CIVIL AERONAUTICS AUTHORITY

WASHINGTON, D. C.

CIVIL AIR REGULATIONS

PROPOSED PART 05

GLIDER AIRWORTHINESS

June 1, 1940

PART 05.-GLIDER AIRWORTHINESS

AIRCRAFT AIRWORTHINESS SECTION

Supersedes IP 12.

REPORT NO. 17

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SUMMARY OF INFORMATION PERTAINING TO GLIDERS

Object. To summarize the Federal regulations and interpretations pertaining to the registration and identification, certification (formerly referred to as licensing), and operation of gliders. Unless otherwise specified, the publications referred to herein may be obtained from:

Publications & Statistics Division
Civil Aeronautics Authority
Washington, D. C.

A. REGISTRATION AND IDENTIFICATION.

1. In accordance with the Civil Air Regulations, Parts 01 and 60, as amended 1940, all gliders must be registered with the Authority and must display identification marks assigned and issued by the Authority. Copies of the necessary application form may be obtained from any of the representatives of the Authority listed in Part C below, or from:

Chief, Certificate Section
Civil Aeronautics Authority
Washington, D. C.

It should be noted that technical data (structural analyses, drawings, etc.) are not required for registration and identification.

B. AIRWORTHINESS CERTIFICATES.

1. Airworthiness certificates for the 'C' or 'R' classification can not be obtained for a domestic glider unless (1) the glider airworthiness requirements (CAR 05) have been complied with, (2) the glider is of a type approved under previous regulations, or (3) the glider has been previously certificated. In any case, the glider must satisfactorily pass an inspection conducted by personnel of the Authority.

2. Drafts of the glider airworthiness requirements, designated as Part 05 of the Civil Air Regulations, and the corresponding Civil Aeronautics Authority Manual, CAAM 05, are now available for distribution. Copies can be obtained from the Authority.

3. When an airworthiness certificate is desired for a home-built glider of domestic design the procedure outlined below should be followed:

a. Procedure for Designer and Builder of Basic Model.

(1) The original designer, manufacturer, or holder of the type certificate (if one has been issued) should submit an application for approval, of the glider for home-building. This application should indicate the extent to which home-fabrication is to be employed, according to the following general classification:

(a) Assembly of major structural components from prefabricated assembly kit;

(b) Building of structure from pre-cut members supplied in the form of a construction kit;

(c) Complete building of structure by home-builder from approved drawings.

(2) If the design has not previously been granted an approval by means of an aircraft specification (which makes the model eligible for certification subject to inspection) complete technical data (drawings and structural analysis) must be submitted in accordance with the procedure outlined in CAR 05, "Glider Airworthiness". A complete file of the drawings and instructions to be supplied to the home-builder must be submitted. (These will usually include all drawings required in CAR 05.) Drawings supplied to the home-builder must contain complete information as to material specifications, correct shop practices and other information considered necessary to insure proper construction by the home-builder. In this connection, drawings of primary structural parts which call for fabrication processes requiring special training, (such as welding), should state that such operations shall be performed by properly qualified persons, the qualifications to be determined by the Authority in accordance with the neture of the construction. In certain cases, it is considered necessary that such qualifications be satisfactorily demonstrated to a representative of the Authority.

(3) If the model has previously been issued an aircraft specification, it is necessary to submit only the drawings and instructions which are to be supplied to the home-builder

These conform to the requirements outlined in the above paragraph. Deviation from the original design may require additional structural investigation, such as structural analysis and/or static tests.

(4) The data submitted will be examined by the Aircraft Airworthiness Section for conformity with the airworthiness requirements and to determine the suitability of the drawings and details of construction for home-assembly or home-building, considering the extent to which prefabrication is to be employed. It is necessary that the first glider constructed in accordance with the drawings and instructions to be furnished the home-builder be inspected, in its completely assembled form, for workmanship, materials and conformity, before the covering operation. This inspection will be authorized upon the completion of a preliminary examination of the data. This procedure is adopted to allow the manufacturer to complete the first article while the examination of the technical data is being completed. See paragraph (6) below regarding further inspections and tests.

(5) As indicated in paragraph (4) it is necessary for the original designer to construct, or to have constructed, at least one complete glider in accordance with the drawings for which approval is desired. This construction incorporates all prefabricated parts which will be supplied to the home-builder. In some cases in which approval of several different classes of home-fabrication (such as outlined in paragraph (1)) are desired, it may be necessary to construct certain portions of the glider in several different ways, unless all classes can be covered in one article

(6) After satisfactory completion of the first inspection and all required static tests (paragraph (4)) the glider may be covered and a final engineering inspection and flight test authorized. Upon satisfactory completion of these inspections an aircraft specification will be issued, (or the original aircraft specification modified if the design has previously been approved). The aircraft specification will specify the extent to which home-building is approved for the particular model and provide for the certification of subsequent gliders of such model, subject to compliance with certain inspection procedures, outlined below.

(7) When the kits are to be partially prefabricated, the manufacturer of such kits must demonstrate, through a factory inspection, that he possesses the facilities necessary to insure the production of kits of exact similarity and in exact agreement with the approved drawings.

b. Procedure for Home-Builder of an Approved Model (TC)

(1) The home-builder must construct his glider in exact accordance with the approved drawings and to use the parts and materials included in the approved assembly or construction kit, if such a kit is supplied. An affidavit to this effect must be submitted to the Authority when the first request for an inspection is made. Any deviation from the approved drawings or any substitution for parts included in an approved kit is considered by the Authority as an alteration, in which case complete substantiating technical data are required. When the glider has been completely assembled and before any covering operations have been completed the home-builder should contact the local Civil Aeronautics Authority representative and request an inspection of the glider.

(2) A Civil Aeronautics Authority representative will inspect each home-built glider, in its completely assembled form for workmanship, materials and conformity, before the covering operation. In some cases this inspection may include certain proof tests, depending on the nature of the construction. Such tests will have been decided upon in the original examination of the drawings and will be definitely specified in the instructions to the home-builder. Another inspection will be conducted after covering. Upon the satisfactory completion of these inspections, and after satisfactory completion of a demonstration flight by a pilot holding an appropriate and currently effective airman certificate witnessed by a representative of the Authority, the glider will be eligible for certification as to airworthiness.

c. Procedure for Home-Builder of Single Glider not of a Previously Approved Model.

(1) If only one glider is involved, a type certificate need not be secured as operation can be authorized on the basis of an airworthiness certificate only. Such a certificating procedure will usually require the minimum amount of technical data to be submitted (see CAR 05.031).

4. When an airworthiness certificate is desired for a home-built glider of approved foreign design, the following procedure should be followed:

a. Evidence of approval of the type design by the proper foreign airworthiness authorities must be secured. This record will be accepted as proof of strength.

b. A statement from the pertinent foreign manufacturer, certifying that the drawings from which the home-built glider is to be (or was) constructed conform to the type design, must be submitted to the Authority.

c. A complete set of the above drawings, together with a drawing list must be submitted.

d. It must be shown that the materials used in constructing the glider are equivalent to, as regards mechanical properties, etc., or better than the materials specified on the original manufacturer's drawings.

e. Upon satisfactory completion of the inspections and tests specified in 3, b above the glider will be eligible for certification.

5. It should be noted that insofar as the Civil Aeronautics Authority is concerned, an airworthiness certificate (see Part C) is not required for the intra-state operation of gliders, provided that the gliders are not flown within the limits of a civil airway or a control zone of intersection. An identification mark is, however, required (see Part A). The details of these requirements are set forth in Part 60 of the Civil Air Regulations, particularly in Section 60.32. It should also be noted that there are no additional Federal Regulations governing the inter-state operation of non-commercial gliders. A large number of States, however, do require all aircraft to have an airworthiness certificate. It is suggested that detailed information on this point be obtained from the pertinent State Aeronautics Authorities.

6. Operation

a. In accordance with the Civil Air Regulations, Parts 01 and 60, experimental operation may be authorized for gliders under the following circumstances:

(1) When the glider is to be used solely for experimental purposes,* and has no apparent unairworthy features. In general, such certification will not authorize airplane towing, unless sufficient satisfactory technical data have been submitted to the Aircraft Airworthiness Section of the Authority to determine safe towing speeds. It should be noted that in experimental operations only personnel essential to the purposes of the flight are to be carried.

(2) In the case of national glider meets which are witnessed by an inspector of the Authority, experimental operation may be authorized for the gliders in the meet for the duration of the meet, provided that the gliders pass the necessary inspections (see (1) above). In certain cases such certificates will permit airplane towing.

b. If the experimental operation is desired, one of the following representatives of the Authority should be contacted for further information:

John E. Sommers, Regional Manager,
Civil Aeronautics Authority,
P. O. Box 449, Newark, New Jersey.

R. C. Copeland, Regional Manager,
Civil Aeronautics Authority,
P. O. Box 4327, Atlanta, Georgia.

Harold R. Neely, Regional Manager,
Civil Aeronautics Authority,
1204 New Post Office Building,
Chicago, Illinois.

L. C. Elliott, Regional Manager,
Civil Aeronautics Authority,
P. O. Box 1689, Fort Worth, Texas.

Leonard W. Jurden, Regional Manager,
Civil Aeronautics Authority,
Eighth Floor, City Hall Building,
Kansas City, Missouri.

* Note. As used herein, the term "experimental" when applied to an aircraft denotes that such aircraft is certificated for experimentation in flight with a view to determining or improving its characteristics or those of its components or accessories, and that inspection has disclosed no unairworthy feature of such aircraft with respect to structural integrity, workmanship or flight characteristics.

J. S. Marriott, Regional Manager,
Civil Aeronautics Authority,
P. O. Box 1010, Santa Monica, Cal.

R. D. Bedinger, Regional Manager,
Civil Aeronautics Authority,
King County Airport, Seattle, Wash.

C. AIRMEN CERTIFICATES. (Formerly called pilot licenses)

1. The prerequisites for the issuance of airmen certificates are set forth in Part 20 of the Civil Air Regulations, particularly Sections 20.15, 20.16, and 20.17. It should be noted that, insofar as the Civil Aeronautics Authority is concerned, airmen certificates are not required for the operation of uncertificated gliders. For information regarding State Regulations, it is suggested that the pertinent State Aeronautics Authorities be contacted.

D. DRAWINGS OF GLIDERS.

1. The Authority does not have any drawings of gliders available for distribution.

05.0 General.

05.00 Scope. Pursuant to the provisions of the Civil Aeronautics Act of 1938, empowering and requiring the Civil Aeronautics Authority to prescribe such minimum standards governing the design, materials, workmanship, construction, and performance of aircraft as may be required in the interest of safety, and to provide for the rating of aircraft as to airworthiness, the requirements hereinafter set forth shall be used as the minimum standards for establishing such rating of gliders.

05.000 Airworthiness certificate. The general requirements for the issuance of an airworthiness certificate, and other information concerning such certificates, are set forth in Part 01. The airworthiness requirements which shall be used as a basis for the certification of gliders are specified hereinafter. (See § 05.003).

05.001 Type certificate. The general requirements for the issuance of a type certificate are set forth in Part 01. In addition to the requirements hereinafter specified for an airworthiness certificate, the special requirements designated as (TC) shall apply when a type certificate is sought.

05.002 Production certificate. The requirements for the issuance of, and the procedure for obtaining, a production certificate are set forth in Part 01.

05.003 Deviations. These requirements are based on the present development in the science of glider design and apply only to conventional types. New types of gliders and new types of construction may, however, incorporate features to which these requirements cannot be logically applied. In such cases, special consideration will have to be given to the particular new problems involved. In cases where the deviation from conventional practice is small, approval may be granted if sufficient evidence is submitted to show that the proposed deviation will not be detrimental to the airworthiness of the design. When the deviation from conventional practice is considerable, special rulings covering the feature(s) in question shall be obtained from the Authority. Insofar as these requirements are concerned, acrobatic, seaplane, and amphibian gliders are considered as new types.

05.01 Classification of gliders. For the purpose of applying these requirements, gliders are classified on the basis of certain imposed operating restrictions, as outlined in Table 05-1.

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TABLE 05-1 - GLIDER CLASSIFICATIONS

Class	Type of Glider	Minimum "Never Exceed" Placard Speed m.p.h.	Mimimum Auto-Winch Tow Placard Speed m.p.h.	Airplane Tow	Blind Flying
I	High performance Intermediate Utility	80	50	Permitted	See Note 1
II	High performance Intermediate Utility Primary	65	50	Permitted	Not Permitted
III	Utility Primary	50	45	Not Permitted	Not Permitted

Note 1. Blind flying will be permitted if § 05.510 is complied with.

05.02 Airworthiness requisites. As a basis for an airworthiness rating, the suitability of gliders with respect to the following factors shall be demonstrated in a manner satisfactory to the Authority:

(a) The structural strength of wing, tail and control surfaces, fuselage, fittings, control systems and landing gear.
(b) Pilot compartment, cabin and control arrangements.
(c) Equipment and instruments.
(d) Design details.
(e) Products, materials, construction, and workmanship.
(f) Flying characteristics and qualities.
(g) Safety features.

05.03 Technical data required. When technical data are submitted as a basis for an airworthiness rating, they shall include information which, in conjunction with suitable inspection and test procedure, will enable the Authority to establish such rating.

05.030 Submission to branch office. When data are submitted to a branch office of the Authority, extra copies of the three-view drawing, correspondence, and applications shall be included for the Washington office files.

<u>05.031 Data required when airworthiness certificate only is sought</u>. The minimum data required as a basis for the issuance of an airworthiness certificate only are as follows:

(a) A three-view drawing of the glider, to a designated scale, specifying the external dimensions, manufacturer's designation, gross weight, empty weight, wing and control surface areas, seating arrangement, baggage and ballast capacities (in pounds) and equipment supplied.

(b) Such additional technical data as are deemed necessary by the Authority to show compliance with the requirements of § 05.02 or their equivalent.

<u>05.032 Data required for type certificate (TC)</u>. As a basis for the issuance of a type certificate, the following technical data shall be submitted in a satisfactory form:

(a) Drawings.
(b) Drawing list (in duplicate).
(c) Equipment lists.
(d) Preliminary weight and balance report.
(e) Balance diagram.
(f) Weight table.
(g) Structural report (structural analysis and/or test reports).

<u>05.04 Procedure when airworthiness certificate only is sought</u>. (See also Part 01).

<u>05.040 General</u>. The applicant shall prove the suitability of the glider with respect to the airworthiness requisites listed in § 05.02 or their equivalent, to the satisfaction of the Authority.

<u>05.041 Structural inspection</u>. An official representative of the Authority will conduct all necessary inspections of the structure and methods of fabrication prior to completion of the glider and will witness structural tests in compliance with these regulations. (See § 05.333).

<u>05.042 Flight tests</u>. The glider shall be subjected to the flight tests necessary to prove compliance with the flight and operation requirements specified in § 05.7.

<u>05.05 Procedure for type certificate</u>.

<u>05.050 Examination of data</u>. The Authority will examine partial units of the required technical data, provided that each unit is complete in itself with respect to both analysis and drawings. When the required technical data contain serious errors or omissions, the examination may be discontinued until the data have been properly corrected.

05.051 Structural inspection. An official representative of the Authority will conduct all necessary inspections of the structure and methods of fabrication prior to completion of the glider and will witness structural tests in compliance with these regulations. (See § 05.35).

05.052 Type inspection authorization. A type inspection will be authorized after the required technical data have been found to be satisfactory.

05.053 Type inspection procedure. The type inspection shall consist of a ground inspection and a flight test of a glider built to conform with the technical data previously submitted and on which the type inspection authorization was based. Prior to, or at the time of, the type inspection, the applicant shall submit the following to the representative of the Authority:

(a) Affidavit of conformity.
(b) Weight and balance (actual) report.
(c) Applicant's flight test report.

If during any part of the ground inspection or flight test there is noted any unfavorable characteristic or serious defect, the type inspection may be discontinued until proper corrective measures have been taken by the applicant.

05.054 Certification of gliders. The procedure specified in the following subparagraphs will be followed in certifying as to the airworthiness of a glider.

05.0540 Issuance of aircraft specification. Upon completion of all reports, tests and inspections required to prove compliance with the airworthiness requirements to the satisfaction of the Authority and upon receipt of the certification of the inspector (or inspectors) who conducted the type inspection to the effect that the glider inspected was found to be airworthy, together with properly executed inspection forms specified in the preceding paragraphs, an Aircraft Specification will be issued for the type and model of the glider in question. The Aircraft Specification will certify as to the airworthiness of gliders of the type in question when manufactured and inspected in accordance with the provisions noted thereon.

05.0541 Issuance of type certificates. A type certificate such as is described in Part 01 will be issued to the applicant upon compliance with the requirements therein.

05.0542 Authenticated data. As a part of the type certificate, the Authority will furnish the applicant, upon issuance of such certificate, one set of drawing lists on which the seal of the Authority is impressed. These lists shall show acceptance of the drawings as partial proof of the airworthiness of the type of glider to which they apply.

05.055 Confidential data. All technical data submitted by the applicant for the Authority's file will be held confidential and will be used only in connection with the airworthiness rating of the glider or gliders to which such data apply; provided, however, that the Authority may at its discretion make such use of the confidential data as is required in the interests of public safety. Access to confidential data will be provided to accredited representatives of the holder of, or applicant for, a pertinent certificate. Confidential data will not be used for reference purposes in connection with the repair, alteration or remodeling of certificated gliders by persons other than the holder of the pertinent certificate without the written consent of such holder unless he is out of business or has given the Authority blanket permission for such use.

05.06 Changes, repair and alteration.

05.060 Change, repair or alteration of certificated gliders. Change, repair or alteration of a certificated glider renders such glider subject to re-rating as to airworthiness in accordance with the maintenance, repair and alteration requirements of the Authority, but does not affect the type certificate on which the airworthiness certification may have been based.

05.061 Changes affecting type certificate. The holder of a type certificate shall apply for approval of any specific change or revision of the approved drawings or specifications which, in the opinion of the Authority, affect the airworthiness of the glider and shall 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. If, in the opinion of the Authority, 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 glider type unairworthy the Aircraft Specification may be modified to include gliders embodying the approved changes, and sealed copies of the revised drawing list pages will be returned to the applicant.

05.1 Definitions.

05.100 Gross weight, W. The maximum weight of the glider and its contents used for purposes of showing compliance with the requirements hereinafter specified.

05.101 Design wing area, S. The area enclosed by the projection of the wing outline, including ailerons and flaps, but ignoring fairings and fillets, on a surface containing the wing chords. The outline is assumed to extend through the fuselage to the plane of symmetry.

05.102 Design wing loading, $s = W/S$. The gross weight (& 05.100) divided by the design wing area (& 05.101).

05.103 Air density, ρ. The mass density of the air through which the glider is moving, in terms of the weight of a unit volume of air divided by the acceleration due to gravity. The symbol ρ_o denotes the mass density of air at sea level under standard atmospheric conditions and has a value of 0.002378 slugs per cubic foot. (See & 05.124 for definition of standard atmosphere.)

05.104 True airspeed, V_t. The velocity of the glider, along its flight path, with respect to the body of air through which the glider is moving.

05.105 Indicated airspeed, V. The true airspeed multiplied by the term $\sqrt{\rho/\rho_o}$ (see & 05.103.)

05.106 Design auto-winch tow speed, V_{taw}. The maximum indicated airspeed at which the glider is assumed to be towed by automobile or winch.

05.107 Design aircraft tow speed, V_{ta}. The maximum indicated airspeed at which the glider is assumed to be towed by aircraft. ($V_{ta} = V_g$, usually.)

05.108 Design gliding speed, V_g. The maximum indicated airspeed to be used in determining the pertinent structural loading conditions. (See && 05.211 and 05.732).

05.109 Design stalling speed, V_s. The computed indicated airspeed in unaccelerated flight based on the maximum lift coefficient of the wing and the gross weight. When high-lift devices are in operation the corresponding stalling speed will be denoted by V_{sf}.

05.110 Design flap speed, V_f. The indicated airspeed at which maximum operation of high-lift devices is assumed. (See && 05.211 and 05.732).

05.111 Design gust velocity, U. A specific gust velocity assumed to act normal to the flight path. (See & 05.2121).

05.112 Dynamic pressure, q. The kinetic energy of a unit volume of air.

$$q = 1/2\rho V_t^2 \text{ (in terms of true airspeed in feet per second).}$$
$$= 1/2\rho_o V^2 \text{ (in terms of indicated airspeed in feet per second).}$$
$$= V^2/391 \text{ pounds per square foot, when V is miles per hour indicated airspeed.}$$

(See § 05.103 for definition of ρ)

05.113 Load factor or acceleration factor, n. The ratio of a load to the gross weight. When the load in question represents the net external load acting on the glider in a given direction, n represents the load factor or acceleration factor in that direction.

05.114 Limit load. A load (or load factor, or pressure) which it is assumed or known may be safely experienced but will not be exceeded in operation.

05.115 Factor of safety, j. A factor by which the limit loads are multiplied for various design purposes.

05.116 Ultimate factor of safety, j_u. A specified factor of safety used in determining the maximum load which the glider structure is required to support.

05.117 Yield factor of safety, j_y. A specified factor of safety used in connection with the prevention of permanent deformations.

05.118 Ultimate load. A limit load multiplied by the specified ultimate factor (or factors) of safety. (See above definitions and § 05.200).

05.119 Yield load. A limit load multiplied by the specified yield factor (or factors) of safety. (See above definitions and § 05.201).

05.120 Strength test. A static load test in which the ultimate loads are properly applied. (See §§ 05.200 and 05.331).

05.121 Proof test. A static load test in which the yield loads are properly applied for a period of at least one minute. (See § 05.201).

05.122 Balancing loads. Loads by which the glider is placed in a state of equilibrium under the action of external forces resulting from specified loading conditions. The state of equilibrium thus obtained may be either real or fictitious. Balancing loads may represent air loads, inertia loads, or both. (See §§ 05.218 and 05.2210).

05.123 Aerodynamic coefficients, C_L, C_M, CP, etc. The coefficients hereinafter specified are those of the "absolute" (nondimensional) system adopted as standard in the United States. The subscripts N and C used hereinafter refer respectively to directions normal to and parallel with the basic chord of the airfoil section. Other subscripts have the usual significance. When applied to an entire wing or surface, the coefficients represent average values and shall be properly correlated with local conditions (load distribution) as required in §05.217.

05.124 Standard atmosphere (standard air). Standard atmosphere refers to that variation of air conditions with altitude which has been adopted as standard in the United States. (See any aeronautics text book or handbook, or NACA Technical Report No. 218).

05.125 Primary structure. Those portions of the glider, the failure of which would seriously endanger the safety of the glider.

05.2 Structural loading conditions.

05.20 General structural requirements.

05.200 Strength. The primary structure (see § 05.125) s be capable of supporting the ultimate loads (see § 05.118) dete mined by the loading conditions and ultimate factors of safety h inafter specified, the loads being properly distributed and appli

05.201 Deformations. The primary structure shall be capable of supporting without detrimental permanent deformations, for a period of at least one minute, the yield loads (see § 05.119) determined by the loading conditions and yield factors of safety hereinafter specified, the loads being properly distributed and applied. Where no yield factor of safety is specified a factor of 1.0 shall be assumed. In addition, temporary deformations which occur before the yield load is reached shall be of such a nature that their repeated occurrence will not weaken or damage the primary structure.

05.202 Stiffness. The primary structure shall be capable of supporting the limit loads (see § 05.114) determined by the loading conditions hereinafter specified without deflecting beyond whatever limits may be hereinafter prescribed or which may be deemed necessary by the Authority for the case in question.

05.21 Flight loads.

05.210 General. The airworthiness rating of a glider with respect to its strength under flight loads will be based on the airspeeds and accelerations (from maneuvering or gusts) which can safely be developed in combination. For certain classes of gliders, the acceleration factors (specified in terms of load factors) and gust velocities are arbitrarily specified hereinafter and shall be used for those classes. The airspeeds which can safely be developed in combination with the specified load factors and gusts shall be determined in accordance with the procedure hereinafter specified and shall serve as a basis for restricting the operation of the glider in flight. (See §§ 05.731, 05.732, and 05.733).

05.211 Design airspeeds. The design airspeeds shall be so selected by the desi-ner that the resulting operation limitations will be consistent with the type and intended use of the glider. Minimum values for V_g, V_{taw}, and V_f are specified in Table 05-2.

