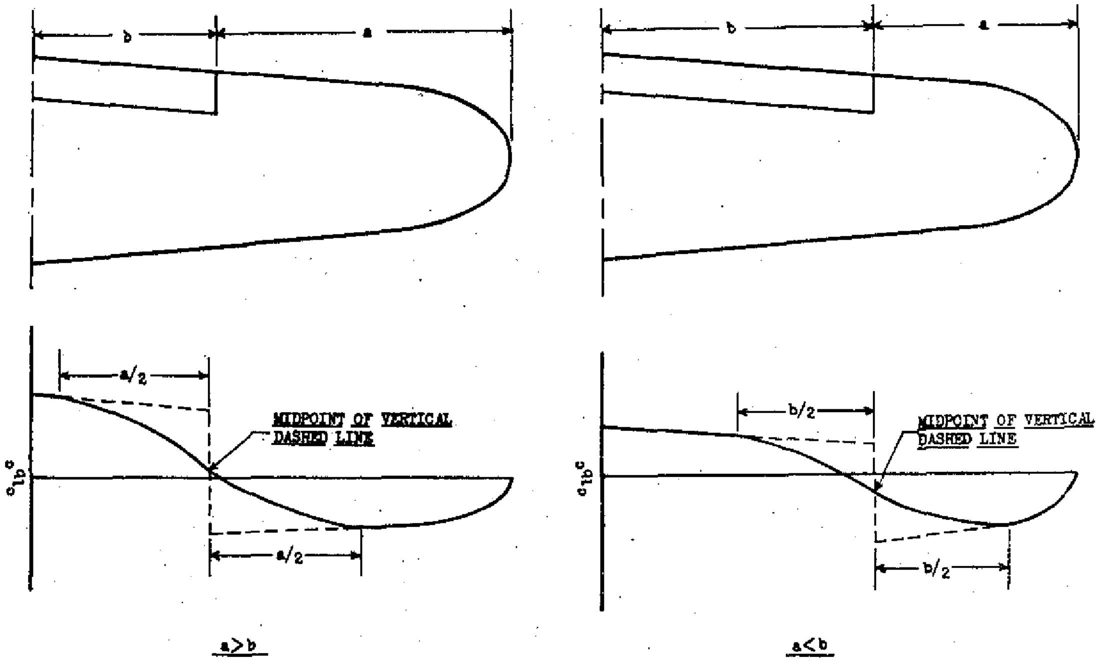
EXAMPLE FROM N.A.C.A. T.R. 585
FINAL c_l DISTRIBUTION FOR A WING C_L of 1.72