TABLE 05-2 — Minimum Design Airspeeds and Minimum Limit Manoeuvering Load Factors for the Symmetrical Flight Conditions

1	Class of Glider (See Table 05-1)	I	II	III
2	Design Gliding Speed V_g, mph (See Note 1)	88	72	56
3	Design Auto-Winch Tow Speed, V_{taw}, mph.	55	55	50
4	Design Flap Speed V_f, mph.	$1.67V_{sf}$	$1.67V_{sf}$	$1.67V_{sf}$
5	Positive Load Factor, n (See note 2)	5.33	4.67	4.0
6	Negative Load Factor, n	-2.67	-2.33	-2.0

(1) The design gliding speed, V_g, shall not be less than the design aircraft tow speed, V_{ta}. (See §05.107)

(2) The minimum positive limit load factor, n, shall not be less than that obtained from the following formula:

$$n = \frac{1}{(s-e)} \left[\frac{V^2_{taw}}{256} - e \right], \text{ where}$$

s = design wing loading, psf
e = unit wing weight, psf
V_{taw} - see item 3 of table

05.212 Load factors. The flight load factors specified hereinafter shall represent wing load factors. The net load factor or acceleration factor shall be obtained by proper consideration of balancing loads acting on the glider in the flight conditions specified in § 05.213 (See § 05.218).

05.2120 Manoeuvering load factors. The limit manoeuvering load factors specified in Table 05-2 shall be considered as minimum values unless it can be proved, to the satisfaction of the Authority, that the glider embodies features of design which make it impossible to develop such values in flight, in which case lower values may be used subject to the approval of the Authority.

05.2121 Gust load factors. The gust load factors shall be computed on the basis of a gust of the magnitude specified, acting normal to the flight path. Proper allowance shall be made for the effects of aspect ratio on the slope of the lift curve.

05.2122 Factors of safety. A minimum limit factor of safety of 1.0 and a minimum ultimate factor of safety of 1.5 shall be used unless otherwise specified. See also § 05.27 for multiplying factors of safety required in certain cases.

05.213 Symmetrical flight conditions (flaps retracted).

05.2130 Basic flight envelope. The basic flight envelope, representing limit loads, shall be determined from the following conditions.

05.2131 For the positive portion of the basic flight envelope, the flight speed shall vary between the speed corresponding to the minimum positive load factor at the positive dynamic C_L maximum and the design gliding speed, V_g. For the negative portion of the basic flight envelope, the flight speed shall vary between the speed corresponding to the minimum negative load factor at the negative dynamic C_L maximum and the design gliding speed, V_g.

05.2132 The minimum positive limit load factor throughout the speed range specified in § 05.2131 shall not be less than that specified in Table 05-2, or that corresponding to a 30 fps. "up" gust.

05.2133 The minimum negative limit load factor throughout the speed range specified in § 05.2131 shall not be less than that specified in Table 05-2, or that corresponding to a 30 fps. "down" gust.

05.2134 A number of loading conditions sufficient to cover all critical conditions shall be determined from the basic flight envelope and investigated in the structural report (§ 05.032).

05.214 Symmetrical flight conditions (flaps or auxiliary devices in operation).

05.2140 General. When flaps or other auxiliary high-lift devices are installed on the wings, suitable provisions shall be made to account for their use in flight at the design flap speed V_f (§ 05.110). Minimum values of the design flap speed are specified in Table 05-2. These provisions shall be based on the intended use of such devices.

05.215 Unsymmetrical flight conditions.

05.2150 General. When deemed necessary by the Authority, suitable investigations shall be made for unsymmetrical flight conditions.

05.216 Special flight conditions.

05.2160 Gust at reduced weight. The requirements for gust conditions (excepting tail surface gust conditions) under any loading between minimum and maximum weights shall be met by primary structure critically loaded thereby.

05.217 Wing load distribution. The limit air loads and inertia loads acting on the wing structure shall be distributed and applied in a manner closely approximating the actual distribution in flight.

05.218 Balancing loads. The balancing loads referred to in § 05.212 shall be computed by a rational method or by a suitable arbitrary method which is considered satisfactory by the Authority.

05.22 Control surface loads.

05.220 General. In addition to the flight loads specified in § 05.21, the primary structure shall meet the requirements hereinafter specified to account for the loads acting on the control surfaces. The following loading conditions include the application of balancing loads (§§ 05.218 and 05.122) derived from the symmetrical flight conditions and also cover the possibility of loading the control surfaces in operating the glider and by encountering gusts. A minimum limit factor of safety of 1.0 and a minimum ultimate factor of safety of 1.5 shall be used unless otherwise specified. See also § 05.27 for multiplying factors of safety required in certain cases.

05.221 Horizontal tail surfaces.

05.2210 Balancing. The limit load acting on the horizontal tail surfaces shall not be less than the maximum balancing load obtained from the loading conditions specified in § 05.213. (See also § 05.218). The load shall be distributed in accordance with Figure 05-1.

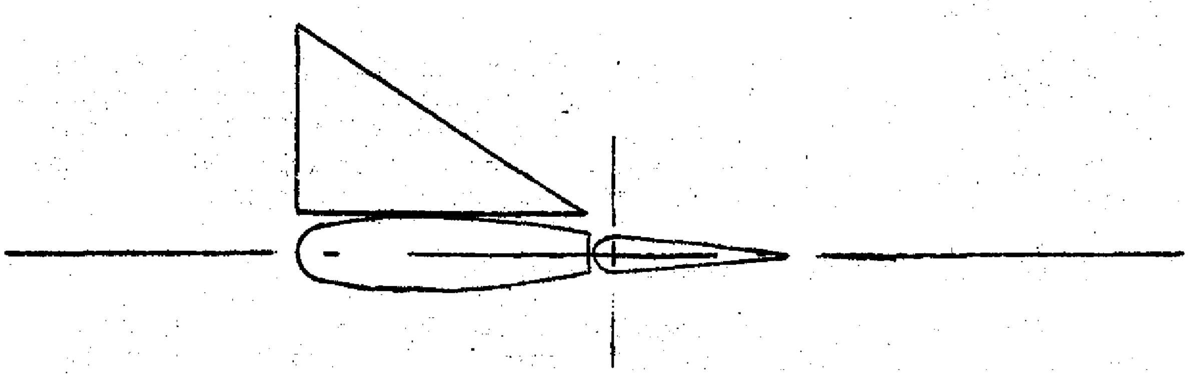


FIG. 05-1 "BALANCING" AND "DAMPING" TAIL LOAD DISTRIBUTION

05.2211 Maneuvering (horizontal surfaces). The minimum average limit pressure specified in Figure 05-2 shall be applied in either direction and distributed in accordance with Figure 05-3.

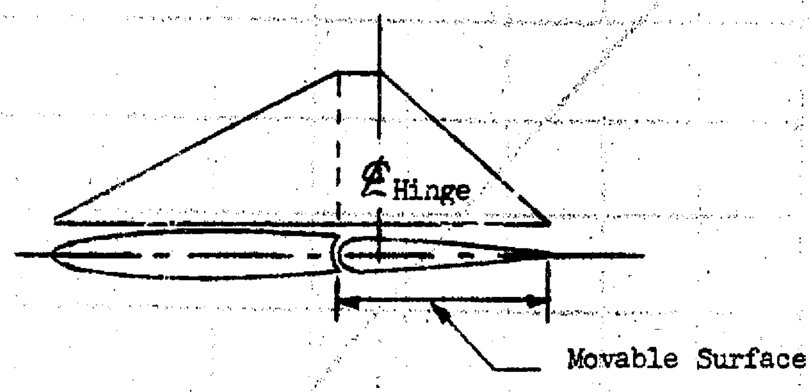


FIG. 05-3 "MANEUVERING" TAIL LOAD DISTRIBUTION

05.2212 Damping (horizontal surfaces). The total limit load acting on the fixed surface (stabilizer) in the maneuvering condition (§ 05.2211) shall be applied in accordance with load distribution of Figure 05-1, acting in either direction. The load acting on the movable surface in the maneuvering condition may be neglected in determining the damping loads.

05.222 Vertical tail surfaces.

05.2220 Maneuvering. The minimum average limit pressure specified in Figure 05-4 shall be applied in either direction and distributed in accordance with Figure 05-3.

05.2221 Damping (vertical surfaces). The total limit load acting on the fixed surface (fin) in the maneuvering condition shall be applied in accordance with the load distribution of Figure 05-1, acting in either direction. The load acting on the movable surface in the maneuvering condition may be neglected in determining the damping loads.

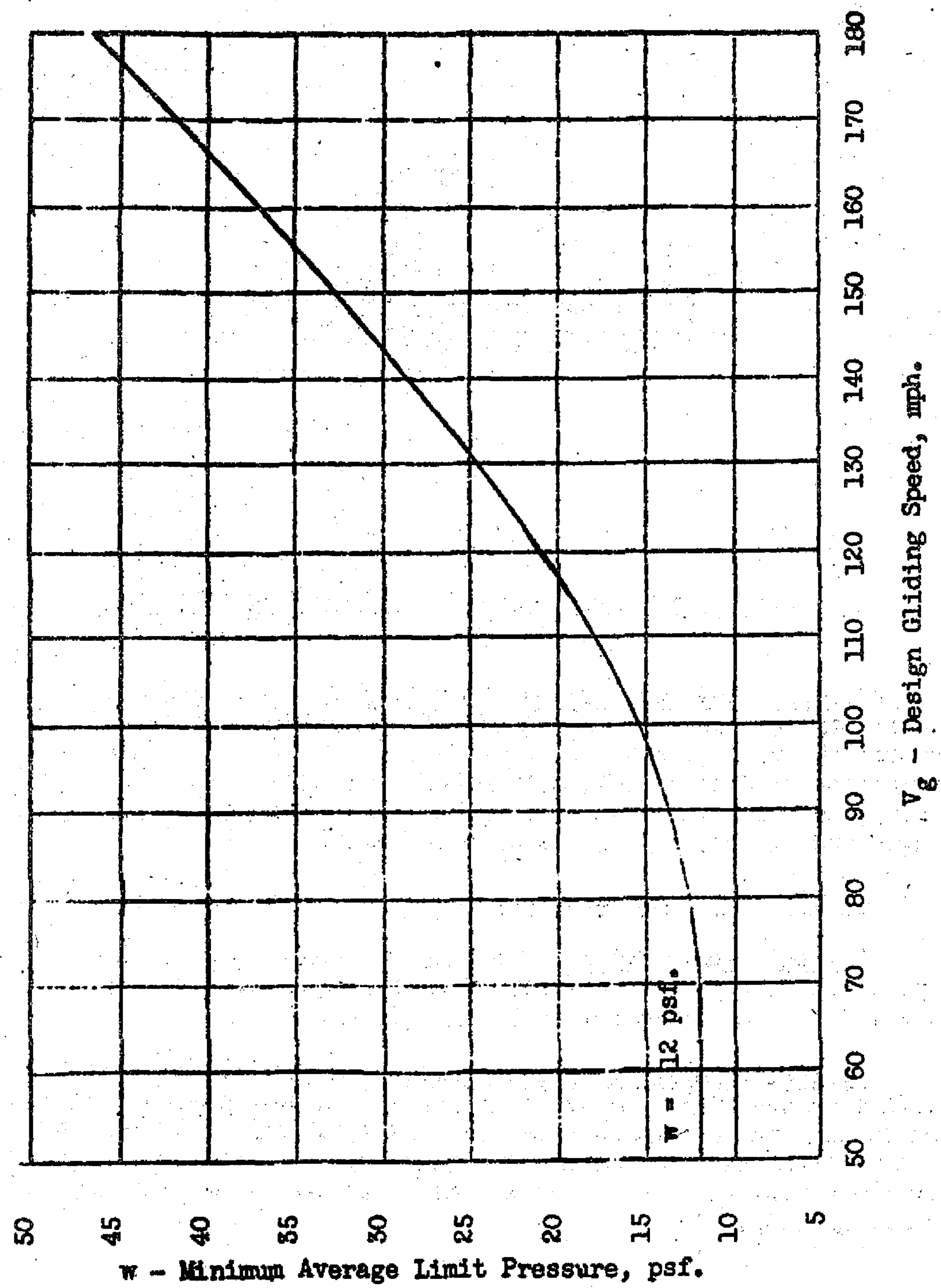


FIG. 05-2 "MANEUVERING" HORIZONTAL TAIL LOADING

05.2222 Gusts (vertical surfaces). The minimum average limit pressure shall not be less than that corresponding to a 15 fps. sharp-edged gust at the design gliding speed, V_g. The gust shall be assumed to act normal to the plane of symmetry in either direction. For the purpose of determining the slope of the tail lift curve, the aspect ratio shall not be taken as less than 2.0. The chord distribution shall simulate that for a symmetrical airfoil, except that the distribution of Figure 05-1 may be used where applicable.

05.223 Ailerons.

05.2230 Maneuvering. The minimum average limit pressure specified in Figure 05-5 shall be applied in either direction and distributed in accordance with Figure 05-6.

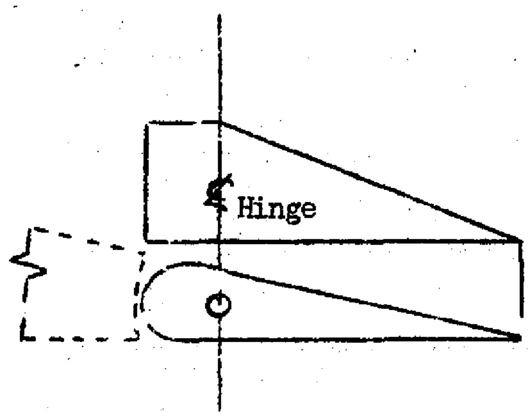


FIG. 05-6 AILERON LOAD DISTRIBUTION

05.224 Wing flaps. Wing flaps shall be loaded in accordance with the provisions of § 05.2140. In any case the average limit pressure shall not be less than 9 psf. and shall be considered uniformly distributed unless a more rational distribution is used.

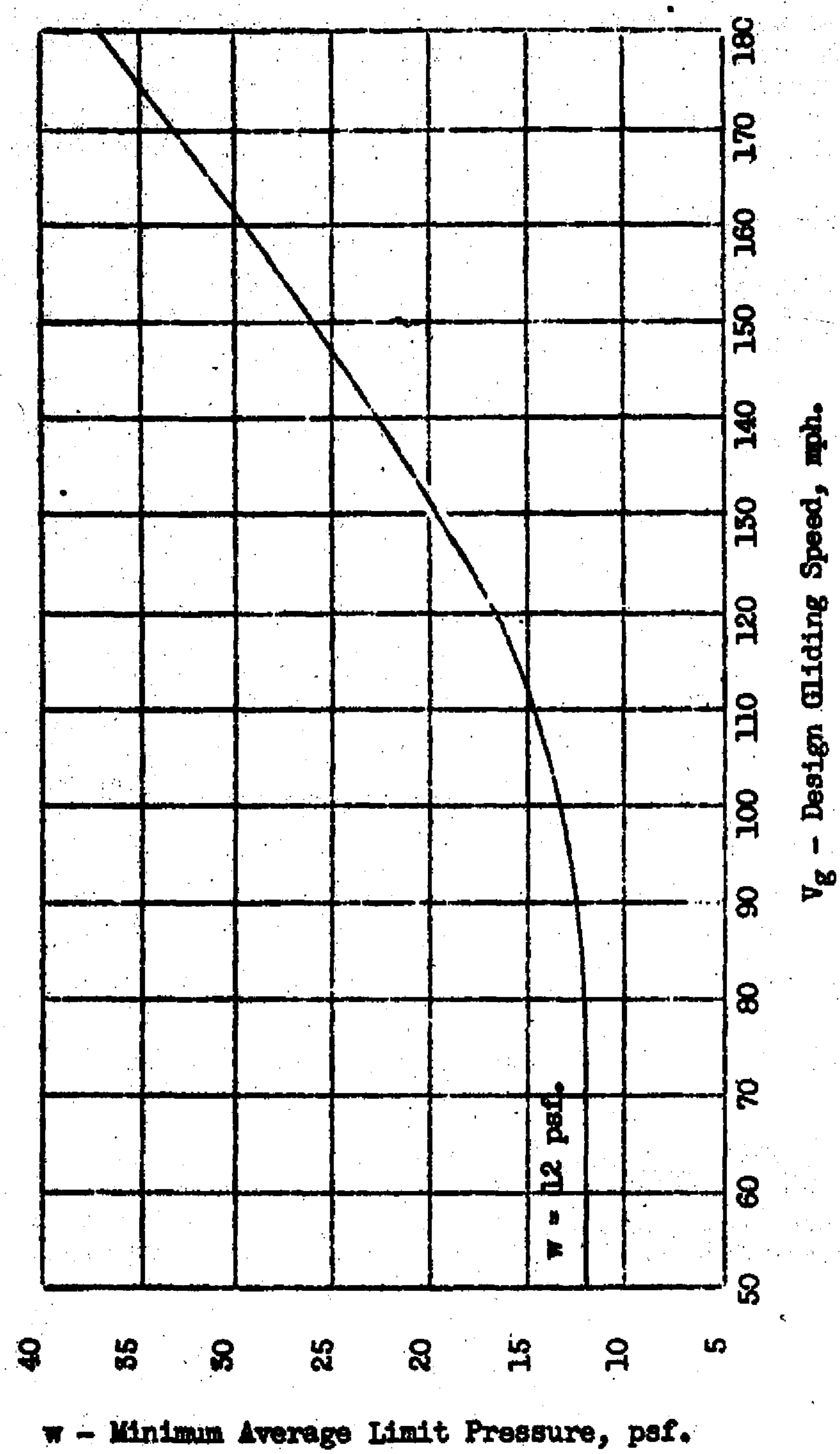


FIG. 05-4 "MANEUVERING" VERTICAL TAIL LOADING

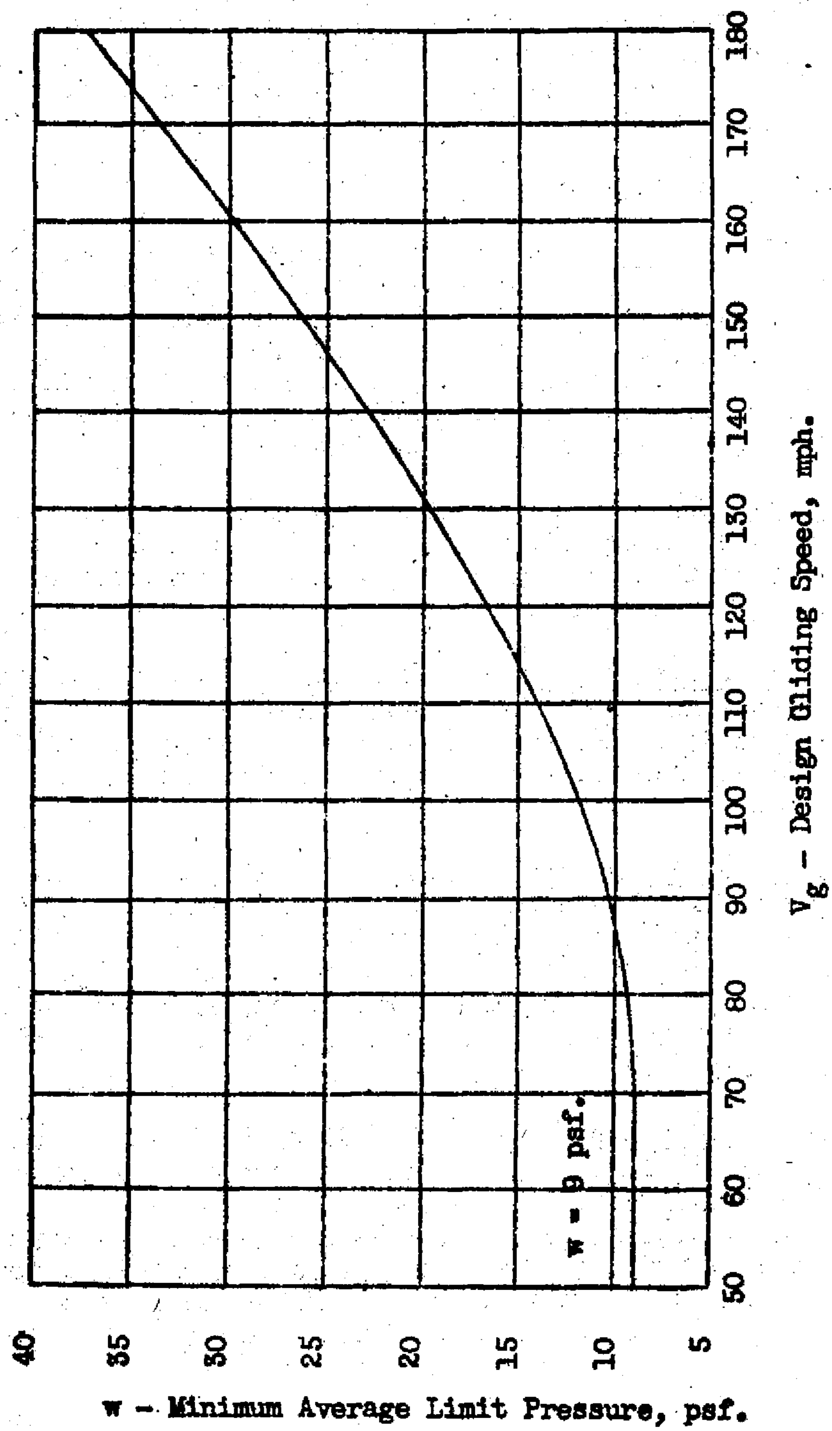


FIG. 05-5 "MANEUVERING" AILERON LOADING

05.225 Special devices. Special rulings shall be obtained from the Authority in connection with the design and analysis of surfaces equipped with tabs, wing-slot structures, spoilers, unconventional ailerons, auxiliary airfoils and similar devices. Requests for special rulings shall be accompanied by suitable drawings or sketches of the structure in question, together with general information and an outline of the method by which it is proposed to determine the structural loading conditions.

05.23 Control system loads.

05.230 General. All control systems shall be designed for at least the limit forces hereinafter specified, unless a more rational method acceptable to the Authority is used. Unless otherwise specified, a minimum limit factor of safety of 1.0 and a minimum ultimate factor of safety of 1.5 shall be used. See also § 05.27 for multiplying factors of safety required in certain cases. See also § 05.342 for operation requirements for control systems.

05.2300 The forces in the control system members, cables or push rods operating the movable surfaces shall be computed and their effect on the rest of the structure shall be investigated and allowed for in the design of such structure.

05.231 Elevator systems. In applying § 05.230, a control force of 150 pounds shall be assumed to act in a fore-and-aft direction and shall be applied at the grip of the control stick, or shall be equally divided between two diametrically opposite points on the rim of the control wheel.

05.232 Rudder systems. In applying § 05.230 a control force of 150 pounds shall be assumed to act in a direction which will produce the greatest load in the control system and shall be applied at the point of contact of the pilot's foot with the control pedal. As a separate condition, two forces of 200 pounds magnitude shall be assumed to act simultaneously at both points of contact of the pilot's feet with control pedal.

05.233 Aileron systems. In applying § 05.230, it shall be assumed that the ailerons are loaded in opposite directions. A control force of 60 pounds shall be assumed to act in a lateral direction at the grip of the control stick, or shall be assumed to act as part of a couple equal to the specified force multiplied by the diameter of the control wheel. Suitable assumptions shall be made as to the distribution of the control force between the ailerons.

05.234 Flap and auxiliary control systems. In applying § 05.230, suitable minimum manual forces shall be assumed to act on flap control systems and other similar controls.

05.235 Towing and lauching release mechanism control systems. In applying § 05.230, a control force of 75 pounds shall be assumed to act in a suitable direction and shall be applied at the grip of the control handle or lever.

05.24 Ground loads.

05.240 General. The following conditions represent the minimum amount of investigation required for conventional landing gear. For unconventional types, it may be necessary to investigate other landing attitudes, depending on the arrangement and design of the landing gear members. Consideration will be given to a reduction of the specified limit load factors when it can be proved that the shock absorbing system will positively limit the acceleration factor to a definite lower value. A minimum limit factor of safety of 1.0 and a minimum ultimate factor of safety of 1.5 shall be used, unless otherwise specified. See also § 05.27 (especially § 05.279) for multiplying factors of safety required in certain cases.