FIG. 4

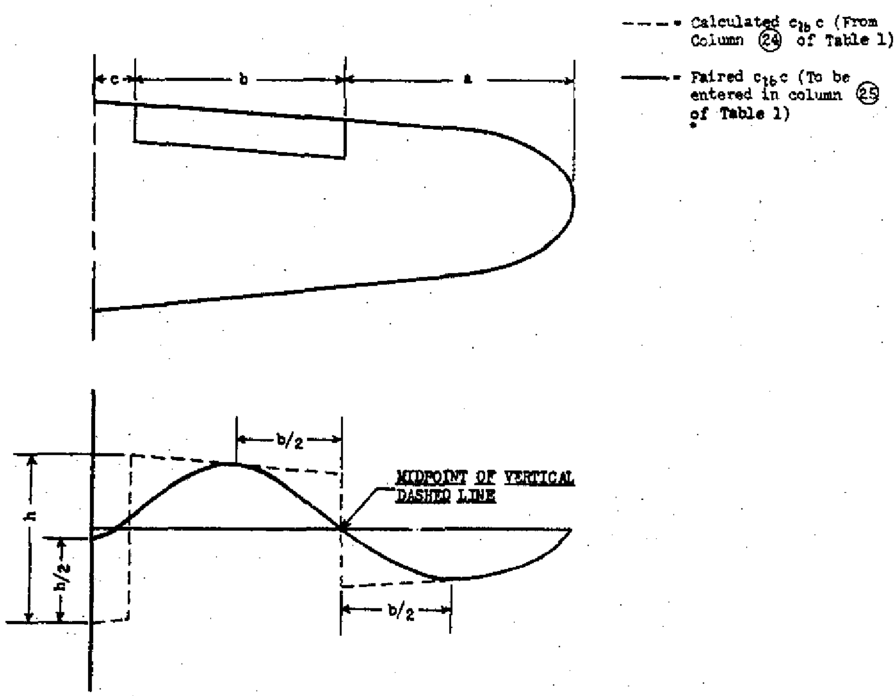


COMPARISON EXAMPLE FROM N.A.C.A. T.R.585

FIG. 5



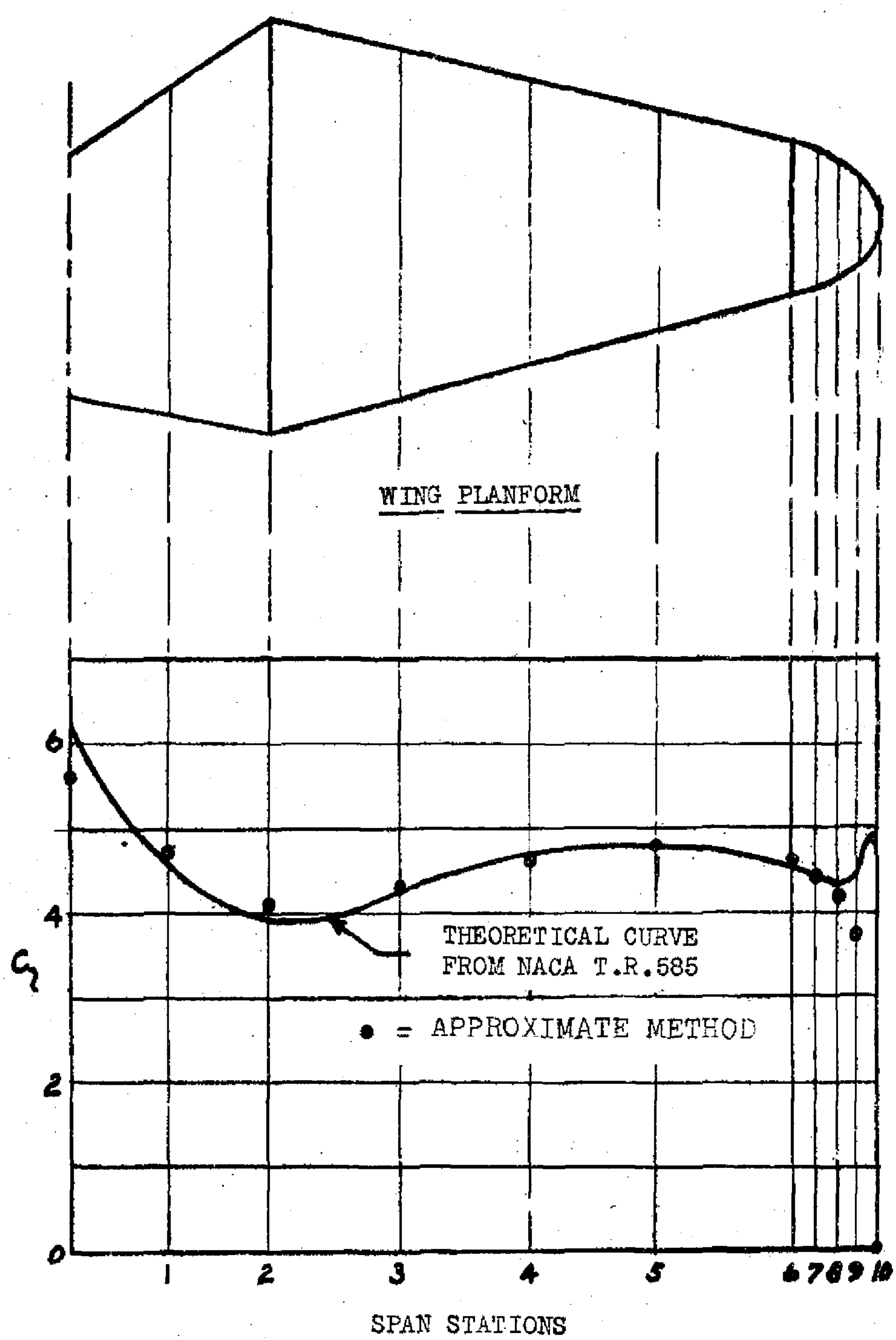
FLAP EXTENDING OUT FROM ℄



FLAP EXTENDING OUT FROM SIDE OF FUSELAGE

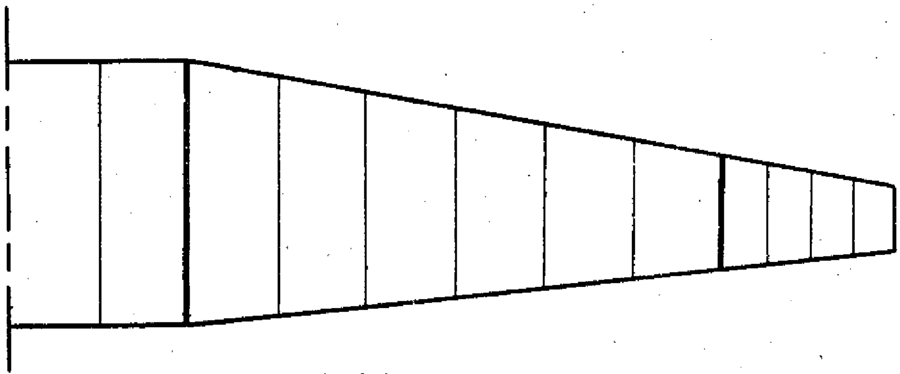
EXAMPLES OF FAIRING THE CURVE OF $C_{lb}C$ VS. $\frac{Y}{b/2}$