05.241 Level landing. The glider shall be assumed to make contact with the ground while in a level attitude. The basic vertical component shall be equal to the gross weight of the glider. The following additional assumptions shall be made:

(a) For wheel type landing gears, the minimum vertical limit load factor shall be 4.0. The resultant ground reaction shall pass through the center of the axle, and shall be obtained by combining the vertical component with a rearward acting horizontal component equal to one-quarter of the vertical component.

(b) For skid type landing gears, the minimum vertical limit load factor shall be 5.0. The resultant ground reaction shall pass through the center of the skid's contact area, and shall be obtained by combining the vertical component with a rearward acting horizontal component equal to one-half of the vertical component.

05.242 Level landing with side load. As a separate condition, a limit side load equal to 0.167 times the limit vertical component (see § 05.241) shall be assumed to act at the center of the contact area of the tire or skid, together with the loads specified in § 05.241. The side load shall be assumed to act in either direction normal to the plane of symmetry.

05.243 Nose-down landing. For this landing condition, the following assumptions shall be made:

(a) For wheel type landing gears, the glider shall be assumed to make contact on the nose skid and wheel. The minimum limit re-

resultant inertia force shall act at the center of gravity of the glider, shall be equal to 4.0 times the weight, and shall act forward and downward at an angle 14° with vertical. The direction of the ground reactions at the contact points shall be opposite to the resultant inertia force.

(b) For skid type landing gears, the glider shall be assumed to be nosed down 15°. The minimum limit, vertical component of the ground reaction shall be equal to 5.0 times the weight. The resultant ground reaction shall pass through the most forward point suitable for the application of oblique loads, and shall be obtained by combining the vertical component with a horizontal component equal to one-half of the vertical component.

05.244 Head-on landing. The forward portion of the fuselage shall be capable of resisting an ultimate load of 4.0 times the gross weight of the glider acting aft through the foremost point(s) suitable for the application of such a load. Partial failure of the structure under these conditions is permissible, provided that the specified ultimate load can be resisted without endangering the occupants, assuming safety belts to be fastened.

05.245 Wing tip landing. Suitable provisions shall be made to provide adequate structure to resist loads encountered in wing tip landings (rotating landings).

05.25 Launching and towing loads.

05.250 General. The following requirements do not apply to the entire glider structure, but have particular reference to the towing and launching, and holding fittings (and/or mechanisms) and the structures to which they are attached. These requirements are somewhat arbitrary in nature and will be suitably revised when satisfactory test results are available. A minimum limit factor of safety of 1.0 and a minimum ultimate factor of safety of 1.5 shall be used, unless otherwise specified. See also § 05.27 for multiplying factors of safety required in certain cases.

05.251 A limit load of 1200 pounds or 3.0 times the gross weight, whichever is greater, shall be assumed to act in the following separate cases:

(a) Forward at the towing and launching fitting (or mechanism), and aft at the rear holding fitting.

(b) At the towing and launching fitting, and directed forward and upward at an angle of 30° with the longitudinal axis.

(c) At the towing and launching fitting, and directed forward and downward at an angle of 75° with the longitudinal axis.

(d) At the towing and launching fitting, and directed forward and sideward at an angle of 30° with the longitudinal axis.

05.252 Unless the strength of the wing in resisting rearward acting chord loads is equal to or greater than the strength against forward acting chord loads, suitable provisions shall be made to provide adequate strength of wing drag trusses to resist chord inertia loads developed in shock chord and winch launches.

05.26 Special loading conditions.

05.260 Pilot and passenger loads. Pilot and passenger loads in the accelerated flight conditions shall be computed for a standard passenger weight of 170 pounds and a minimum ultimate factor of safety of 1.50 shall be used, except that seats need not be designed for the reduced weight gust conditions specified in § 05.2160. This shall not exempt the primary structure from such gust conditions.

05.261 Structures to which safety belts are attached shall be capable of withstanding an ultimate load of 1,000 pounds per person applied through the safety belt and directed upward and forward at an angle of 45 degrees with the longitudinal axis.

05.262 Local loads. The primary structure shall be designed to withstand local loads caused by dead weights and control loads. Baggage and ballast compartments shall be designed to withstand loads corresponding to the maximum authorized capacity. The investigation of dead weight loads shall include reduced weight gust conditions to insure that the most severe combinations have been investigated. See § 05.90 for standard weights.

05.27 Multiplying factors of safety.

05.270 General. In addition to the minimum factors of safety specified for each loading condition, the multiplying factors specified in Table 05-3 and the following paragraphs shall be incorporated in the structure. The total factor of safety required for any structural component or part equals the minimum factor of safety specified for the loading condition in question multiplied by the factors of safety hereinafter specified, except that certain multiplying factors may be included in others, as indicated in Table 05-3.

05.271 Fittings. All fittings in the primary structure shall incorporate the multiplying factor of safety specified in Table 05-3. For this purpose, fittings are defined as parts used to connect one primary member to another and shall include the bearing of those parts on the members thus connected. Continuous joints in metal plating and welded joints between primary structural members are not classified as fittings. (See also § 05.4).

05.272 Castings. All castings used in the primary structure shall incorporate a multiplying factor of safety not less than specified in Table 05-3.

05.273 Parallel double wires. When parallel double wires are used in wing lift trusses, each wire shall incorporate a multiplying factor of safety not less than that specified in Table 05-3.

05.274 Wires at small angles. Wire or tie-rod members of wing or tail surface external bracing shall incorporate a multiplying factor of safety computed as follows:

$K = L/2R$ (except that K shall not be less than 1.0), where
K = the additional factor,
R = the reaction resisted by the wire in a direction normal to the wing or tail surface plane, and
L = the load required in the wire to balance the reaction R.

TABLE 05-3. - Additional (Multiplying) Factors of Safety (See § 05.27)

Item	Component	Reference Part .05	Additional Yield Factor of Safety, j_y	Additional Ultimate Factor of Safety, j_u	May be Covered by Item No.
1	Fittings (except control system fittings)	.271	None	1.20	2,4,5,6,7, 8,9.
2	Castings	.272	None	2.00	7,8
3	Parallel double wires in wing lift truss	.273	None	1.05	4
4	Wires at small angles	.274	None	See Ref.	—
5	Double drag truss wires	.275	None	See Ref.	—
6	Torque tubes used as hinges	.276	None	1.5	—
7	Control surface hinges(1)	.277	None	6.67	—
8	Control system joints (1)	.277	None	3.33	—
9	Wire sizes	.278	None	See Ref.	—
10	Wing lift truss (when affected by landing loads)	.279	None	1.20	

(1) For bearing stresses only.

05.275 Double drag trusses. Whenever double drag trussing is employed, all drag wires shall incorporate a multiplying factor of safety varying linearly from 3.0, when the ratio of overhang to root chord of overhang is 2.0 or greater, to 1.20 when such ratio is 1.0 or less, assuming an equal division of drag load between the two systems.

<u>05.276</u> <u>Torque tubes used as hinges</u>. When steel torque tubes are employed in direct bearing against strap-type hinges they shall incorporate a multiplying factor of safety at the hinge point not less than that specified in Table 05-3. (See also § 05.4)

<u>05.277</u> <u>Control surface hinges and control system joints</u>. Control surface hinges and control system joints subjected to angular motion, excepting ball and roller bearings and AN standard parts used in cable control systems, shall incorporate multiplying factors of safety not less than those specified in Table 05-3, with respect to the <u>ultimate</u> bearing strength of the softest material used as a bearing. For ball or roller bearings a <u>yield</u> factor of safety of 1.0 with respect to the manufacturer's non-Brinell rating is considered sufficient to provide an adequate <u>ultimate</u> factor of safety.

<u>05.278</u> <u>Wire sizes</u>. (See § 05.4).

<u>05.279</u> <u>Wing lift trusses</u>. When landing loads are carried through a portion of the lift truss, the lift truss members so affected shall incorporate a multiplying factor of safety not less than that specified in Table 05-3.

05.3 Proof of structure.

05.30 General. Proof of compliance with the loading requirements outlined in § 05.2 shall be made in a manner satisfactory to the Authority and may consist of structural analyses, load tests, flight tests, references to previously approved structures, or combinations of the above. Any condition which can be shown to be non-critical need not be further investigated.

05.31 Determination of loadings. The determination of spar running loads, weight distribution, etc., shall be made by methods contained in the Manual 05 or their equivalent.

05.32 Structural analyses. Structural analyses will be accepted as complete proof of strength only in the case of structural arrangements for which experience has shown such analyses to be reliable. References shall be given for all methods of analysis, formulas, theories, and material properties which are not generally accepted as standard. The acceptability of a structural analysis will depend to some extent on the indicated excess strength incorporated in the structure.

05.320 General. The structural analysis shall be based on the guaranteed minimum mechanical properties of the materials specified on the drawings, except in cases where exact mechanical properties of the materials used are determined. The effects of welding, brazing, form factors, stress concentrations, discontinuities, cutouts, bolt holes, instability, redundancies, secondary bending, joint slippage of wood beams, rigging, end fixity of columns, eccentricities, air loads on struts, and vibration shall be properly accounted for when such factors are present to such an extent as to influence the strength of the structure.

05.33 Combined structural analysis and tests. In certain cases it will be satisfactory to combine structural analysis procedure with the results of load tests of portions of the structure not subjected to accurate analysis. (See § 05.341)

05.34 Load tests. Proof of compliance with structural loading requirements by means of load tests only is acceptable provided that strength and proof tests (see §§ 05.120 and 05.121) are conducted to demonstrate compliance with §§ 05.200 and 05.201, respectively, and further provided that the following sub-paragraphs are complied with.

05.340 If load tests do not prove compliance with multiplying factor of safety requirements, the tests shall be supplemented by special tests or analyses to prove compliance with such requirements. (See § 05.27).

<u>05.341</u> When a unit other than the specific one tested is incorporated in the glider presented for certification, the results of <u>strength</u> tests shall be reduced to correspond to the minimum guaranteed mechanical properties of the materials specified on the drawings, unless test loads are carried at least 15 per cent beyond the required values.

<u>05.342</u> <u>Load tests required</u>. The following load tests are required in all cases:

(a) Strength tests of wing ribs. The strength of ribs shall be proved by tests to at least 125 per cent of the <u>ultimate</u> loads for the most severe loading conditions.

(b) Proof tests (see § 05.121) of tail and control surfaces are required to prove compliance with the <u>yield</u> load requirements.

(c) Proof and operating tests of control systems. Proof tests are required to prove compliance with the <u>yield</u> load requirements. Operation tests are required to prove that control systems will operate properly when so loaded as to correspond to one-half the <u>limit</u> control forces specified in § 05.23 for the design of the systems.

(d) Proof and operating tests of launching and towing release mechanisms. Proof tests are required to prove compliance with the <u>yield</u> load requirements. Operation tests are required to prove that release mechanisms will function properly when so loaded as to correspond to one-half the <u>limit</u> forces specified in § 05.251 for the design of the mechanisms, except that the force specified in § 05.251(b) can be neglected for Class III gliders. This section does not apply to "open-hook type" launching fittings used solely for shock chord launching.

<u>05.343</u> The determination of test loads, the apparatus used, and the methods of conducting the tests shall be satisfactory to the Authority. The tests shall be conducted in the presence of a representative of the Authority, unless otherwise directed by the Authority.

<u>05.344</u> <u>Flight load tests</u>. Proof of strength by means of flight load tests will not be accepted unless the necessity therefor is established and the test methods are proved suitable to the satisfaction of the Authority.

<u>05.35</u> <u>Vibration tests</u>. In cases involving exceptionally high design speeds or design features which may be conducive to flutter, tests may be required, at the discretion of the Authority, to determine the natural frequencies of wings, tail surfaces, and control surfaces.

05.4 Design details, products, materials, construction, and workmanship. The primary structure (§ 05.125) shall not incorporate design details which experience has shown to be unreliable or otherwise unsatisfactory. The suitability of all design details shall be established to the satisfaction of the Authority. Products, such as bolts, pins, screws, tie-rods, wires, wire terminals, etc., used in the primary structure shall be of a type and size considered satisfactory by the Authority. The primary structure shall be made from materials which experience or conclusive tests have proved to be uniform in quality and strength and to be otherwise suitable for glider construction. The methods of fabrication employed in constructing the primary structure shall be such as to produce a uniformly sound structure which shall also be reliable with respect to maintenance of the original strength under reasonable service conditions. The workmanship of the primary structure shall be of sufficiently high grade to insure proper continued functioning of all parts.

05.5 Equipment.

05.50 General. The equipment required will be dependent upon the type of operation for which certification is sought. The requirements specified herein (§ 05.5) shall be basic equipment requirements. Additional equipment may be required for special cases as specified in other sections of the Civil Air Regulations.

05.500 Each item of equipment specified in the Civil Air Regulations shall be of a type and design satisfactory to the Authority, shall be properly installed, and shall function to the satisfaction of the Authority. Items of equipment for which certification is required shall have been certificated in accordance with the provisions of Part 15 or previous regulations.

05.51 All gliders. All gliders shall have at least the following equipment:

(a) One airspeed indicator. (See §05.5200 for installation requirements).

(b) Certified safety belts for all passengers and members of the crew. (See Part 15 for belt requirements and § 05.5210 for installation requirements.)

(c) A log-book.

(d) Rigging instructions, in the case of gliders with wire-braced wings, either in the form of a sketch or listed data, which shall include sufficient information to facilitate proper rigging.

Note: Items (c) and (d) need not be carried in the gliders.

05.510 Instrument day flying. Gliders certified for blind flying during daylight hours shall have the equipment specified in § 05.51 and, in addition, there shall be installed:

(a) One vertical speed indicator.

(b) One altimeter.

(c) One turn and bank indicator.

(d) One magnetic compass. (See § 05.5201 for installation requirements.)

05.511 Visual-contact night flying. Gliders used for visual-contact night flying shall have the equipment specified in § 05.51 and, in addition, there shall be installed:

(a) A set of certificated standard forward position lights in combination with a certificated tail light. (See Part 15 for light requirements and § 05.5224 for installation requirements.)

(b) An instrument light, unless all of the required instruments can be easily read in the dark. (See § 05.5223 for installation requirements.)

(c) A battery suitable as a source of energy supply for such items of equipment as are installed. (See § 05.5221 for installation requirements.)

(d) A set of spare fuses. (See § 05.5222 for installation requirements.)

05.52 Installation requirements. The following regulations apply to the installation of specific items of equipment.

05.520 Instruments. The following regulations shall apply to the installation of instruments when such instruments are required by these regulations.

05.5200 Airspeed indicator. This instrument shall be so installed as to indicate true airspeed at sea level with the maximum practicable accuracy, but in no event shall the instrument error be more than plus or minus 5 miles per hour at speeds between the auto-winch tow placard speed and the maximum certified speed. (See § 05.732).

05.5201 Magnetic compass. This instrument shall be properly damped and compensated, and shall be located where it is not seriously affected by electrical disturbances and magnetic influences.

05.5202 Navigation instruments. Navigation instruments for use by the pilot shall be so installed as to be easily visible to him with the minimum practicable deviation from his normal position and line of vision when he is looking out and forward along the flight path.

05.521 Safety equipment installation.

05.5210 Safety belts. Safety belts shall be so attached that no part of the attachment will fail at a load lower than that specified in § 05.261.

05.522 Electrical equipment installation.

05.5220 General. Electrical equipment shall be installed in accordance with accepted practice and shall be suitably protected from detrimental substances. Adequate clearance shall be provided between moving parts and wiring carrying appreciable current.

05.5221 Battery. Batteries shall be easily accessible. Adjacent parts of the aircraft structure shall be protected with a suitable acid-proof paint if the battery contains acid or other corrosive substance and is not completely enclosed. If the battery is completely enclosed, suitable ventilation shall be provided. All liquid (wet) batteries shall be so installed that spilled liquid will be suitably drained or absorbed without coming in contact with the glider structure.

05.5222 Fuses. Fuses shall be so located that they can readily be replaced in flight. They shall break the current in the electrical system at a sufficiently small current flow to adequately protect all parts of the circuit.

05.5223 Instrument light. The instrument light shall be so installed as to provide sufficient illumination to make all flight instruments easily readable.

05.5224 Position lights. Position lights shall be so installed as to provide the light intensity and ranges of visibility prescribed in Part 15 for tail lights and for standard forward position lights, as the case may be. Forward position lights shall be spaced laterally as far apart as practicable.

05.523 Miscellaneous equipment installation.

05.5230 Seats. Seats or chairs, even though adjustable, shall be securely fastened in place, whether or not the safety belt load is transmitted through the seat. (See § 05.260)

05.6 (Unassigned).

05.7 Flight characteristics.

05.70 Flight requirements. Gliders shall comply with the following flight requirements at all weights up to the maximum for which certification is desired, except that such weight shall not exceed the gross weight (§ 05.100) and under all loading conditions within the center of gravity range to be certified (§ 05.711). There shall be no flight characteristics which, in the opinion of the Authority, render the glider unairworthy.

05.700 Controllability and maneuverability. All gliders shall be controllable and maneuverable at all flying speeds including the minimum flying speed and the maximum certified speed in free flight and for such launching and towing methods for which they are to be certificated.

05.701 Balance. All gliders shall be longitudinally, laterally and directionally balanceable at a normal operating speed.

05.702 Stability. Class I gliders shall be statically and dynamically stable about all three axes under all normal operating conditions. Class II and III gliders shall be at least statically stable about all three axes under all normal operating conditions.

05.703 Spinning. Class I gliders shall be able to recover from a 3-turn "cross control" or "aileron-in" spin in no more than one additional turn. During the spin the control surfaces shall exert no back pressure on the control column. All gliders showing an undue tendency to spin will be considered unairworthy.

05.704 Flutter and vibration. All gliders shall be free from flutter and objectionable vibration in all normal attitudes or conditions of flight between the minimum flying speed and the maximum indicated airspeed attained in official flight tests (see § 05.712).

05.705 Ground characteristics. All gliders shall be free from dangerous ground handling characteristics.

05.71 Flight tests.

05.710 General. Compliance with the foregoing flight requirements shall be demonstrated by means of suitable flight tests of the type glider.

05.7100 The applicant shall provide a person holding an appropriate airman certificate to make the flight tests, but a designated inspector of the Authority may pilot the glider during such parts of the tests as he may deem advisable.

05.7101 Parachutes shall be worn by members of the crew during the flight tests when deemed necessary by the inspector.

05.7102 The applicant shall submit to the inspector of the Authority a report covering all computations and tests required in connection with calibration of flight instruments. The inspector will conduct any tests which appear to him to be necessary in order to check the calibration report or to determine the airworthiness of the glider.

05.711 Loading conditions. The loading conditions used in flight tests shall be such as to cover a normally expected range of loads and center of gravity positions, and will be those for which the glider will be certificated.

05.7110 Ballast. Permanent ballast may be used to enable gliders to comply with performance requirements as to longitudinal stability and balance, provided that the place or places for carrying such ballast is (are) properly designed and installed and plainly marked "Do Not Remove".

05.712 Maximum airspeed. The flight tests shall include steady flight at the design gliding speed (V_g) for which compliance with the structural loading requirements (§ 05.21) has been proved, except that the speed need not exceed the value of V_g specified in Table 05-2. When high-lift devices having non-automatic operation are employed, the tests shall also include steady flight at the design flap speed V_f (§ 05.110), except that they need not involve speeds in excess of $1.67V_{sf}$ (see § 05.109). In cases where the high-lift devices are automatically operated, the tests shall cover the range of speeds within which the devices are operative.

05.713 Airspeed indicator calibration. In accordance with § 05.5200, the airspeed indicator of the type glider shall be calibrated in flight, unless the instrument is satisfactorily calibrated by other means and properly located on the glider. The method of calibration used shall be subject to the approval of the Authority.

05.73. Operation limitations.

05.730 Center of gravity limitations. The maximum variation in the location of the center of gravity for which the glider is certificated to be airworthy shall be established. Means shall be provided, when necessary in the opinion of the Authority, by which the operator is suitably informed of the permissible loading conditions which result in a center of gravity within the certified range.

05.731 Towing limitations. Class III gliders (§ 05.01) and other gliders which are not aircraft towed during official flight tests shall not be aircraft towed. Means shall be provided to so inform the operating personnel.

05.732 Air speed limitations. The maximum certified air speed shall be limited to a value of at least 10 per cent less than either the design gliding speed V_g or the maximum value attained in the official flight tests, whichever is lower (see § 05.712). The maximum certified auto-winch tow speed shall be limited to a value of at least 5 mph less than either the design auto-winch tow speed V_{taw} or the maximum value attained in the official flight tests, whichever is lower. The maximum certified aircraft tow speed shall be limited to a value of at least 10% less than either the design gliding speed, V_g, or the maximum speed attained in the official flight tests, whichever is lower. The maximum certified speed for the operation of high-lift devices shall be limited to 5 mph less than either the design flap speed V_f or the maximum value attained in the official flight tests, whichever is lower. Means shall be provided to effect such limitations or to inform the operating personnel thereof.

05.733 Flight limitations. Class III gliders (§ 05.01) and others which have not complied with § 05.2134 shall not be flown inverted. Class I and II gliders which are not equipped as specified in § 05.510 and Class III gliders shall not be used for any type of operation other than visual-contact day flying. Means shall be provided to inform the operating personnel of these limitations.

05.8 (Unassigned).

05.9 Miscellaneous requirements.

05.90 Standard weights. In computing weights the following standard values shall be used:

Crew and passengers .... 170 lbs. per person, unless otherwise specified by the Authority.
Parachutes .... 20 lbs. each.
Water .... 8.5 lbs. per gallon.

05.91 Leveling means. Adequate means shall be provided for easily determining when the aircraft is in a level position.

CIVIL AERONAUTICS AUTHORITY MANUAL

Part 05.-Glider Airworthiness

05 GLIDER AIRWORTHINESS

05.0 GENERAL

05.005 DEVIATIONS

If the applicant for an airworthiness rating can show that the glider is unconventional with respect to any specific requirement or that the objective of the requirement has been attained, deviations from the letter of the requirement may be permitted.

05.01 CLASSIFICATION OF GLIDERS

It should be noted that the choice of the glider classification rests to a large extent with the designer; e.g., a utility type glider may be certified under any of the three classifications (see CAR Table 05-1). All primary training type gliders must, however, be certified in either Class II or Class III, while high performance type gliders must be certified in either Class I or II. (See CAAM 05.211). In general, the various glider types will be considered to be defined as follows:

a. Primary (Training). A low performance glider designed for use in the primary training of student glider pilots. The fuselage of this type usually consists of a single truss, leaving the pilot entirely exposed to the air stream.
b. Utility (or Secondary). A medium performance glider designed for general purposes. An enclosed primary training glider is not considered as a utility glider.
c. Intermediate. A moderately high performance glider designed for advanced training and general sport flying.
d. High Performance. A high performance or advanced type of glider especially designed for the maximum performance within a selected range of conditions.

When there is any doubt as to the classification of a particular glider, the designer should submit the basic flight envelope (see CAR 05.2130), together with the three-view drawing, to the Authority during the initial stages of the design of the glider in question, so that a definite ruling can be obtained.

NOTE: Since the Authority interprets blind or cloud flying as an acrobatic maneuver (see CAR 60.152), it is required that each person in a glider engaged in such type of flying be properly equipped with a parachute manufactured under a valid type certificate and maintained in accordance with the provisions of the Civil Air Regulations (see CAR 60.72). It is recommended, therefore, that provisions for parachutes be provided in Class I gliders.

05.03 TECHNICAL DATA REQUIRED

A technical data file for each model glider for which an airworthiness rating is desired is necessary. However, reference can be made to previously submitted data for a similar model. Form AC-01-9 (Application for Type Certificate) should refer to one model only. Form AC-01-19 (Application for Production Certificate) should refer to all models involved. When more than one model is covered by the technical data submitted, separate applications for type certificate should be executed and forwarded for each model.

05.030 SUBMISSION TO BRANCH OFFICE

When dealing with Branch Offices of the Authority, particular care should be taken to submit the applications, correspondence and three-view drawings listed in CAR 05 in duplicate.

05.031 DATA REQUIRED FOR AIRWORTHINESS CERTIFICATE ONLY

1. General. When an airworthiness certificate only is desired, the data required are dependent on the particular problems involved in the design concerned. As specified in CAR 01.22, there are two kinds of airworthiness certificates which are classified by the symbols C and R. It will be noted in CAR 02 that the symbol C classifies a glider as complying fully with the airworthiness requirements of CAR 01 and CAR 05, whereas the symbol R classifies the glider as complying in all but some limited respects with these requirements. The data required as a basis for the issuance of a glider airworthiness certificate, specifying either the C or R classification, are substantially the same. In fact, the process of demonstrating that the deficiencies of R classification gliders can be and are compensated for by suitable operation limitations (see 4 below) will usually entail a special study, by the applicant, of design data which have been previously approved and used as a basis for the issuance of an airworthiness certificate specifying C classification.

2. C - Classification. An airworthiness rating, under the terms of this paragraph, may be obtained for a single glider (see 3 below).

3. Single Glider. In addition to the application and three-view drawing, the following data and information are needed:

a. A complete explanation of the current status of the model glider involved.
b. The data specified in CAR 05.031. The applicant for approval is free to develop and present any means he chooses for showing compliance with the specified requirements. Reports on satisfactory "strength" tests may be substituted for structural analyses. (See CAR 05.120 and CAAM 05.34.) Structural components which have undergone "strength" tests can not be incorporated in the glider structure unless satisfactory evidence is submitted to prove that the structure was not damaged in the test.

4. R - Classification. The necessary data and information listed under 3 above should also be submitted by an applicant desiring an approval under R classification. The extent of the additional data required as a basis for issuance of an airworthiness certificate which specifies, as explained in CAR 02.111(b), the use, or uses for which a glider bearing the letter R is deemed airworthy, depends largely upon the nature and extent of the deficiencies which exist. Since it is practically impossible to anticipate, in this publication, just what deficiencies may be discovered by the applicant in each case, no specific references to necessary additional data can be made.

5. It should be noted that there are possibly many deficiencies which cannot be compensated for by operation limitations. In such cases the glider cannot be made eligible for R classification unless revised to eliminate such deficiencies. For example, in the following cases the deficiency cited cannot be compensated for by operation limitations:

a. Deficiency in strength with respect to minimum maneuvering load factors.
b. Unsatisfactory flight characteristics.

6. For a single glider, a complete set of shop drawings, such as is generally supplied by manufacturers seeking type certificates, is unnecessary. Drawings for a single glider are required by the Authority for the following purposes:

a. To explain the details of design and construction, and methods of fabrication with sufficient accuracy to show whether any unairworthy features exist.

b. To give sufficient dimensions and material specifications so that the structural report (see CAAM 05.032 (g)) can be readily checked by the Authority.
c. To enable a representative of the Authority to verify by inspection that the glider has actually been constructed in accordance with the design under consideration.

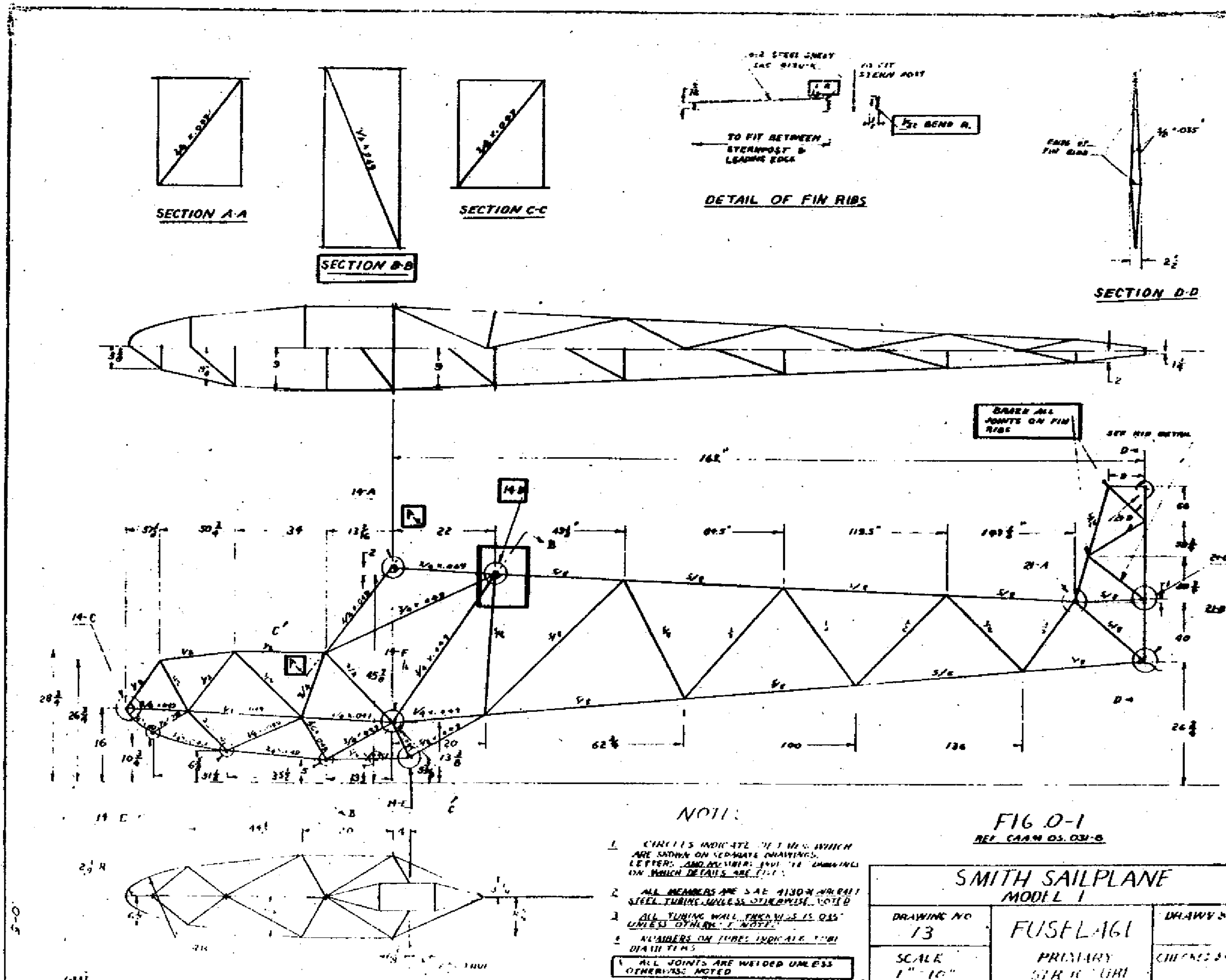
Keeping the above in mind, many satisfactory short cuts and simplifications are possible. In certain cases, line drawings of structural assemblies are satisfactory, provided that they are accompanied by notes and/or detailed drawings (preferably the latter). Semi-free-hand, dimensioned sketches of simple parts which closely approximate the correct scale, and are easily understandable, are acceptable to the Authority. Some of the accepted practices in aircraft drawings such as putting each detail on a separate sheet and making a separate detail of each part, may be eliminated. Such simplifications are limited only by the necessity that all information must be clear and easily accessible. In Fig. .0-1 generally acceptable methods are illustrated by a sample fuselage drawing. The items enclosed in the heavy rectangles are typical omissions. It is recommended that before submitting drawings to the Authority, they be checked by a person who is unfamiliar with the structural details of the particular glider involved so that errors or omissions that might pass unnoticed by an engineer in close contact with the work, will be caught and rectified before the data are submitted to the Authority. Fig. .0-2 illustrates how one drawing can be dispensed with by combining the three-view and final assembly drawings.

05.032 DATA REQUIRED FOR TYPE CERTIFICATE (TC)

When submitting data in support of applications for type certificates covering projects that may require the attention of the Authority over an extended period of time, it is desirable that a schedule of approximate dates of submittal of data be forwarded at an early date.

a. Drawings. A set of drawings should be submitted in blueprint form or equivalent. Drawings should be folded, with the title block out, to a size approximately 9" x 12" and should contain at least the following information.

(1) The manufacturer's designation of the original model to which each drawing applies.
(2) All dimensions essential to the reproduction of



SECTION A-A
SECTION B-B
SECTION C-C
DETAIL OF FIN RIBS
TO FIT BETWEEN STERNPOST & LEADING EDGE
SECTION D-D
BRAZE ALL JOINTS ON FIN RIBS
162"
NOTES
1. CIRCLES INDICATE DETAILS WHICH ARE SHOWN ON SEPARATE DRAWINGS. LETTERS AND NUMBERS INDICATE DRAWINGS ON WHICH DETAILS ARE GIVEN.
2. ALL MEMBERS ARE S.A.E. 4130-X AIRCRAFT STEEL TUBING UNLESS OTHERWISE NOTED
3. ALL TUBING WALL THICKNESS IS .035" UNLESS OTHERWISE NOTED
4. NUMBERS ON TUBES INDICATE TUBE DIAMETERS
5. ALL JOINTS ARE WELDED UNLESS OTHERWISE NOTED
FIG. O-1
REF. CAAM 05.031-6
SMITH SAILPLANE
MODEL 1
DRAWING NO 13
SCALE 1" = 10"
FUSELAGE
PRIMARY STRUCTURE
DRAWN BY
CHECKED BY

an identical glider with respect to structural strength and dimensions.

(3) All dimensions essential for checking the structural report.

(4) Specifications of all materials used in the primary structure (see CAR 05.125), including the guaranteed physical properties in the case of materials the strength properties of which are developed through manufacturing processes, and specifications of all bolts, nuts, rivets, and similar standard parts essential to the strength of the primary structure.

(5) Details of the primary structure, seating arrangement, exits, control systems, equipment installations, and other factors affecting the airworthiness of the glider.

Attention to the following list of frequently omitted items will be of assistance in expediting approval:

(a) Complete dimensions, and references to all standard parts, such as bolts, nuts, and rivets used in assembling each part.

(b) Adequate material specifications and bend radii on all shop drawings.

(c) Location and details of control system pulleys and brackets and of control surface stops.

(d) Suitable assembly drawings showing the method of assembly and calling out the detail parts required for all major installations.

(e) Adequate drawings and descriptions of the operation of spoiler and flap control devices.

(f) Complete structural drawings of all components.

Whenever a drawing previously submitted for one model is also applicable without change to a new model, an additional copy of the drawing is not required. However, as noted below, the drawing list should include a reference to the particular model glider for which the drawing was originally submitted. Whenever the manufacturer's drawing numbering system permits, all drawings received by the Authority are incorporated into a single consecutive file. The drawing lists for each model will in such cases be filed separately according to model. In this manner duplication of files is avoided.

(6) Revision blocks stating the revision letter, the nature of the revision and the date it was made. The manufacturer should hold available a current record of the glider serial numbers to which revisions resulting in a modified finished primary structural part apply.

(7) A three-view drawing of the glider, made to a designated scale, specifying only the external dimensions of the glider (including dimensions and areas of wing and control surfaces). Since equipment is covered by a separate list and eventually appears on the aircraft specification, it is recommended that items of equipment be omitted from the three-view drawing. Inasmuch as the Authority does not certify as to performance, all references to performance should be omitted. A sample three-view drawing is shown in Fig. .0-2 (This drawing includes final assembly information (see CAAM 05.031-6)).

b. <u>Drawing List</u>. A drawing list should be submitted in duplicate, listing in numerical order or by suitable classification the number and title of each drawing submitted under CAR 05.032(a).The drawing list should include references to all drawings originally submitted in connection with applications for airworthiness ratings of other models and which apply to the model in question without change. The drawing list should also indicate, by letter, the latest revision of each drawing.

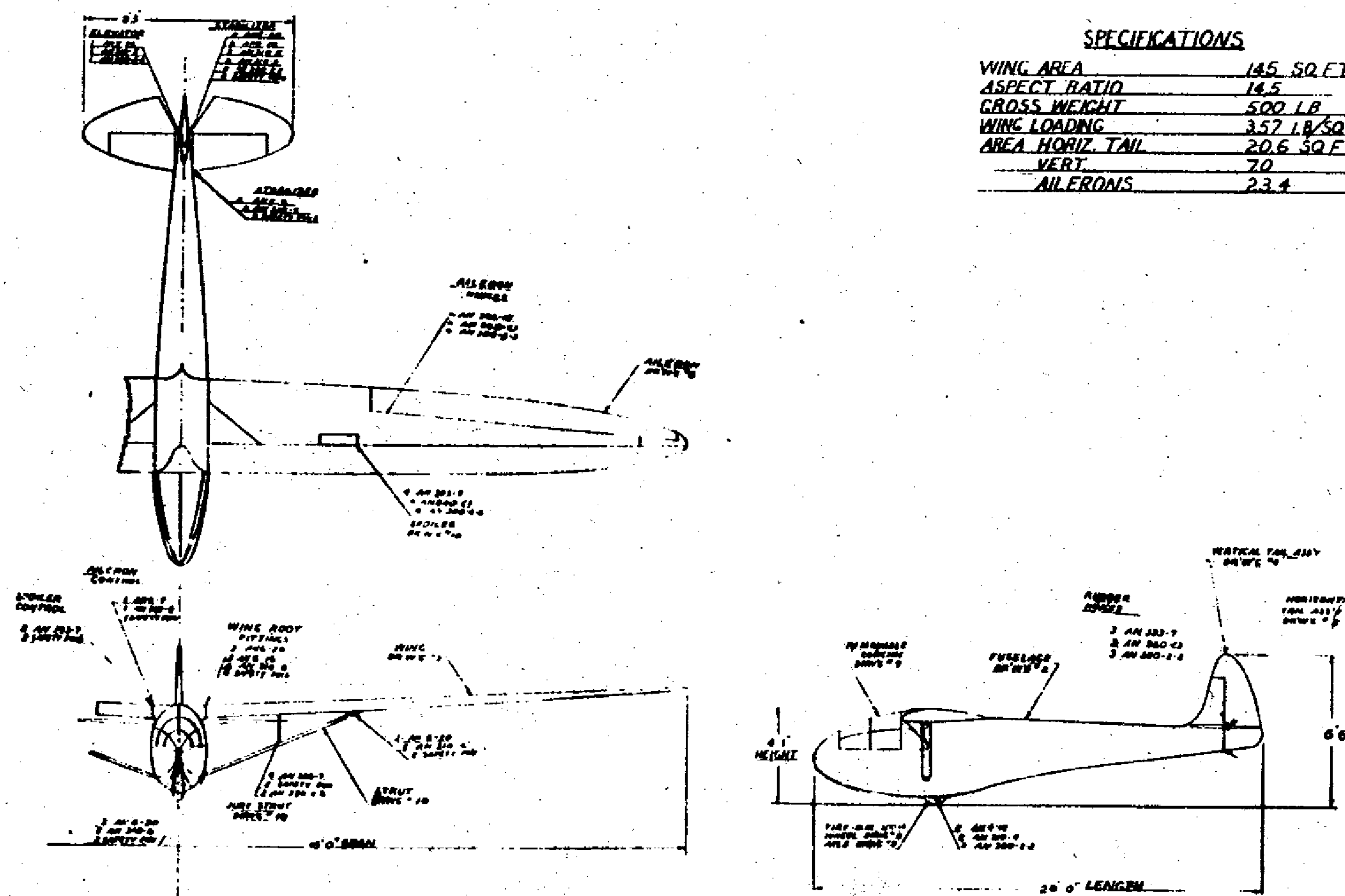
In the preparation of drawing lists it is desirable that the drawings be grouped according to component, such as Wing Group, Fuselage Group, etc. Within each group the drawings should be listed in consecutive order.

When submitting data for approval of revisions to an approved file, the pertinent pages of the drawing list should be attached in duplicate. The date of the latest revision should be noted on such pages.

The duplicate drawing lists required for each approved file may take various forms, dependent upon whether the drawings submitted pertain to one or more models. Sample lists to demonstrate an acceptable form for the usual cases involved are shown in Fig. .0-3.

SPECIFICATIONS

WING AREA	145 SQ FT
ASPECT RATIO	14.5
GROSS WEIGHT	500 LB
WING LOADING	3.57 LB/SQ FT
AREA HORIZ. TAIL	20.6 SQ FT
VERT.	7.0
AILERONS	23.4



NOTES

SMITH SAILPLANE
MODEL 1

THREE VIEW AND FINAL ASSEMBLY

SCALE 1" = 20"

FIG 0-2

I. List when only one new model is involved.

MODEL 1 DRAWING LIST

Drawing No.	Change	Title	Originally Submitted For Model
		WING GROUP	
22001	B	Frame Assembly, Outer Wing	1
22002	K	Spar Assembly, Outer Front	1
		FUSELAGE GROUP	
		EMPENNAGE GROUP, ETC.	

Latest Revision 7/21/38.

II. List when new model has only minor variations from previously approved basic model (1).

MODEL 2 DRAWING LIST

With the exception of the drawings listed under A and B below, the drawing list of Model 1 applies also to Model 2.

A. Model 1 Drawings not pertinent to Model 2. (See arrangement under I above).
B. Drawings pertinent to Model 2 which are in addition to Model 1 list less group A above. (See arrangement under I above).

III. List when new model is a major revision of a previously approved model or models.

MODEL 5 DRAWING LIST

Drawing No.	Change	Title	Originally Submitted For Model
		WING GROUP	
25001	—	Frame Assembly, Outer Wing	5
25002	—	Spar Assembly, Outer Front	5
25003	A	Fitting, Front Spar, Root Attachment	1
25004	E	Fitting, Front Spar, Strut, Etc.	2

Latest Revision 3/27/39.

FIG. .0-3 — SAMPLE DRAWING LISTS
(Ref. CAAM 05.032b)

c. Equipment Lists. Lists specifying the equipment supplied with each glider should be submitted. The location, weight, and model designation of each item of equipment, including the additional weight necessary for installation, should be specified.

A recommended form for equipment lists is shown in Fig. .0-4. This list shows a method of handling items in a simplified form which may include a number of related models and which makes it unnecessary to prepare separate lists for each model.

In the checking of equipment lists by the Authority, particular attention is paid to ascertain:

(1) The effects of the equipment installation on the aircraft structure. The Authority ascertains that satisfactory analyses and drawings are submitted for such items as batteries, radios, flares, etc.
(2) That items for which approval is required by CAR 05, such as position lights, safety belts, etc., are of an approved type; and
(3) The effects of the equipment installation weights on the longitudinal balance of the glider.

d. Preliminary Weight and Balance Report. A report should be submitted in which the range of center of gravity locations for which rating is sought is determined versus weight and with respect to suitable reference planes or lines. The critical CG positions and the weights determined in this report should be used in the balancing tail load computations (see CAAM 05.218). If the CG limits determined during the Type Inspection (see CAR 05.730) appreciably exceed the design limits, use of the values determined during the Type Inspection should be substantiated in so far as they affect the design computations.

e. Balance Diagram. The preliminary weight and balance report (see CAAM 05.032d) should include a diagram showing the location of the centers of gravity of the component parts of the glider and its contents, the location of a suitable reference chord for the wing system, and the location of the assumed center

Item No.	Item	Make and Model	When Required	Horizontal Datum	Weight on Models A	B	C
1	Wheel	Drawing No. 15014	Always	7	5	5	5
2	Safety Belt	Rusco-AE200	Always	-7	-	-	-
3	Position Lights	Grimes - A	Night Flying	-	-	-	-
4	Battery	4 Smith Drycells	Optional	2	-	-	2.5
5	Battery	4 Jones Drycells	Optional	5	2	2	-
6	Battery Box	Drawing No. 67052	Optional	2	-	-	.5
7	Battery Box	Drawing No. 67025	Optional	5	.5	.5	-
8	Instrument board light	Drawing No. 63009	Night Flight				
9	Instruments						
	a. Airspeed		Always				
	b. Altimeter		Instrument flight				
	c. Variometer		Instrument flight				
	d. Compass		Instrument flight				
	e. Bank and turn		Instrument flight				

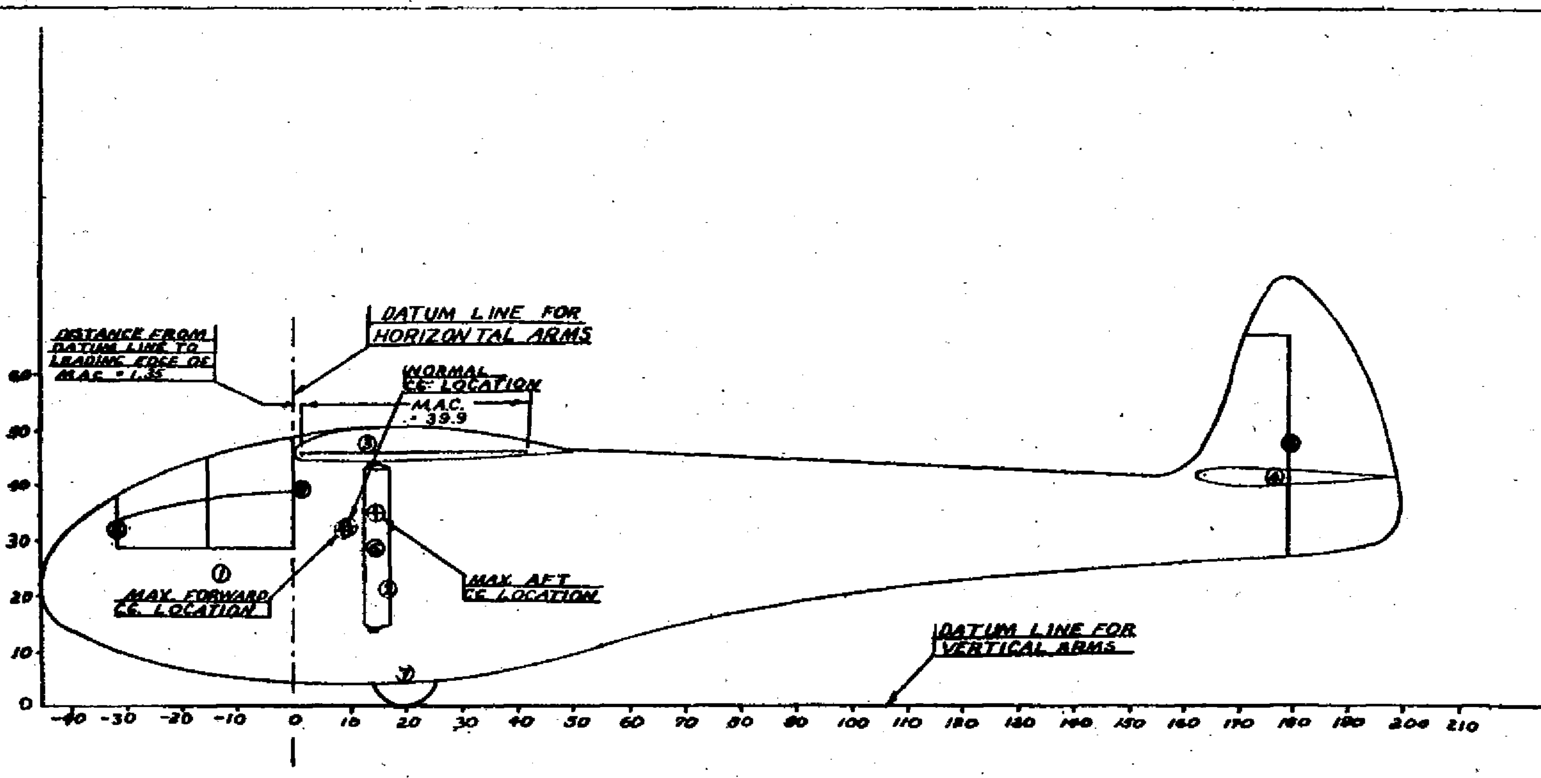
FIG. .0-4 — RECOMMENDED FORM FOR EQUIPMENT LISTS of pressure of the horizontal tail. The locations of the above items should be indicated by reference to suitable horizontal and vertical planes or lines.

The balance diagram should include the following:

(1) Outline of the glider (side view).
(2) Horizontal and vertical scales. For horizontal arms it is preferable that the datum be chosen at some definite and accessible point on the glider, such as a point at the leading edge of the wing. This facilitates checking in the field. Distances aft of this point are generally assumed as positive and those forward as negative. For vertical arms the datum may be chosen at some arbitrary location below the extended landing gear, so that all distances are up and positive.
(3) Item designations. These designations (usually members) should correspond with the designation used in the weight table (CAR 05.032(f)) and, when possible, with the designations used in the equipment list and weight and balance reports.
(4) Item locations. The various items should be shown in the proper location on the outline noted in (1) above. Such location may be indicated by a small circle together with the item designation noted in (3) above.
(5) Dimensions. The following should be given.
    (a) Length of MAC.
    (b) Horizontal distance from datum to the leading edge of the MAC.
    (c) Vertical distance from the datum to a definite and accessible point on the glider.