FIG. 6

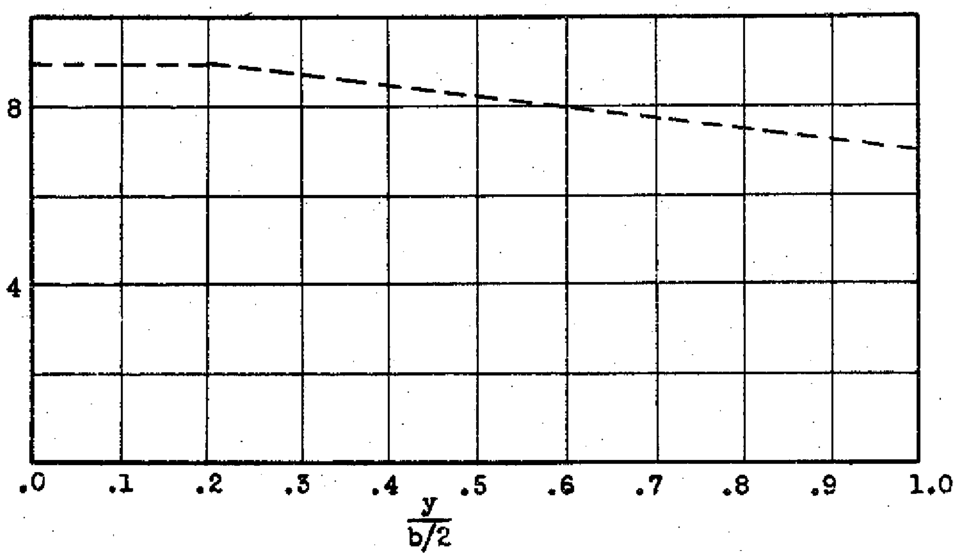


WING WITH REVERSE TAPER
COMPARISON EXAMPLE FROM T.R. 585

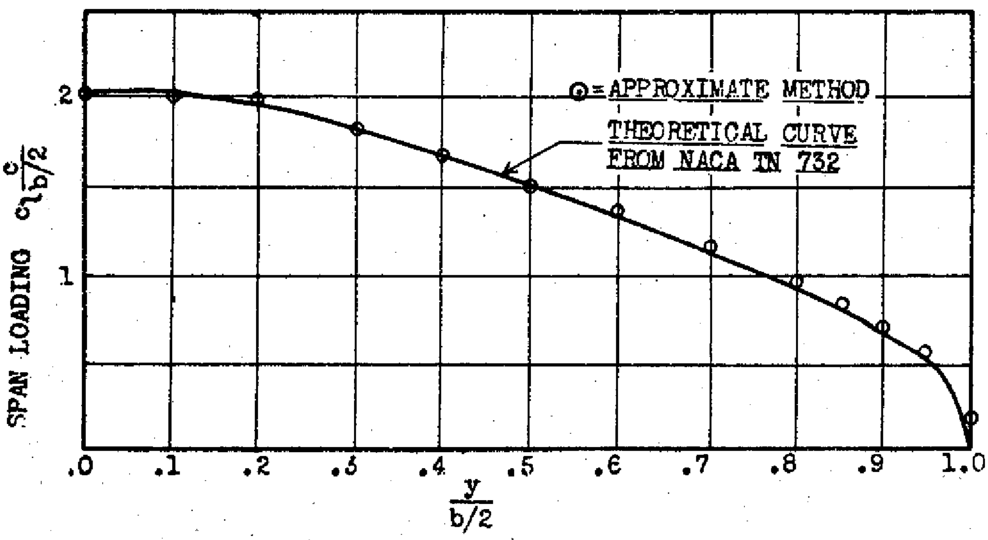
FIG. 7



(a) PLANFORM



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



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

FIG. 8

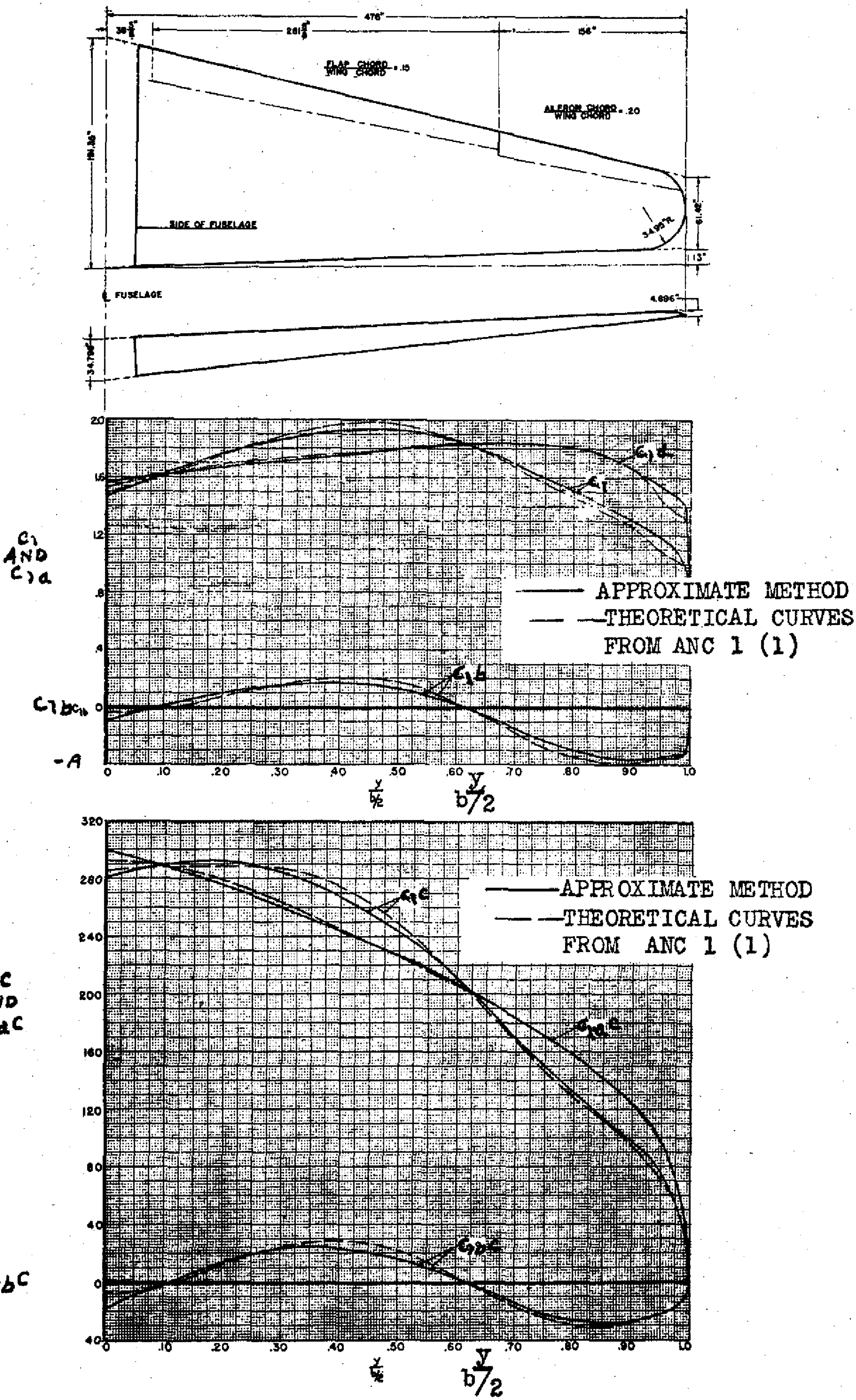


FIG. 9

COMPARISON EXAMPLE FROM ANC-1(1)

SPANWISE AIR-LOAD DISTRIBUTION

														Final $c_{l a_1}$								Computed c_{lb}	Computed $c_{lb}c$	Faired $c_{lb}c$	Final c_{lb}
1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26
$y/_{b/2}$	c - ins.	Δ - ins.	Multi-plier	2 x 3 x 4	a_o C_L per degree	*(1) 5 x 6	*(2) $6/\bar{a}_o$	$\frac{4\bar{c}}{\pi}/2$	1^2	1 - 10	$\sqrt{11}$	9 x 12	8 + 13	14/2	*(3) i	α_{lo}	16 - 17	*(4) 5 x 18	*(5) 7 x 18	6/2	18 + α_{RO}	*(6) 21 x 22	2 x 23	*(7) $c_{l_b}c$	25/2
			Σ5 =		Σ7 =												Σ19 =		Σ20 =						

*NOTES:

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

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

(3) See sketch page 6 The signs of the angles should be carefully observed.

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

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

(6) Values taken from Column 23 are final values of c_{lb} when no high-lift device is operating.

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

CALCULATIONS FOR CONSTANTS

b/2 =	ins.	$\bar{a}_o = \frac{\Sigma 7}{\bar{c}\ b/2}$ =		C_L per degree
S (total wing area) = 2Σ5 =	sq. ins.			
$\bar{c}$ (mean chord) = $\frac{\Sigma 5}{b/2}$ =	ins.	$\alpha_{R_0} = \frac{-\Sigma 19}{\bar{c}\ b/2}$ =		degrees [this value of α_{R_0} used only when (a = constant)]
$\frac{4\bar{c}}{\pi}$ =	ins.	$\alpha_{R_0} = \frac{-\Sigma 20}{\Sigma 7}$ =		degrees [this value of α_{R_0} used only when (a = variable)]

TABLE 1

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