A suggested form for the balance diagram, including a weight table, is shown in Fig. .0-5.

f. <u>Weight Table</u>. The preliminary weight and balance report (see CAAM 05.032d) should include a table or list of the weights of all component parts of the glider and its contents. The weights shown in the weight table should be broken down and itemized so that they may readily be used in the structural reports of the individual components, such as wing, fuselage, etc. (See Fig. .0-5).



WEIGHT AND BALANCE TABLE

MAC = 39.9"; Leading edge of MAC 1.35" back of Leading edge at root.

ITEM	WEIGHT	ARM	MOMENT +	MOMENT −
1 Pilot and Chute	190	-14.0		-2660
2 Fuselage	75	16.5	1240	
3 Wing	170	13.6	2300	
4 Horizontal Tail Surfaces	11	172.6	1900	
5 Vertical " "	8	177.5	1400	
6 Struts	15	13.9	208	
7 Wheel	9	18.9	187	
8 Instruments	5	-32.4		-162
9 Luggage or Equipment	18	4.0	72	
	500		7501	-2722
			-2720	
			4781	

Normal C.G. Location

170# Pilot
20# Parachute
18# Luggage

$$\bar{x} = \frac{4781}{500} = 9.56 = 24\%\ MAC$$

Maximum Forward C.G. Location

190# Pilot
20# Parachute
No luggage

$$\bar{x} = \frac{4781 - 18(4+14)}{500} = 8.91 = 20.8\%\ MAC$$

Maximum Aft C.G. Location

100# Pilot
18# Luggage
No parachute

$$\bar{x} = \frac{4781 - (190-100)(14)}{500 - (190-100)} = 18.7 = 31.0\%\ MAC$$

0-13

g. Structural Report (structural analysis and/or test reports).

General. The structural report referred to in CAR 05.032(g) is a report in which the strength of the structure is determined with reference to the strength requirements specified in CAR 05. The structural report should include the computations of the limit loads required in CAR 05 and should demonstrate the ability of the structure to develop the factors of safety required in CAR 05 with respect to the limit loads either by suitable analytical methods (stress analysis) or by reference to authenticated test data (test reports) or by a combination of both. (See CAR 05.3). The structural report should also include all computations or tests necessary to prove compliance with the miscellaneous structural requirements of CAR 05.

Although there is no requirement specifying that reports submitted to the Authority must be checked for arithmetical accuracy prior to transmittal, it should be noted that CAR 05.050 provides for discontinuance of the examination of reports in the event that they contain errors which render them unsatisfactory. In order to avoid delays in the checking of data, it is recommended that all computations be given an independent check by the manufacturer and be signed by both the original computer and the checker. A preferred form of title page, and of title block of subsequent pages, of the structural reports is shown in Fig. .0-6.

(1) Structural Analysis Reports. Computations submitted as a part of the structural report should include a table, or tables, including the minimum margins of safety computed for all structural members, and should bear the signature(s) of the responsible engineer(s). Such tables should include the name of the member involved (such as spar), design condition, margin of safety, and page number reference.

(2) Test Reports. Test reports submitted as a part of, or referred to in, the structural report should bear the signature of the Authority's representative who witnessed the tests, except in the case of minor tests, in which case the applicant's certification that the report accurately represents the complete results of the

(Sample title page for structural reports)

NAME OF MANUFACTURER

GLIDER MODEL NO. ___

REPORT NO. ___

TITLE OF REPORT

Date ________ Prepared by ________

Revisions ________ Checked by ________

________ Approved by ________

________ Witnessed by ________

____

____

Subject ________ Prepared by ________ Date ________ Checked by ________	MANUFACTURER	Page ________ Model ________ Report No. ________

(Sample title block for report pages)

REF. CAAM 05.052g.

FIG. .0-6 — SAMPLE TITLE PAGE AND TITLE BLOCK tests will be accepted. In the preparation of such test reports, it is essential that they contain, as a minimum, the following information:

(a) Determination of test loads (including references to pertinent page and number of stress analysis report).
(b) Distribution of loads during test.
(c) Description and photographs of test set-up. (Detail views are necessary in some cases.)
(d) Description of method of testing.
(e) Results of tests, including photographs of structures found to be critical.
(f) Log of deflection data. (Including sketches to show location of points at which deflections were measured.)
(g) Curves of deflection vs. load for each such point to permit determination of any evidence of permanent set.
(h) Signature of Authority's representative(s) and manufacturer's engineer(s) who witnessed the test.
(i) Signature of company engineer(s) responsible for test report.

05.041 STRUCTURAL INSPECTION

In the event that, during the engineering inspection and flight tests, modifications of the primary structure are required which may involve repeating certain tests (exclusive of flight tests) and the preparation of stress analyses and revised drawings, the manufacturer should contact the Authority for comments and rulings, so that the flight tests may be resumed with a minimum of delay. This may, in certain cases, make it possible for the manufacturer to proceed with flight tests prior to the approval of reports covering tests which have been repeated.

05.052 TYPE INSPECTION AUTHORIZATION

Before the type inspection is authorized, the following should be fulfilled:

a. Completion of examination of the structural report and drawings and correction by the applicant of all errors and omissions of a serious nature.

b. Completion, and acceptance by the Authority, of all structural tests required by CAR 05 to prove compliance with the structural requirements.

c. Submission of test reports conforming to the requirements of CAR 05.342.

05.053 TYPE INSPECTION PROCEDURE

Before conducting any flight tests, the representative of the Authority will complete a ground inspection to determine that all items affecting the safety of flight are satisfactory.

a. The statement of conformity required by CAR 05.053(a) should be executed upon the form supplied by the Authority. On this form the manufacturer's chief engineer (or other responsible technical representative) certifies that the glider submitted for type inspection has been manufactured in accordance with the latest technical data submitted to and approved by the Authority (including all revisions and additions required by the Authority in connection with authorization of the type inspection) except for any deviations therefrom, which are listed and described.

b. The weight and balance (actual) required by CAR 05.053 (b) should cover the determination of the weights and center of gravity locations for which certification is desired. The actual weight and balance of the glider should be determined in the presence of the representative of the Authority.

c. The applicant's flight test report required by CAR 05.053(c) should be a detailed report of the flight tests of the glider involved. This report should be signed by the applicant's test pilot and should certify that the glider has been flown by him in all maneuvers required by CAR 05.7 for proof of compliance with the flight requirements and found to conform with such requirements. In order to expedite checking of this report, it is advisable that the results of the applicant's flight tests be recorded on a form of the type used by the Authority inspectors in connection with the type inspection. Copies of this form may be obtained from the local engineering inspector.

The glider will be subjected to such flight tests as are necessary to prove compliance with the flight and operation requirements specified in CAR 05.7.

If the ground inspection or flight test is discontinued because of an unfavorable characteristic or serious defect, the following procedure should be followed:

(1) The inspector will note each unsatisfactory item upon a form supplied for the purpose, with sufficient detail to make it clear to all concerned.
(2) One copy of such form will be transmitted to the manufacturer.
(3) The manufacturer should advise the Authority when the glider, incorporating the necessary changes, will be available for continuance of the type inspection.
(4) The manufacturer should furnish the Authority with technical data descriptive of all structural changes except those of an obviously minor nature. The type inspection should not be resumed until such changes have been approved by the Authority.

05.060 CHANGE, REPAIR OR ALTERATION OF CERTIFICATED GLIDERS.

This matter is covered by CAAM 01.33.

05.061 CHANGES AFFECTING TYPE CERTIFICATE

a. Drawing Changes. When a revised drawing is submitted to the Authority and gliders previously constructed according to the original drawings are not to be changed, the manufacturer should keep a record of the serial numbers of the gliders to which the change applies Corrected pages of the drawing lists should be submitted in duplicate for each model to which the revision applies. Alternate installations should be so designated and properly indicated on the drawing lists.

b. Minor Changes. Minor changes which obviously do not impair the structural strength or reliability of the glider nor affect its flying characteristics may be approved by authorized inspectors of the Authority. Shop drawings showing such changes should be forwarded to the Authority for record purposes.

05.1 DEFINITIONS (INCLUDING STANDARD SYMBOLS, VALUES, AND FORMULAS)

A DEFINITIONS ADDITIONAL TO THOSE GIVEN IN CAR 05.1.

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 CAAM 05.123-C.

2. Margin of Safety, MS. 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.
B-	b - distance between spars, fraction of chord; span of wing.
C - chord, feet; coefficient; constant;	c - subscript, "chord".
CP - center of pressure, fraction of chord.	
CG - center of gravity.	
D - subscript "drag".	
	e - unit weight, psf.

F - force, lbs.

f - unit stress, psi; front spar location, fraction of chord; subscript, "fuselage".

fps - feet per second.

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

h - distance measured perpendicular to MAC, in terms of MAC.

i - subscript, "induced".

j - position of wing CG, fraction of chord; factor of safety.

K - a general factor.

L - subscript, "lift" or "level".

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

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

MAC - mean aerodynamic chord.

mph - miles per hour.

MS - margin of safety.

N - subscript, "normal force".

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

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

psf - pounds per square foot.

psi - pounds per square inch.

q - dynamic pressure, psf.

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

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

S - design wing area, sq. ft. (See CAR 05.101)

T - tail load, lbs.

U - gust velocity, fps.

V - glider speed, mph.

W - total weight of glider and contents, lbs.

r - rear spar location, fraction of chord.

s - wing loading, psf; subscript, "stall".

t - subscript, "tail".

u - subscript, "ultimate".

v - glider speed, fps.

w - unit pressure, psf. subscript "wing"

$\bar{w}$ - average unit pressure, psf.

x - distance measured parallel to MAC in terms of MAC; subscript*.

y - subscript, "yield".

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

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

Δ - (delta) - increment.

ρ - (rho) - mass density of air.

*With subscript "x", n refers to a limit load applied parallel to the basic wing reference chord. (See Fig. .2-11).

C STANDARD VALUES AND FORMULAS.

Air Density:

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

Dynamic Pressures:

2. $q = 1/2\ \rho_o V_i^2$

$= .00119\ v_i^2$ (where v_i is "indicated" speed, fps.)

$= .00256\ V_i^2$ (where V_i is "indicated" speed, mph.)

Basic Glider Parameters:

3. $s = W/S$

Aerodynamic Coefficients:

4. $C_R = (C_L^2 + C_D^2)^{1/2}$

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

6. $C_c = - C_L \sin \alpha + C_D \cos \alpha$ (positive rearward)

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

Forces, Unit Loadings, and Couples:

8. $F_x = C_x S q$ (Where x may be R, L, D, N, c, or M)

9. $F_D = S_D q$

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

$= C_M S\ q\ C$

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

12. $n = F/W$

Speeds:

13. $V_s = 19.76\ (s/C_{L\ max})^{1/2}$ (mph) = indicated stalling speed

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

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

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

17. $m = m_6 \dfrac{4}{3 + 6/R}$ where m_6 = slope of the lift curve when aspect ratio equals 6.

m = slope of the lift curve when aspect ratio equals R.

05.100 GROSS WEIGHT, W.

The gross weight should include the weights of all items of equipment, such as instruments, parachutes, radio, etc. in addition to pilot and passenger weights (see CAR 05.90).

05.101 DESIGN WING AREA.

1. In computing the design wing area the plan form of tapered or eliptical wings may be represented by a number of trapezoids closely approximating the actual plan form and having an equivalent area.

2. The application of CAR 05.101 to several typical cases is illustrated in Fig. .1-1.

05.105 INDICATED AIRSPEED, V.

For structural analysis purposes, all airspeeds are expressed as "indicated" airspeeds. The "indicated" airspeed is defined as the speed which would be indicated by a perfect airspeed indicator; namely, one which would indicate true airspeed at sea level under standard atmosphere conditions.

05.107 DESIGN AIRCRAFT TOW SPEED, V_{ta}.

Except in very special cases, the design aircraft tow speed V_{ta} will be equal to the design gliding speed V_g (CAR 05.108) for Class I and II gliders (see CAR Table 05-1). (The aircraft towing of Class III is not permitted.)

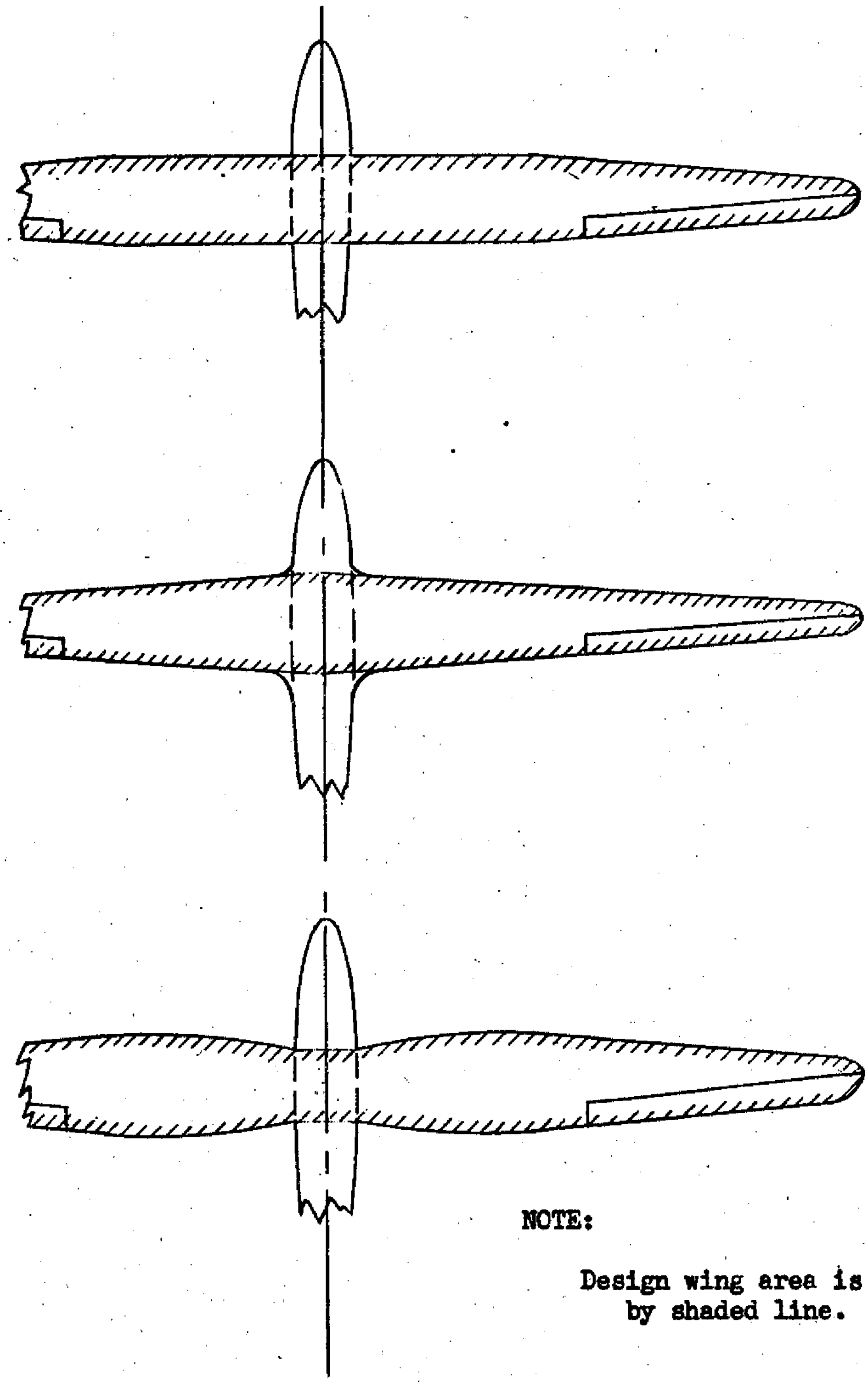
05.108 DESIGN GLIDING SPEED, V_g.

The choice of the design gliding speed V_g should be governed by the type and intended use of the glider. Minimum values are specified in CAR Table 05-2.

05.123 AERODYNAMIC COEFFICIENTS.

A GENERAL.

1. The coefficients are "absolute" (non-dimensional) coefficients.



NOTE:

Design wing area is outlined by shaded line.

FIG. .1-1 TYPICAL DESIGN WING AREAS
(Ref. CAR 05.101)

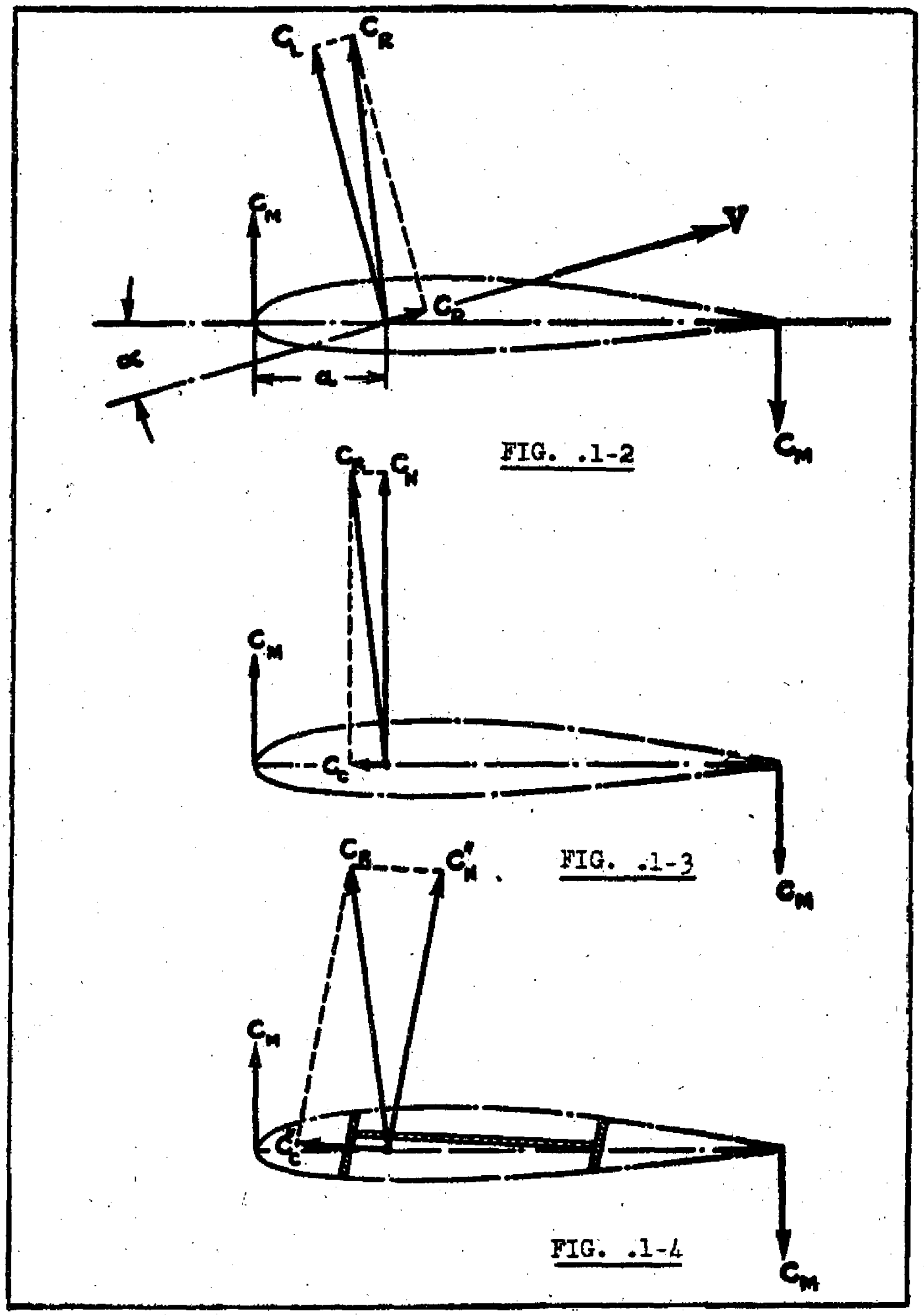
When applied to an airfoil surface of given area, they represent the ratio between an actual average unit pressure referred to the projected area of the airfoil and the dynamic pressure corresponding to the flight condition being considered. The subscripts denote the direction along which the force is measured, but do not change the basic reference area.

2. The subscripts "L" and "D" refer to directions normal to and parallel to the relative wind, while the subscripts "N" and "c" refer to directions respectively normal to and parallel to the basic wing chord. Subscript "R" refers to the direction of the resultant force. These factors are illustrated in Figs. .1-2 and .1-3. When the planes of the drag truss and lift trusses do not coincide respectively with the planes of the basic chord and the plane of the normal forces, a correction is necessary before the coefficients can be used directly in the wing analysis method outlined in CAAM 05.3. The corrected coefficients are obtained by resolving the resultant force coefficients into components in the plane of the lift truss and drag truss, as shown in Fig. .1-4. The effect on the chord coefficient may be considerable, but the correction for C_N will usually be negligible.

3. The moment coefficient may be considered to be of the same nature as the force coefficients if the force to which it corresponds is applied as a couple at the leading and trailing edges of the wing chord, as shown in Figs. .1-2, .1-3, and .1-4. A positive moment coefficient requires an upward force at the leading edge, as shown. The conversion of center of pressure position into a moment coefficeent about any given point can be easily accomplished by means of Eq. 7 in CAAM 05.1-C. It should be noted that the center of pressure and the moment coefficient are alternative in nature and can not both be used at the same time.

B DETERMINATION OF STANDARD AIRFOIL CHARACTERISTICS.

The standard airfoil characteristics for conventional airfoils are obtainable from NACA Reports and Technical Notes. In so far as structural analysis purposes are concerned, these characteristics need not be corrected for aspect ratio, provided that the characteristics are based on an aspect ratio of 5.0 or greater. The slope of the lift curve, m (CAAM 05.1-B), must be corrected for aspect ratio when used in conjunction with the gust load factor formula (CAAM 05.2121). Aspect ratio corrections should also be made for performance computations. It should be clearly understood that for a given lift coefficient C_L, the values of the angle of attack α , and the drag coef-



FIGS. .1-2, .1-3, .1-4 -- ILLUSTRATION OF AIRFOIL FORCE COEFFICIENTS
(Ref. CAR 05.125-A2)

ficient C_D will be fictitious unless the aspect ratio R of the glider in question is the same as that used in the wind tunnel tests. For a given normal force coefficient C_N, however, the values of the chord force coefficient C_c, moment coefficient C_M, and center of pressure CP will remain practically constant for all values of the aspect ratio normally used.

C COMPUTATION OF ADDITIONAL CHARACTERISTICS

As indicated in Table .1-I, certain additional characteristics are desirable. These characteristics may be determined as follows:

a. The normal force coefficient C_N can be determined from Eq. 7, CAAM 05.1C. The steps involved are shown as items 1 through 8 of Table .1-I.

b. The chord force coefficient C_c is determined from Eq. 6, CAAM 05.1C. The steps are outlined as items 9 to 11 of Table .1-I.

c. (1) The moment coefficient about the aerodynamic center C_{M_a} is usually given in airfoil reports. In such cases the C_{M_a} curve for a conventional airfoil may be represented as a horizontal straight line (see Fig. .1-5), and the value of C_{M_a} will be constant for all values of C_L. If the moment coefficient about another point x (such as the quarter chord point) is desired, it may be determined by the following equation:

$$C_{M_x} = C_{M_a} - (a - x)\ C_N$$

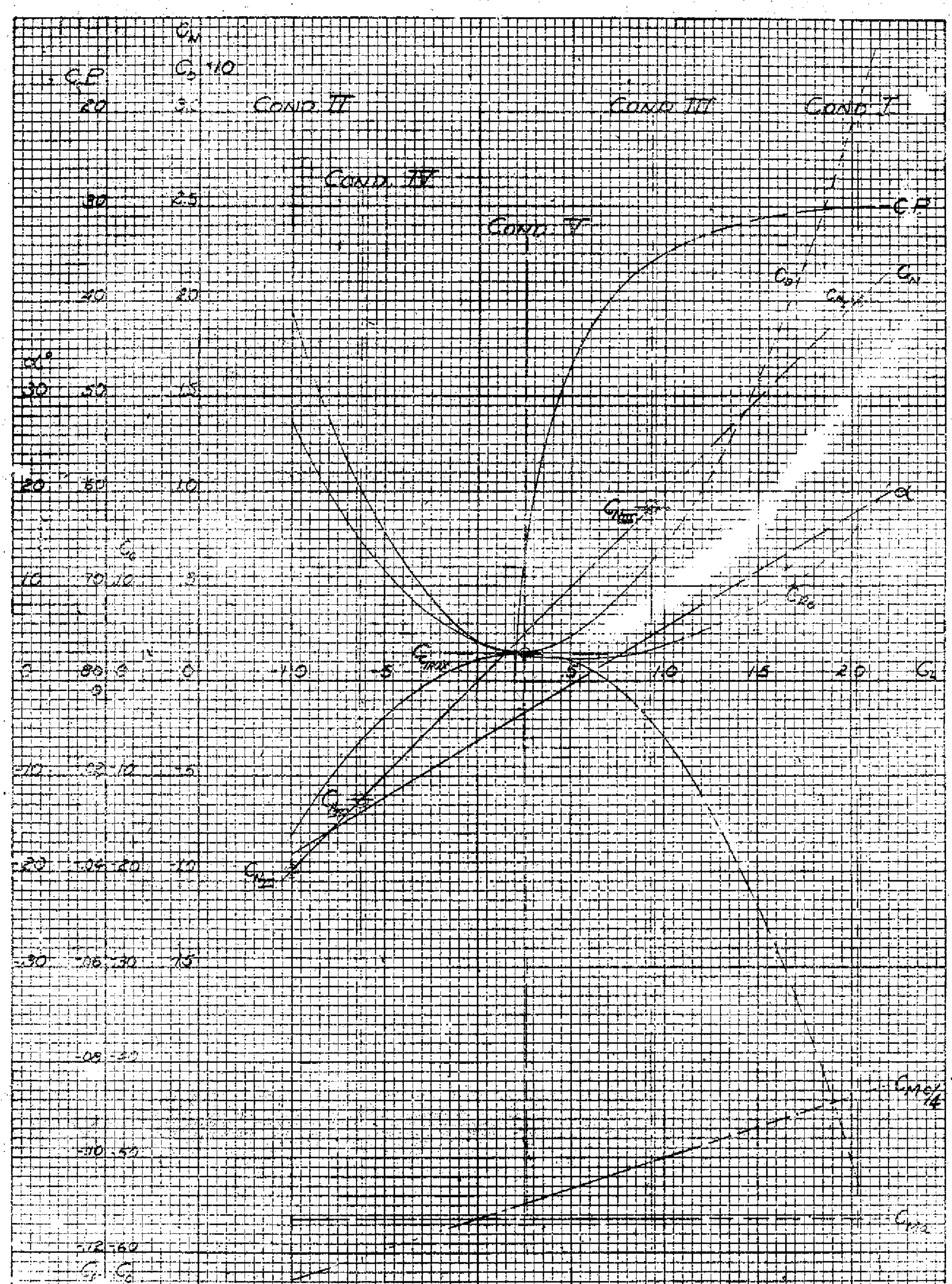
where x is the point in question in terms of fraction of the chord.

(2) If the moment coefficient about the aerodynamic center C_{M_a} is not known but the moment coefficient about the quarter chord point $C_{M_{c/4}}$ versus C_L is given, a straight line can be drawn to fit the $C_{M_{c/4}}$ curve as closely as possible (see Fig. .1-5). The average value of C_{M_a} is then obtained from the straight line where $C_L = 0$. The position of the aerodynamic center can then be obtained from the following equation:

	COMPUTATION OF AIRFOIL CHARACTERISTICS																	
1	C_L	-1.0	-.8	-.6	-.4	-.2	0	.2	.4	.6	.8	1.0	1.2	1.4	1.6	1.8	2.0	2.2
2	a																	
3	C_D																	
	C_D ext. = (15) + (16)																	
4	COS a = COS(2)																	
5	SIN a = SIN(2)																	
6	C_L COS a = (1) x (4)																	
7	C_D SIN a = (3) x (5)																	
8	C_N = (6) + (7)																	
9	C_L SIN a = (1) x (5)																	
10	C_D COS a = (3) x (4)																	
11	C_C = (10) - (9)																	
12	C P																	
	C P ext. = $a - C_{M_a}/(8)$																	
13	$C_{M_{c/4}}$ = (.25 - (12)) x (8)**																	
14	C_{M_a} = (13) (a - .25) x (8)**																	
15	$C_{D_i} = C_L^2/\pi R = (1)^2/\pi R$*																	
16	C_{D_0} = (3) - (15)																	

*R = Aspect ratio of airfoil model used in NACA Reports (or equivalent).

** For most modern, conventional airfoils, the CM_a may be assumed constant for all values of CL and can be obtained from NACA publications (See CAAM 05.10-c(1)).



AIRFOIL CHARACTERISTICS CURVES

FIG. 1-5

(Ref CAAM 05123-D)

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

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

d. The values of a and C_{M_a} can also be obtained directly from the CP curves as outlined in items 12 and 13 of Table .1-I in which the values of $C_{M_{c/4}}$ are determined. However, unless the values of CP can be accurately read from the curve, this procedure may give misleading results. These values can be plotted against C_L and the process for determining a and C_{M_a} can then be carried out as outlined in c(2) above. In any event, the operations should be confined to the values of C_L which lie on the substantially straight portion of the lift coefficient curve.

e. For unconventional airfoils which do not have a well-defined aerodynamic center it may be advisable to determine local values of C_{M_a}. The value of C_{M_a} can be separately determined for any given value C_L by means of the equation:

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

Provision is made under item 14 of Table .1-I for determining local values of C_{M_a}.

D EXTENSION OF CHARACTERISTIC CURVES

In the accelerated flight conditions the maximum values (positive and negative) of C_L shown on the basic airfoil characteristic curve will be exceeded without the breakdown of the flow characterized by the change in slope of the lift curve. The airfoil characteristic curves to be used for structural analysis purposes can be extended to represent the effect of a sudden change in angle of attack by the following approximations:

a. Referring to Fig. .1-5, extend the curve of angle of attack α to higher values of C_L by means of a straight line coinciding with the substantially straight portion of the original curve. The values of α so obtained should be entered in Table .1-I under item 2. (The dashed lines in Fig. .1-5 indicate extended values).

b. Determine the induced drag coefficient C_{D_i} as outlined in item 15 of Table .1-I. R is defined in CAAM 05.123-B. It should be noted that R, as used in Table .1-I, is the aspect ratio of the wind tunnel model and not necessarily the aspect ratio of the glider in question.

c. Determine the profile drag coefficient, C_{D_O}, item 16 of Table .1-I. Plot these values for the original straight line portion of the C_L curve and extend the curve, so obtained, along the same general path followed at the lower values of C_L, as shown in Fig. .1-5. Enter the values of C_{D_O}, thus obtained, under item 16.

d. Extend the C_D curve by determining the values for item 3 of Table .1-I as indicated.

e. The C_{M_a} curve may be extended as a horizontal straight line.

f. The extended values of C_N and C_c are determined as indicated under items 8 and 11 of Table .1-I, using the extended values of C_D.

g. The C.P curve may be extended by means of the equation:

$$CP = a - C_{M_a}/C_N$$

using the extended values of C_N.

h. If no data are available for the airfoil characteristics in the negative C_L range, the C_N and C_M curves may be obtained in this region by extending the curves as straight lines. In this case, the C_c curve may be obtained by assuming that the C_D curve is symmetrical and then computing values of C_c in accordance with item 11 of Table .1-I.

05.2 STRUCTURAL LOADING CONDITIONS

05.201 DEFORMATIONS

1. Detrimental permanent deformations are, in general, considered as those which correspond to stresses in excess of the yield stress. The yield stress is defined as the stress at which the permanent strain is equal to 0.002 inches per inch.

2. In determining permanent deformations from static test results, the effects of slippage or permanent deformation of the jig may be accounted for if properly measured.

05.211 DESIGN AIRSPEEDS

The values of the design airspeeds specified in CAR Table 05-2 are minimum values. In certain cases it may be desirable to use larger values; e.g., utility type gliders designed under Class III (see CAAM 05.01) or high performance type gliders designed under Class II. (See CAAM 05.70j regarding the possibility of accidentally exceeding the placard speed.) In order to provide for a high auto-winch tow placard speed, it may be advantageous to use both a higher design gliding speed and a constant positive load factor over the design speed range (see CAAM 05.2131).

05.212 LOAD FACTORS

It should be noted that the flight load factors referred to in CAR 05.212 are wing or air load factors. Also, that the net or dead weight load factors should be obtained from balancing computations, such as outlined in CAAM 05.218.

05.2120 MANEUVERING LOAD FACTORS

The limit maneuvering load factors specified in CAR 05.2120 are, in general, minimum values. In certain cases, it may be advisable to use higher values, e.g., when a higher auto-winch tow placard speed is desired (see CAAM 05.211 and CAAM 05.2130).

05.2121 GUST LOAD FACTORS

The limit gust load factors may be obtained from the following formula:

$$n = 1 + \Delta n$$

$$= 1 + \frac{KUVm}{575s}$$

where Δn = limit load factor increment

K = gust reduction factor (see Fig. .2-1)

U = gust velocity, fps. (Note that the "effective" sharp-edged gust equals KU)

V = indicated airspeed, mph

s = W/S, wing loading, psf. (See CAR 05.102 for definition)

m = slope of lift curve, C_L per radian, corrected for aspect ratio, R. (See CAAM 05.1-C, Eq. 17).

05.2130 **BASIC FLIGHT ENVELOPE**

In so far as CAR 05.2130 is concerned, the basic flight envelope or V-n diagram is a locus of points representing the limit wing load factors and the corresponding velocities for the design criteria specified in CAR 05.2131 through 05.2133. Such a basic flight envelope should be plotted on rectangular coordinate paper (common graph or cross-section paper) to a suitable scale with the velocity, V (mph), as the abcissa and the limit wing load factor, n(g-units), as the ordinate. A sample basic flight envelope is shown in Fig. .2-2.

05.2131 When applying the requirements specified in CAR 05.2131, the following procedure should be followed:

a. Plot the following equation on the positive portion.

(line 1 of Fig. .2-2):

$$n = (V/V_s)^2 \geq 1.0$$

where n = maximum possible positive limit wing load factor at the speed, V (mph).

V_s^2 = stalling speed squared corresponding to a C_{Lmax} (dynamic) of 2.0.

$= 391\ s/C_{Lmax} = 196\ s$

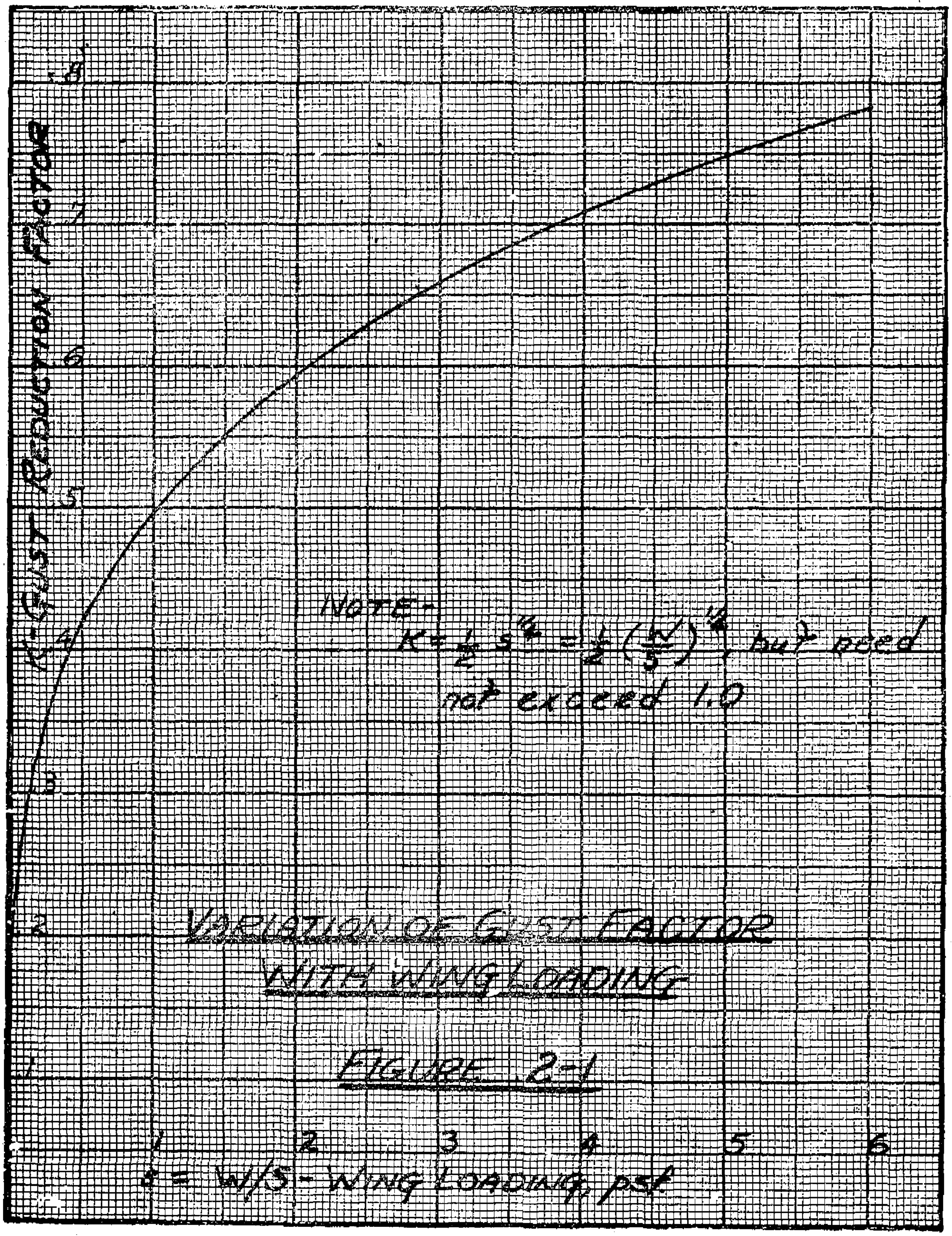
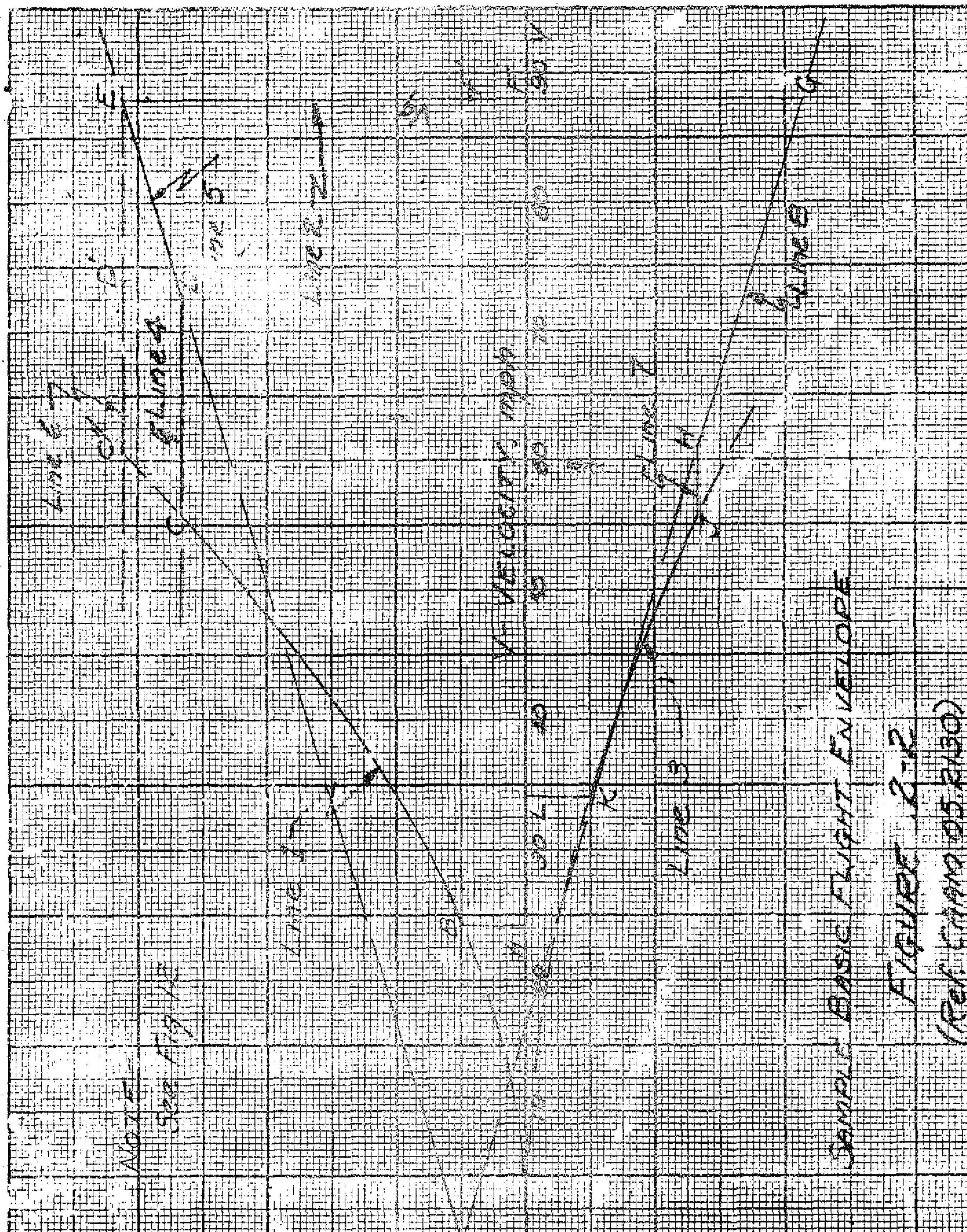


FIG. .2-1 - GUST REDUCTION FACTOR



SAMPLE BASIC FLIGHT ENVELOPE

FIGURE 2-2

(Ref. CAM 05.2130)

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

c. Plot the following equation on the negative portion (line 3 of Fig. .2-2).

$$n = -(V/V_s)^2 \geq -1.0$$

where n = maximum possible negative unit wing load factor at the speed V(mph).

V_s^2 = stalling speed squared corresponding to a C_{Lmax}(dynamic) of 1.0 (negative).

$= 391\ s/C_{L_{max}} = 391\ s$

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

a. Draw a horizontal straight line through the value of n specified in CAR Table 05-2 (item 5). (Line 4 of Fig. .2-3). If a higher value of the auto-winch tow placard speed is desired; i.e., higher than specified in CAR Table 05-1, the value of n should not be less than that obtained from the following formula:

$$n = \frac{1}{(s-e)}\left[\frac{(V'_{taw} + 5)^2}{256} - e\right]$$

where V'_{taw} = desired auto-winch tow placard speed (mph). See CAR 05.211 for definitions of other symbols.

b. Draw a straight line through points V = 0, n = 1 and $V = V_g$, $n = n_{III}$ (point E of Fig. .2-2). The value of n_{III} may be obtained from the formula given in CAAM 05.2121 when $V = V_g$ and U = 30 fps. This line is labeled line 5 in Fig. .2-2.

c. For reasons previously mentioned, the designer may desire to use a constant load factor (positive) over the flight speed range (see CAR 05.2131). In this case line 4 of Fig. .2-2 may be replaced by line 6. Line 6 is a horizontal straight line through point $V = V_g$, $n = n_{III}$ (point E of Fig. .2-2).

It should be noted that the positive portion of the basic flight envelope is represented by the curve ABCDEF or ABC'D'EF, as the case may be, of Fig. .2-2.

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

a. Draw a horizontal straight line through the value of n specified in CAR TABLE 05-2 (Item 6). (Line 7 of Fig. .2-2).
b. Draw a straight line through points $V = 0$, $n = 1$ and $V = V_g$, $n = n_{IV}$ (point C of Fig. .2-2). The value of n_{IV} may be obtained from the formula given in CAAM 05.2121 when $V = V_g$ and $U = -30$ fps. This line is labeled line 8 in Fig. .2-2.

It should be noted that the negative portion of the basic flight envelope is represented by the curve FGHJKL of Fig. .2-2.

05.2134 In general, an investigation of the following specific basic flight conditions, which correspond to points on the basic flight envelope (see Figs. .2-2 and .2-3), will insure satisfactory coverage of the critical loading conditions:

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

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

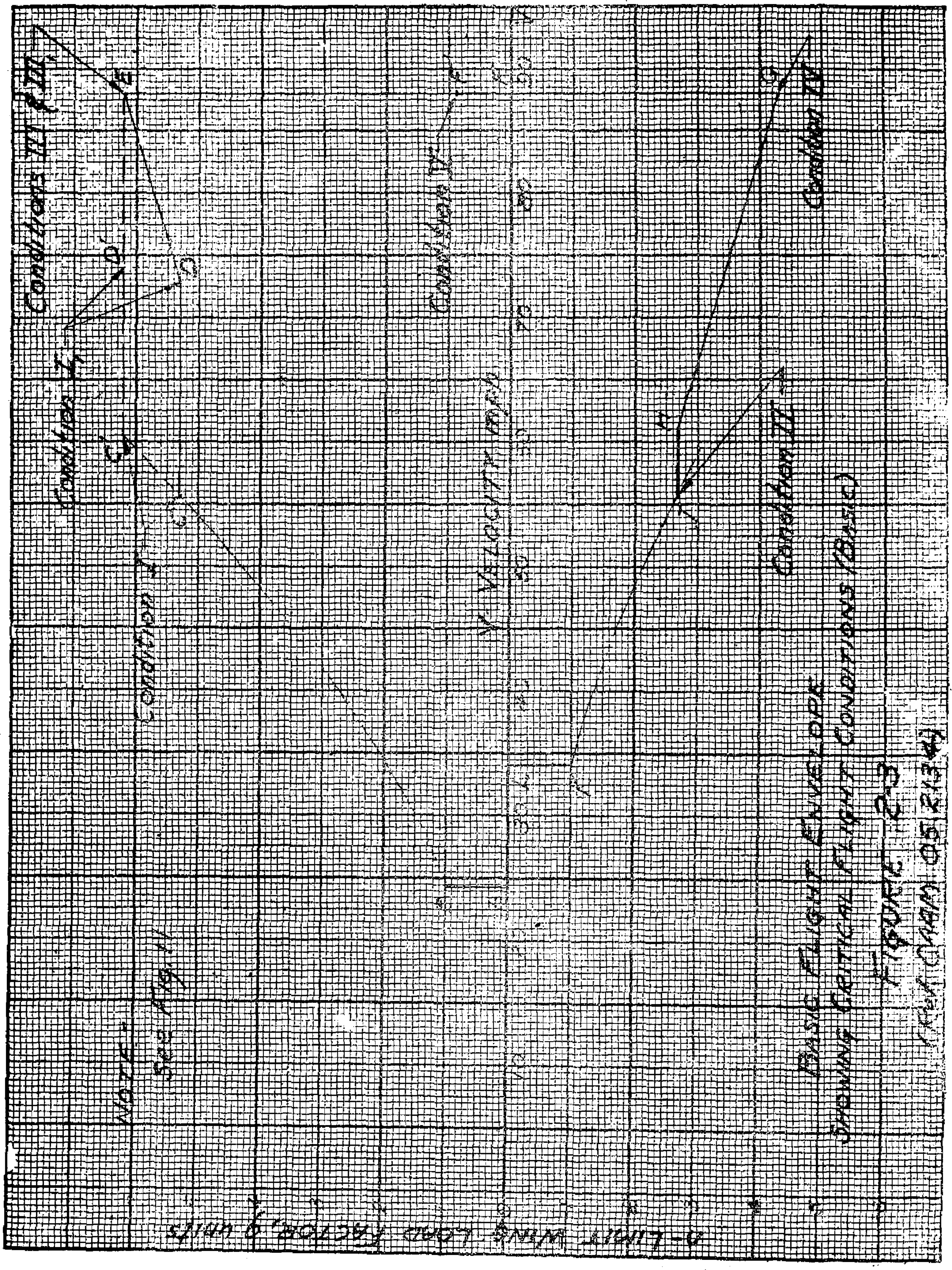
where n_I = wing load factor corresponding to point C or C'.

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

(2) C_C = value corresponding to C_{N_I}, as obtained from the airfoil characteristics curves (see Fig. .1-5).

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

b. Condition I_1 (Modified Positive High Angle of Attack). This condition need only be investigated in special cases. The condition corresponds to point D or D' on the basic flight envelope, provided that point D or D'



Basic Flight Envelope
Showing Critical Flight Conditions (Basic)
Figure 2-3
(Ref. CAM 05.2134)

is approximately midway between point C and line 2 (the midpoint should be used when point D is near either of the points C or E), or point D' when point C' is used in Condition I above. The aerodynamic characteristics to be used in this investigation should be determined as follows:

(1) $C_{N_{I_1}} = \frac{n_I \, s}{q_{I_1}}$

where q_{I_1} = dynamic pressure corresponding to V_{I_1}, which in turn corresponds to point D and D', as the case may be.

(2) C_C = value corresponding to $C_{N_{I_1}}$ (may be assumed equal to zero, if positive).

(3) C P or C_M = value corresponding to $C_{N_{I_1}}$

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

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

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

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

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

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

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

(3) $C_M' = C_M - 0.01$, where C_M is the actual value corresponding to $C_{N_{III}}$.

e. Condition III_1 (Modified Positive Low Angle of Attack). In order to cover the effects of limited use of the ailerons at V_g on the wings and wing bracing, such structure should be investigated for the following:

(1) $C_{N_{III_1}} = C_{N_{III}}$

(2) C_c = value corresponding to $C_{N_{III_1}}$.

(3) $C_M{}'$ = value obtained from Fig. .2-4, where C_M is the value corresponding to $C_{N_{III_1}}$. $C_M{}'$ need only be applied to that portion of the span incorporating ailerons, using the basic value of C_M determined in Condition III over the remainder of the span.

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

(1) $C_{N_{IV}} = \dfrac{n_{IV}\ s}{q_{IV}}$

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

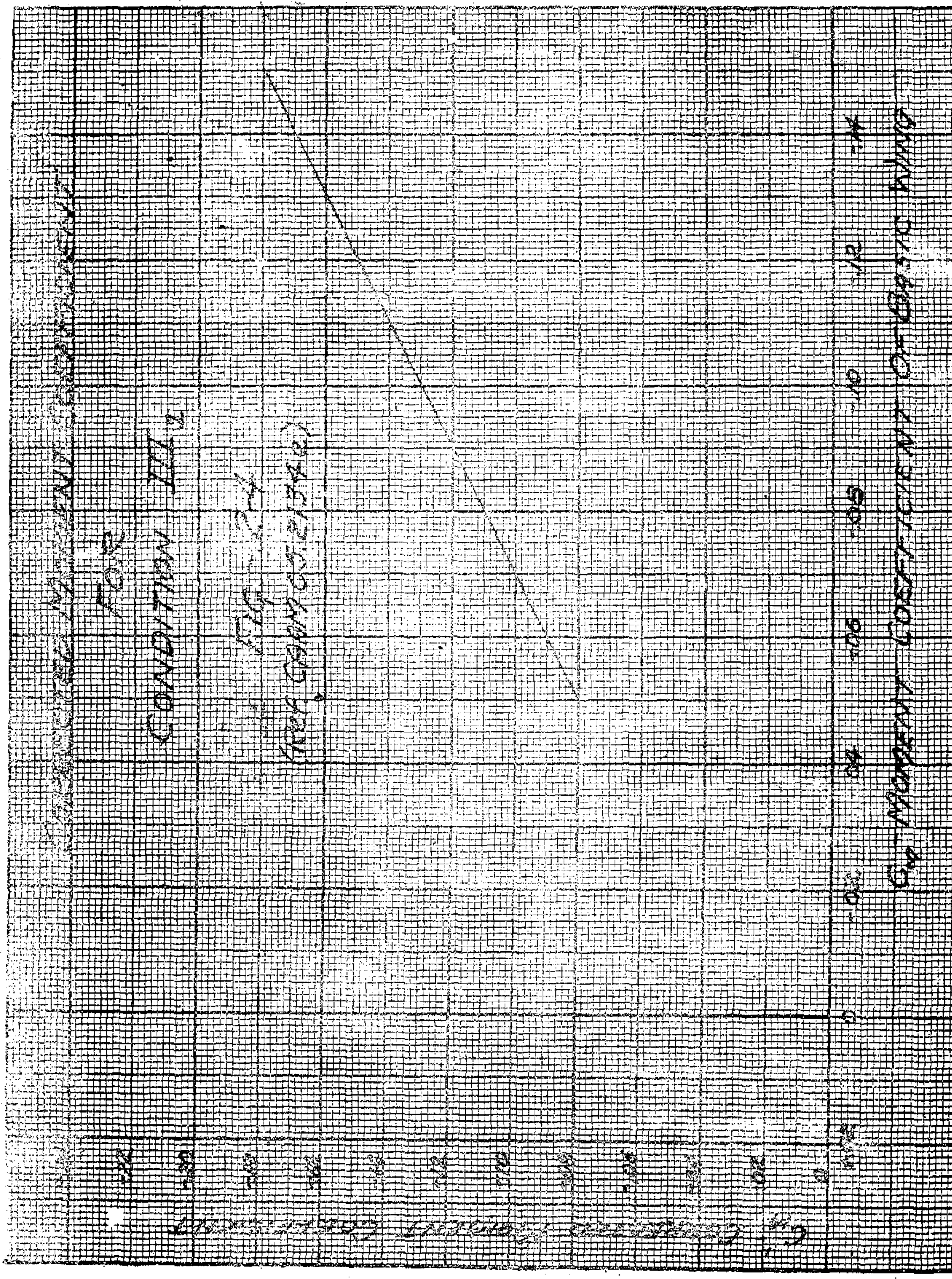
(3) $C_M{}' = C_M - 0.01$, where C_M is the actual value corresponding to $C_{N_{IV}}$.

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

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

(2) $C_c{}' = C_c$ max. (positive) + 0.01.

(3) $C_M{}' = C_M - 0.01$, where C_M is actual value corresponding to C_{N_V}.


FOR
CONDITION III
Cm MOMENT COEFFICIENT OF BASIC WING

05.2140 GENERAL

A INTERNALLY BRACED WINGS

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

a. The average value of C_M' used in design conditions III and IV (CAAM 05.2135-d and -f) should equal or exceed the quantity

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

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

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

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

b. The average value of C_c used in design Condition V (CAAM 05.2135g) should equal or exceed the quantity

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

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

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

2. The foregoing interpretation applies to normal installations in which the flap is inboard of the ailerons, or in which a full span flap is used. For other arrangements it will be

necessary to submit additional computations if it is desired to prove that flap conditions are not critical.

3. In all cases an investigation is required of the local wing structure to which the flap is attached, using the flap design loads as determined from CAR 05.224. The strength of special wing ribs used with split flaps, and the effect of flap control forces, should also be investigated. Reference should be made to current NACA reports and noted for acceptable flap data.

B EXTERNALLY BRACED WINGS

1. The designer or manufacturer of externally braced gliders equipped with trailing edge flaps should contact the Authority for applicable loading conditions for the flap conditions.

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

a. Modify Conditions I and III (see CAAM 05.2135 -a and -d) and the most critical negative condition by assuming 100 per cent of the air load to be acting on one side of the glider and 40 per cent on the other.

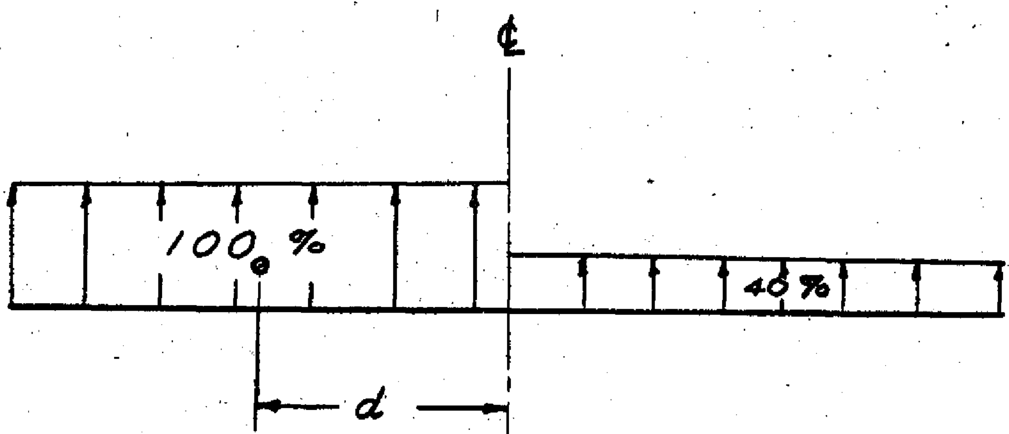
b. Assume the moment of inertia of the entire glider is effective.

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

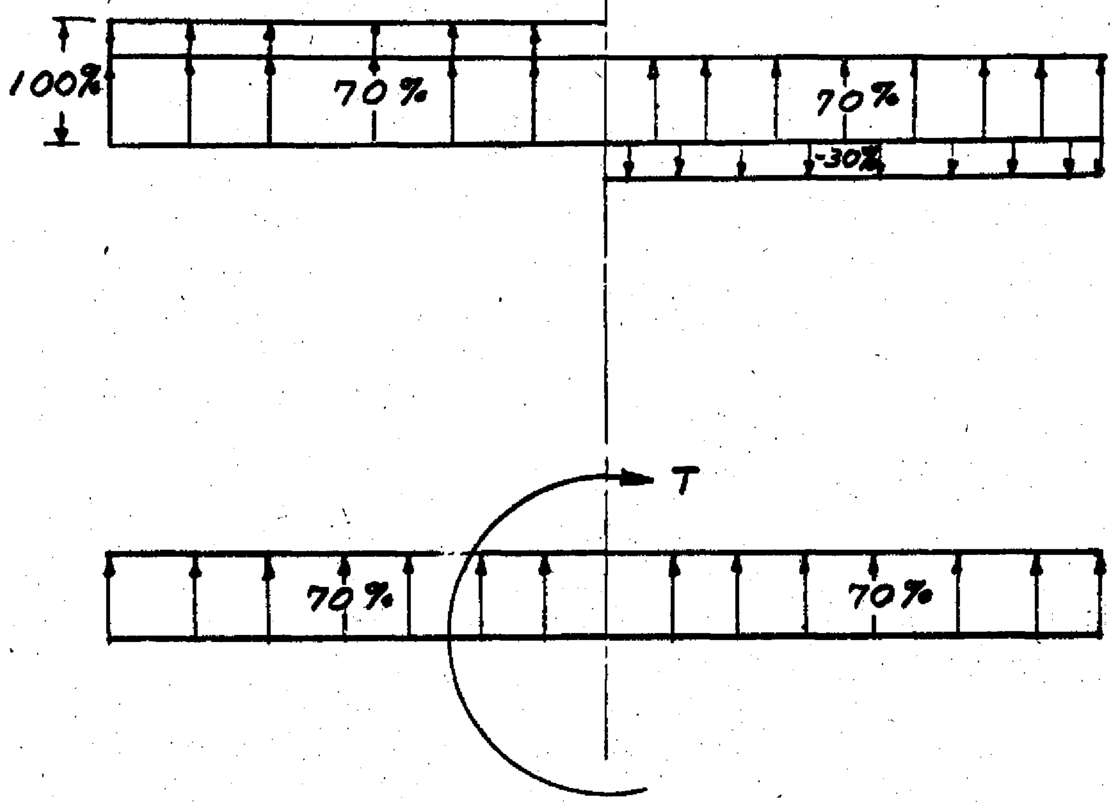
The angular acceleration α resulting from the torque T may be obtained from the following formula:

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

where I_x = the moment of inertia of the glider (and its contents) about the X, or longitudinal axis.



Note:
d = centroid of loading.



Note:
T = .60W x d
where: W = the normal load on one panel.

FIG. .2-5

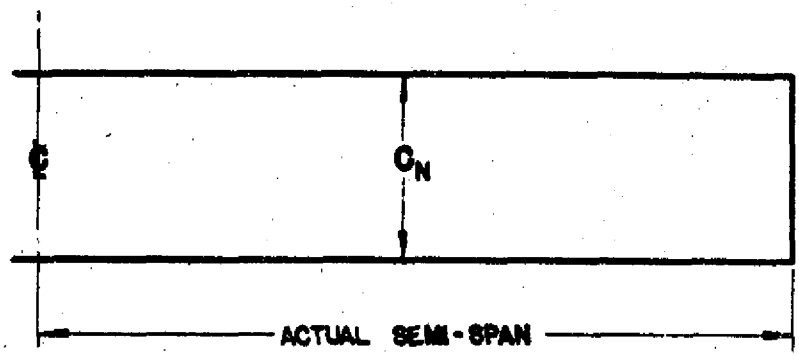


FIG. 2-6: SPAN DISTRIBUTION - NO TIP LOSS.

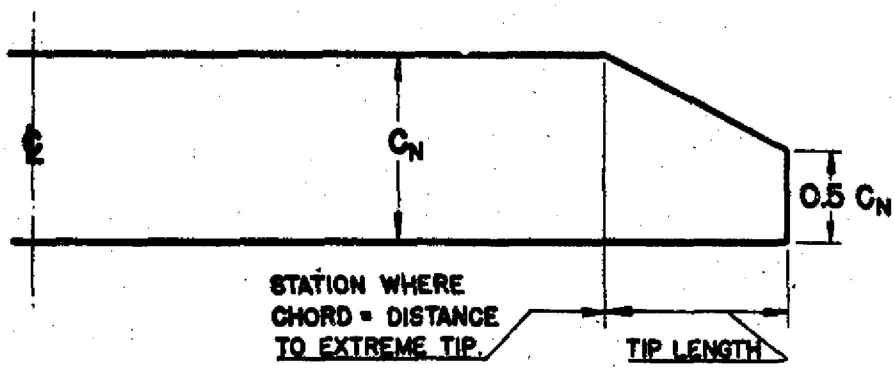


FIG. .2-7: SPAN DISTRIBUTION - WITH TIP LOSS

NOTE: Above figures apply only to wings which have mean taper ratios greater than 0.5 and without aerodynamic twist.

FIGS. .2-6 and .2-7 - SPAN DISTRIBUTION OF C_N FOR WINGS.

Army-Navy-Civil publication ANC-1 (1), "Spanwise Air Load Distribution" (obtainable from the Superintendent of Documents, Washington, D. C., at the nominal sum of 60 cents), in NACA Technical Reports Nos. 572, 585 and 631 and NACA Technical Note No. 606.

3. When the normal force coefficient is assumed to vary over the span, the values used should be so adjusted as to give the same total normal force as the design value of C_N acting uniformly over the span. (See CAAM 05.217-C for additional information).

4. When Figs. .2-6 and .2-7 are used the chord coefficient shall be assumed to be constant along the span.

5. For wings having aerodynamic twist (not geometric twist), it is very important that the effect of twist (wash-out) be considered in the investigation of the wings for the negative conditions (see CAAM 05.2134).

B CHORD DISTRIBUTION

1. The approximate method of chord loadings outlined in CAAM 05.3 for the testing of wing ribs is suitable for conventional two spar construction if the rib forms a complete truss between the leading and trailing edges. An investigation of the actual chord loading should be made in the case of stressed-skin wings if the longitudinal stiffeners are used to support direct air loads. In some cases it is necessary to determine the actual distribution, not only for total load but for each surface of the wing. If wind tunnel data are not available, the methods outlined in NACA Reports Nos. 383, 411, and 465 are suitable for this purpose. These methods consist in determining the "basic" pressure distribution curve at the "ideal" angle of attack and the "additional" pressure distribution curve for the additional angle of attack. These curves can be coordinated with certain values of C_L, so that the final pressure distribution curve can be obtained immediately for any C_L. Curves of this nature for several widely-used airfoils can be obtained directly from the NACA.

2. Leading Edge Loads. On high speed gliders the leading edge loads developed may be exceptionally severe, particularly the "down" loads which are produced by negative gusts when flying at the design gliding speed. The magnitude of such loads can be estimated, without determining the entire chord distribution, by the method outlined in NACA Report No. 413.

3. Effects of Auxiliary Devices. When a design speed higher than required is used in connection with wing flaps or other auxiliary devices, it will be necessary to determine the chord distribution over the entire airfoil. The effect of any device which remains operative up to V_g should be carefully investigated. This applies particularly to auxiliary airfoils, spoilers, and fixed slots.

C. DETERMINATION OF RESULTANT AIR FORCES

1. A general method is outlined below for determining the mean effective value of the normal force coefficient, the average moment coefficient, location of the mean aerodynamic center and value of the mean aerodynamic chord. These factors are needed in order to determine the balancing loads for various flight conditions (see CAAM 05.218). The most general case will be considered, so that certain steps can be omitted when simpler wing forms or span load distribution curves are involved.

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

3. For convenience and clarification, Table .2-I has been developed and the various curves obtained as a part of this method are illustrated in Figs. .2-8, .2-9 and .2-10. It should be particularly noted that when the area under a curve is referred to, the area should be expressed in terms of the product of the units to which the curve is drawn. The procedure is as follows:

a. Fig. .2-8(a) illustrates the actual wing plan form, plotted to a suitable scale. This should agree with the definition of design wing area given in CAR 05.101.

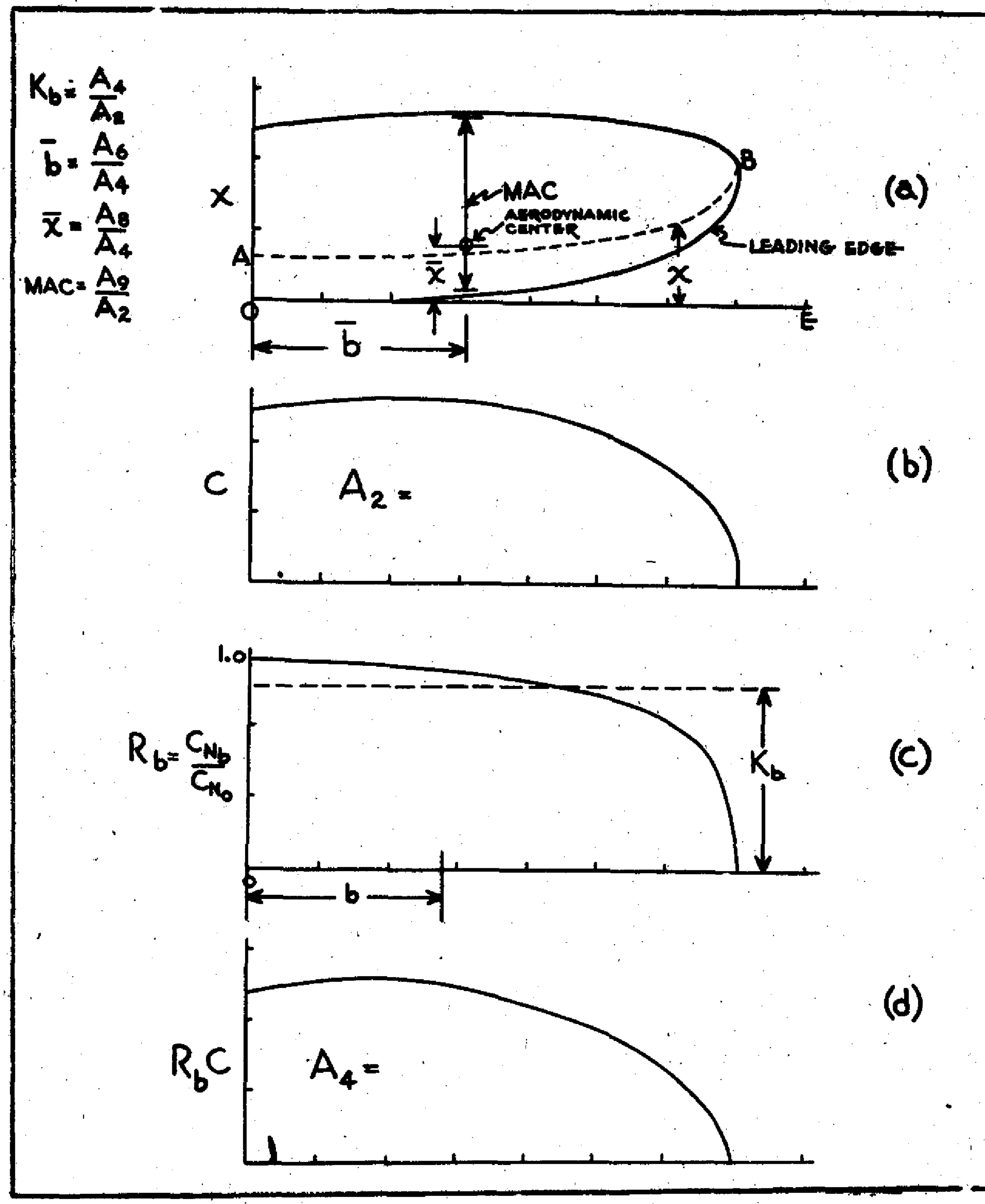


FIG. .2-8 - DETERMINATION OF EFFECTIVE NORMAL FORCE COEFFICIENT
(REF. CAAM 05.217-C3)

b. Fig. .2-8(b) shows the variation of wing chord, C, with span. The values of C are entered in Table .2-I as item (2). The area of the figure should be accurately determined and converted to the proper units. It should be one-half the value of design wing area.

c. Fig. .2-8(c) represents an assumed span distribution curve. The factor R_b represents the ratio of the actual C_N at any point to the value of C_{N_0} at the root of the wing. Values of R_b from this curve are entered in Table .2-I under item (3).

d. Fig. .2-8(d) is obtained by plotting R_bC (item 4) in Table .2-I against span. The ordinates of this curve are proportional to the actual force distribution over the span. The area under curve .2-8(d) should be accurately determined and expressed in the proper units. K_b, the ratio of the mean effective C_N to the value of C_{N_0} (at the root), is obtained by dividing the area under curve .2-8(d) by the area under .2-8(b), using the same units of measurement for each area. This value of K_b is indicated by the dotted line on curve .2-8(c).

e. To determine the location of the mean aerodynamic center along the span, Fig. .2-9(a) is drawn. The ordinates are obtained by multiplying the ordinates of curve .2-8(d) by their distance along the span, as shown in item 5, of Table .2-I. The area under curve .2-9(a) divided by the area under curve .2-8(d) gives the distance from the wing root to the chord on which the mean aerodynamic center of the wing panel is located. This distance is indicated on Fig. .2-8(a) by the dimension $\overline{b}$.

f. The locus of the aerodynamic centers of each individual wing chord is plotted on Fig. .2-8(a) as the dotted line A-B. In Table .2-I the distance "x" from the base line O-E to the line A-B is entered under item 6.

g. Fig. .2-9(b) is now plotted, using as ordinates the values of R_b Cx obtained from item 7 of Table .2-I. The area under curve .2-9(b) divided by the area under curve .2-8(d) gives the distance of the mean aerodynamic center from the base line O-E in Fig. .2-10(a). This distance is indicated as $\overline{x}$ on that figure.

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

K_b =

$\overline{b}$ =

$\overline{x}$ =

MAC =

TABLE .2-1

DETERMINATION OF RESULTANT AIR FORCES
(Ref. CAAM 05.217-C)

h. If it is assumed that the moment coefficient about the aerodynamic center of each individual chord is constant over the span, the magnitude of the mean aerodynamic chord is determined by means of Fig. .2-9(c). The ordinates for this curve are determined from item 8 of Table .2-I. The area under curve .2-9(c) divided by the area under curve .2-8(b) gives the value of the mean aerodynamic chord. By way of illustration, it is drawn on Fig. .2-8(a) so that its aerodynamic center coincides with the location of the mean aerodynamic center of the wing panel.

i. In cases in which wing flaps or other auxiliary high-lift devices are used over a portion of the span it is desirable to obtain the mean effective moment coefficient. This is the coefficient to be used for balancing purposes in connection with the mean aerodynamic chord previously determined under the assumption of a uniform moment coefficient distribution. In Table .2-1 under item 9 the local values of the moment coefficient about the aerodynamic center are entered. These are also plotted as Fig. .2-10(a) to illustrate a type of distribution which might exist.

j. Fig. .2-10(b) is plotted from the values indicated under item 10 of Table .2-I. The area under this curve divided by the area under curve .2-9(c) gives the mean effective value of the moment coefficient for the entire wing panel.

k. <u>In the case of twisted wings</u> a different span distribution exists for each angle of attack. The location of the resultant forces can, however, be determined in the above manner for any known span distribution.

05.218 BALANCING LOADS

A GENERAL

1. The basic design conditions (see CAAM 05.2134) must be converted into conditions representing the external loads applied to the glider before a complete structural analysis can be made. This process is commonly referred to as "balancing" the glider and the final condition is referred to as a condition of "equilibrium". Actually, the glider is in equilibrium only in steady unaccelerated (constant

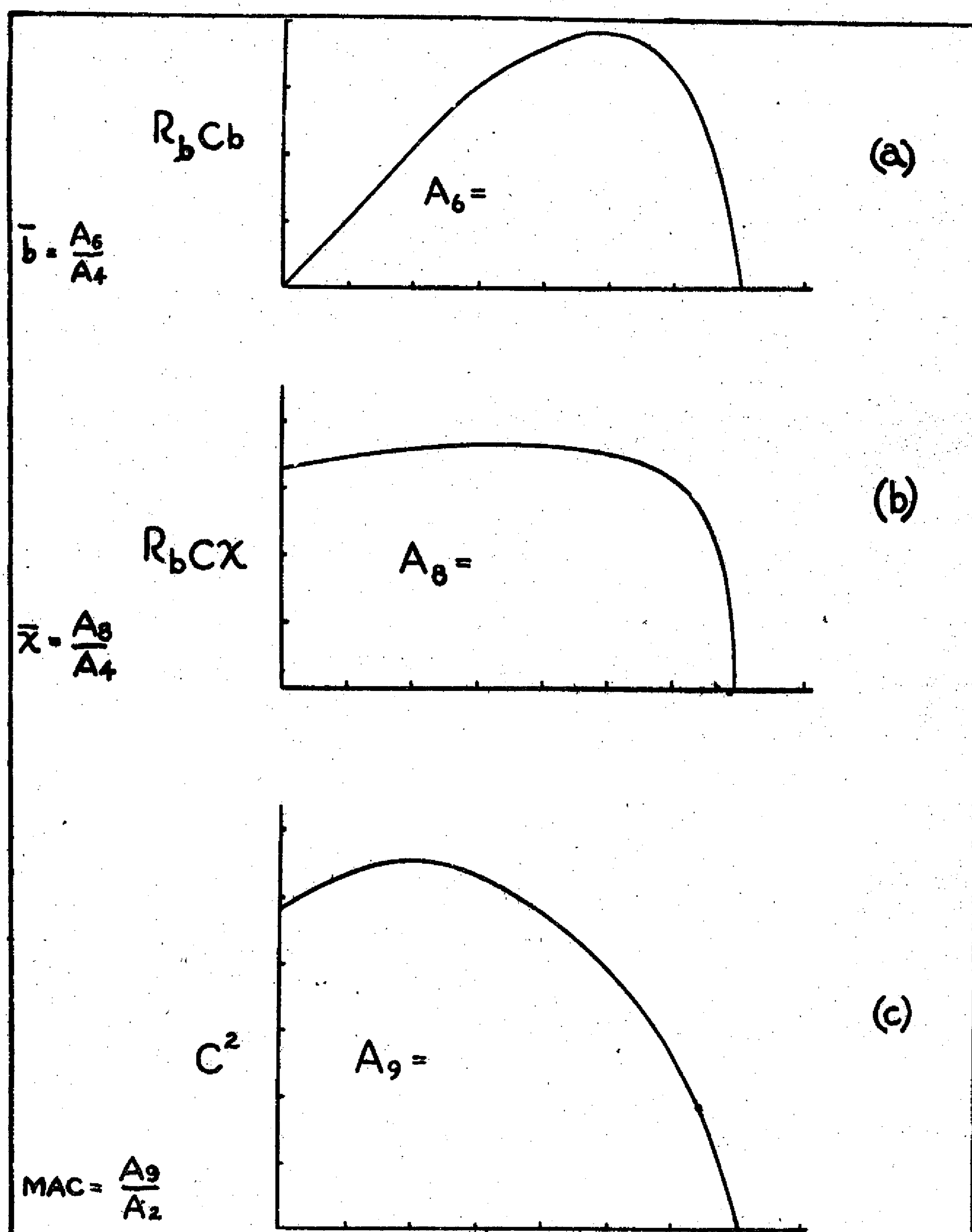


FIG. .2-8 – DETERMINATION OF MAC AND MEAN AERODYNAMIC CENTER
(Ref. CAAM 05.217-C3)

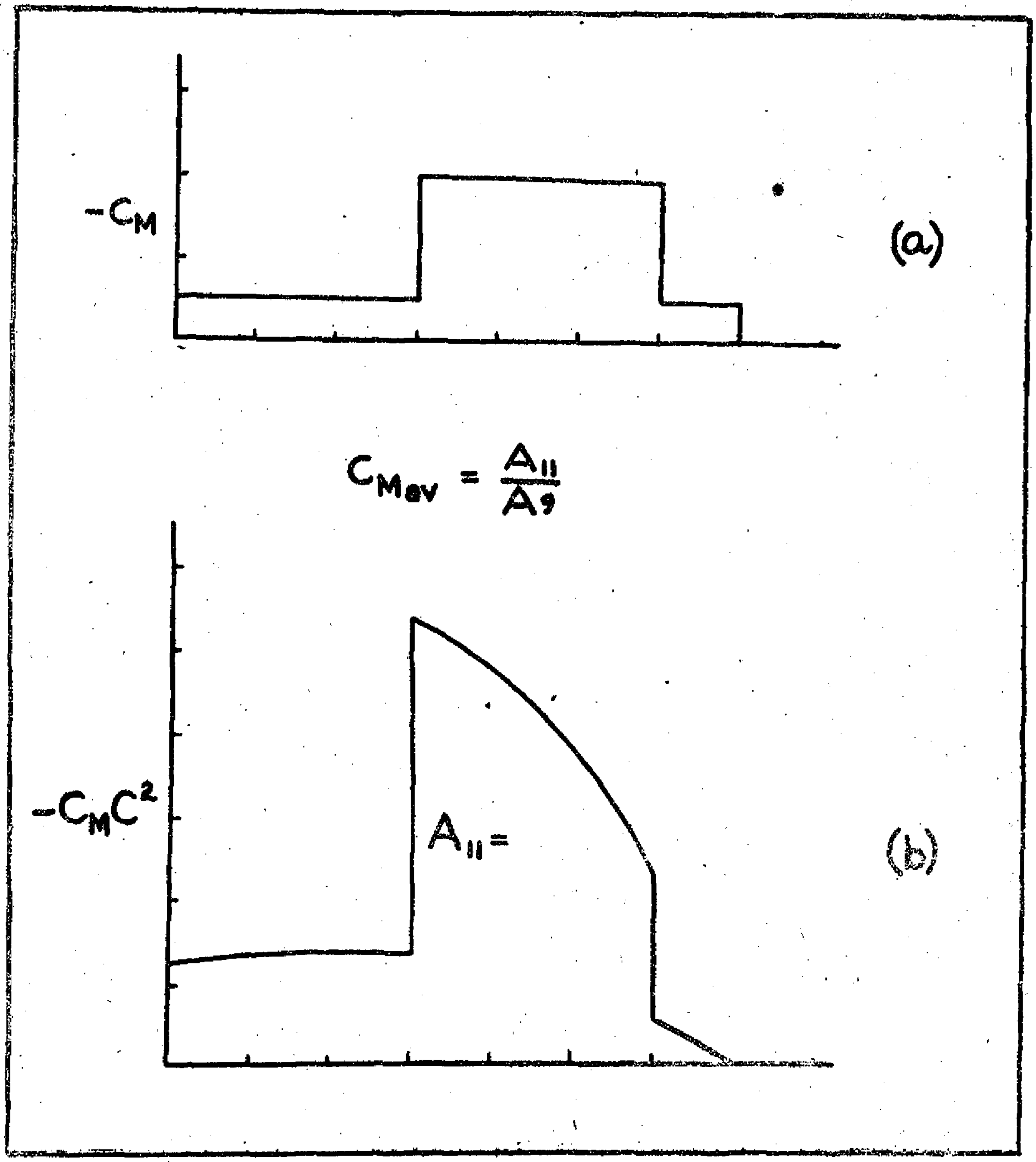


FIG. .2-10 - DETERMINATION OF MEAN EFFECTIVE MOMENT COEFFICIENT
(Ref. CAAM 05.217-C3)

speed) flight; in accelerated conditions both linear and angular accelerations act to change the velocity and attitude of the glider. It is customary to represent a dynamic condition, for structural 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 glider in a certain direction is "n" times the total weight of the glider, each item of mass in the glider is assumed to act on the glider 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 glider 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 glider. In general, such an analysis is not necessary except in certain cases for 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 glider 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 glider. 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 GLIDER

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

a. It is assumed that the limit load factors specified for the basic flight conditions (CAAM 05.2134) are <u>wing</u> load factors. A solution is therefore made for the <u>net</u> load factor acting on the whole glider. 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 glider.

b. 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% 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.

c. In Fig. .2-11 the external forces are assumed to be acting at three points only. The assumption can generally be made that the fuselage drag acts at the center of gravity. When more accurate data are available, the resultant fuselage drag force can of course be computed and applied at the proper point. In cases where large independent items having considerable drag are present it is advisable to extend the set-up shown in Fig. .2-11 to include the additional external forces.

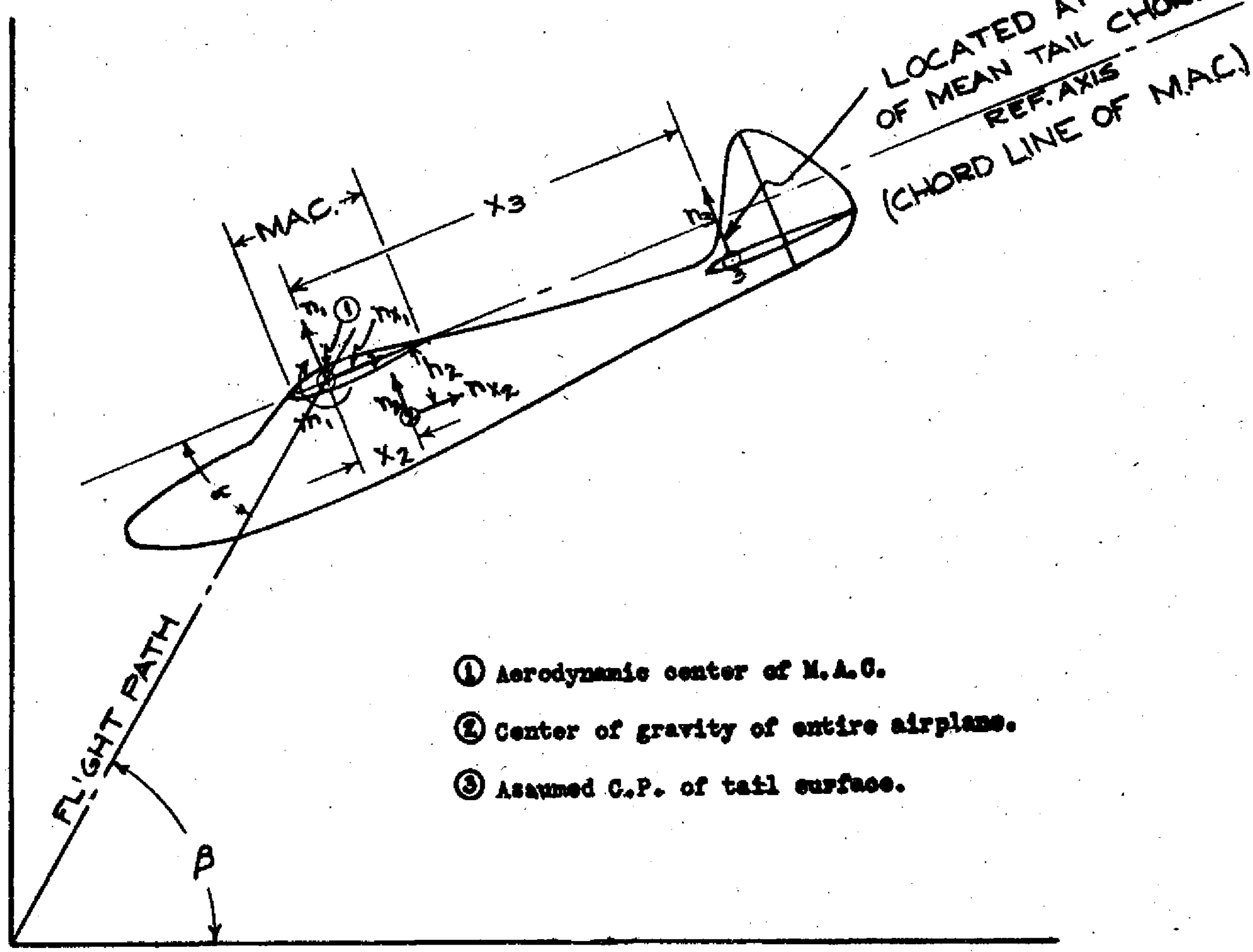
2. As shown in Fig. .2-11, a convenient reference axis is the basic chord line of the mean aerodynamic wing chord. (The basic chord line is usually specified along with the dimensions of the airfoil section). The determination of the size and location of the MAC is outlined in CAAM 05.217-C.

3. A tabular form will simplify the computations required to obtain the balancing loads for various flight conditions. A typical form for this purpose is shown in Table .2-II. In using Fig. .2-11 and Table .2-II the following assumptions and conventions should be employed:

a. If known distances or forces are opposite in direction from those shown in Fig. .2-11, a negative sign should be prefixed before inserting in the computations. For instance, in the case of a low-wing monoplane, h_2 will have a negative sign. The direction of unknown forces will be indicated by the sign of the value obtained from the equations. A negative value of n_3 will usually be determined from the balancing process, indicating a down load on the tail. For conditions of positive acceleration the solution should give a negative value for n_2, as the inertia load will be acting downward. The convention for m_1 corresponds to that used for moment coefficients; that is, when the value of C_M is negative m_1 should also be negative, indicating a diving moment.

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

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



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).

<u>All distances are expressed in terms of the M.A.C.</u>

FIG. .2-11 - BASIC FORCES IN FLIGHT CONDITIONS
(Ref. CAAM 05.218-B)

4. Computation of Balancing Loads. In Table .2-II 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 design 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. .2-II. (Condition IV, Neg. L.A.A., will usually result in the largest balancing tail load.)

a. When the full-load center of gravity position varies appreciably the glider should be balanced for both extreme positions unless it is apparent that only one is critical. In general, only one center of gravity need be considered for single place gliders. In special 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 .2-II:

(4) 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 05.212 and CAAM 05.2134).

(8) Determine C_C as specified in CAAM 05.2134. See also Eq. 6, CAAM 05.1-C.

(10) The value of C_M' is specified in CAAM 05.2134. See also CAAM 05.217-C in cases involving wing flaps.

(12) The net tail load factor n_3 is found by a summation of moments about point (2) of Fig. .2-11, from which the following equation is obtained:

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. 05-1 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 $\bar{w}$ refers to the average limit pressure over the total effective area of the vertical surface. The total load acting is therefore equal to $\bar{w}$ 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.

05.224 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 05 apply only when the flaps are not used at speeds above a certain predetermined design speed. As noted in CAR 05.732, 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.

05.225 SPECIAL DEVICES

1. In lieu of wind tunnel data, it is recommended that spoilers and their attachment structures be designed for the limit loading obtained from the following formula:

$$w_{sp} = .0052\ V_{sp}^2$$

where: w_{sp} = the limit loading, psf.

V_{sp} = the indicated airspeed at which maximum operation of the spoilers is assumed, mph.

It should be assumed that the load is uniformly distributed over the surface. NACA Technical Memorandum No. 926 contains information pertaining to dive control brakes.

05.230 GENERAL

The control forces specified in CAR 05 are of an arbitrary nature; hence they may prove to be somewhat irrational in certain cases. In general, however, they represent simplified requirements which will result in satisfactory control systems. If he so desires, the designer may use a more rational loading for the design of the control system. The following loadings are considered satisfactory. The control systems may be designed for limit loads 25% greater than those corresponding to the limit loads specified in CAR 05 for the control surfaces to which they are attached, assuming the movable surfaces to be in that position which produces the greatest load in the control system, except that the loads should not be less than those listed below:

a. Elevator: 75 lbs. fore-and-aft

b. Rudder: 100 lbs. on one pedal only and 200 lbs. on each pedal simultaneously

c. Aileron: 50 lbs. laterally or as a part of a couple applied to the control wheel

The control forces specified should be applied to the entire control system including the control surface horns. The multiplying factor of safety of 1.20 (CAR 05.271) need not be applied to the fittings in the control system.

05.233 In regard to the distribution of the control force between the ailerons, the following assumptions are considered suitable:

a. For non-differential ailerons, 75% of the stick force or couple should be assumed to be resisted by a down aileron, the remainder by the other aileron; also, as a separate condition, 50% should be assumed to be resisted by an up aileron, the remainder by the other aileron.

b. For differential ailerons, 75% of the stick force or couple should be assumed to be resisted by each aileron in either the up or down position, or rational assumptions based on the geometry of the system should be made.

05.234 FLAP AND AUXILIARY CONTROL SYSTEMS

1. It is recommended that the following limit forces be used as minimum values for the design of flap and auxiliary control systems:

a. Flaps .... 50 lbs. (See note below)

b. Spoilers .... 50 lbs. (See note below)

c. Hand operated brakes .... 75 lbs.

d. Foot operated brakes ....100 lbs.

NOTE: The force used for the design of flap and spoiler control systems should not be less than 1.25 times the force corresponding to limit load used for the design of the surfaces.

2. 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.

05.235 The direction of applying the control force should correspond to the direction normally used by the pilot.

05.240 GENERAL

In so far as the requirements of CAR 05.24 are concerned, landing gears will be considered conventional if they consist of:

a. A single wheel or double co-axial wheels located on the bottom of the fuselage and directly below (or nearly so) the center of gravity of the glider, together with auxiliary skids attached to the bottom of the fuselage. One auxiliary skid running from the wheel forward to the nose, the other running aft to a point below the wing trailing edge (approximately). The rear auxiliary skid may be replaced or supplemented by a suitable tail skid (See Figures .0-2 and .2-14), or

b. A single main skid on the bottom of the fuselage which extends from the nose to a point below the wing trailing edge (approximately). This skid may be supplemented by a tail skid (see Fig. .2-14).

NOTE: Wing tip skids may be employed if desired.

05.241 LEVEL LANDING

The loading for this condition is illustrated in Fig. .2-12. As specified in CAR 05.241 the glider shall be assumed to be in a level attitude. However, as it is difficult to accurately define "level attitude", any <u>reasonable</u> attitude with the tail well off the ground will be considered satisfactory. If in the level landing condition the resultant load does not pass through the center of gravity, it will generally be acceptable to apply a balancing couple composed of an upward force acting near the nose of the fuselage and an equal downward 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 may be divided between the nearest panel points, if desired. These forces are illustrated in Fig. .2-12(a).

05.242 LEVEL LANDING WITH SIDE LOAD

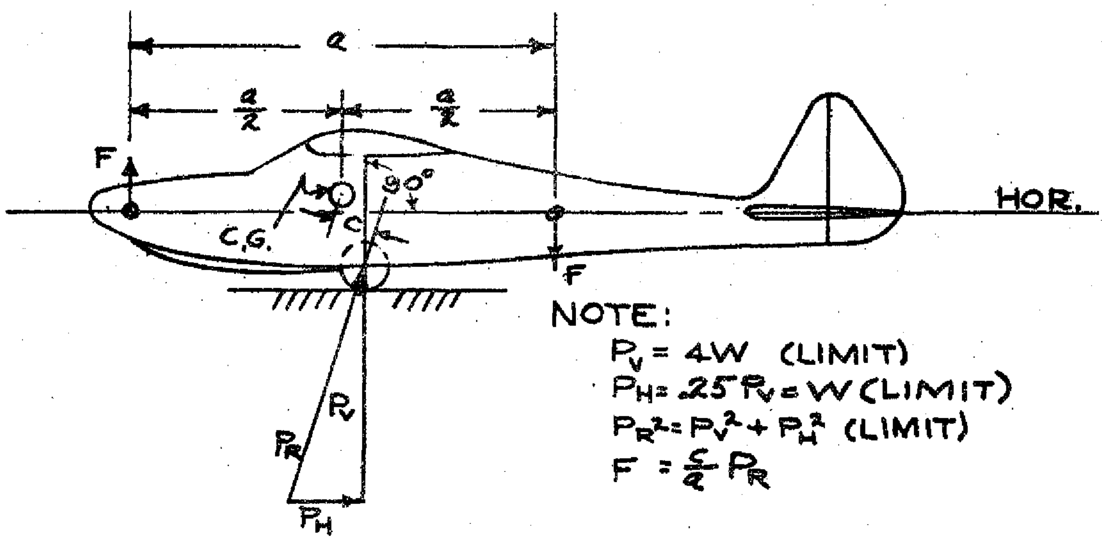
The loading for this condition is illustrated in Fig. .2-13. An acceptable method of balancing externally applied rolling moments about the longitudinal axis resulting from the side load is illustrated in Fig. .2-13(a). The forces resisting angular acceleration are assumed to be applied by the wing. 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 glider.

05.243 NOSE-DOWN LANDING

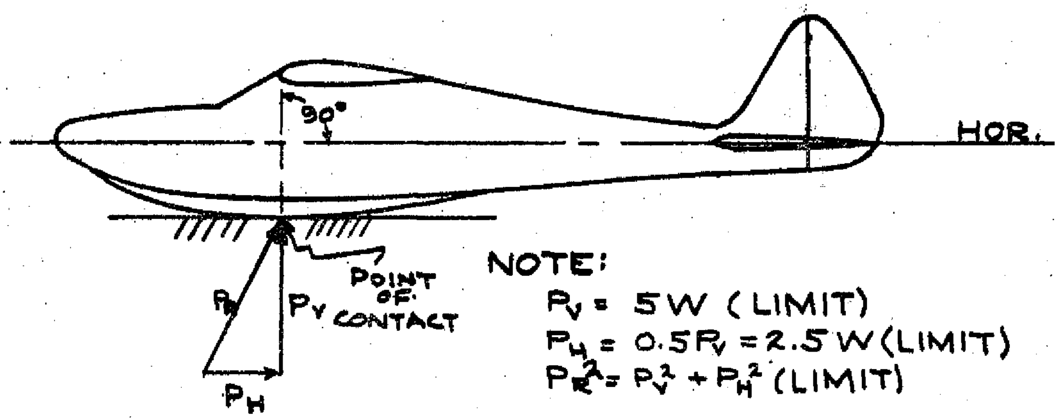
This condition is illustrated in Fig. .2-14.

05.244 HEAD-ON LANDING

It should be noted that the load factor specified in CAR 05.243 (4.0) is an <u>ultimate</u> load factor and not a <u>limit</u> load factor.

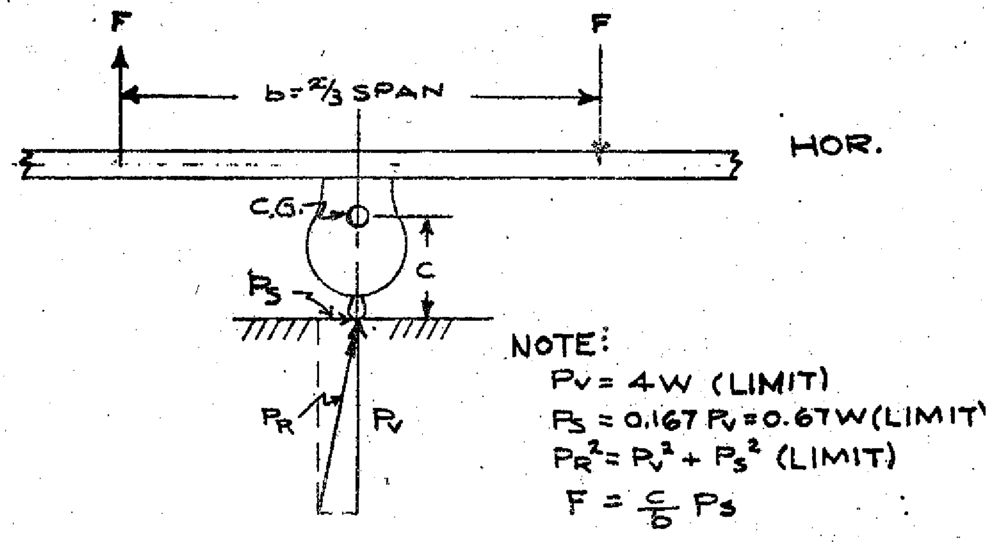


(a) For Wheel Type Landing Gears

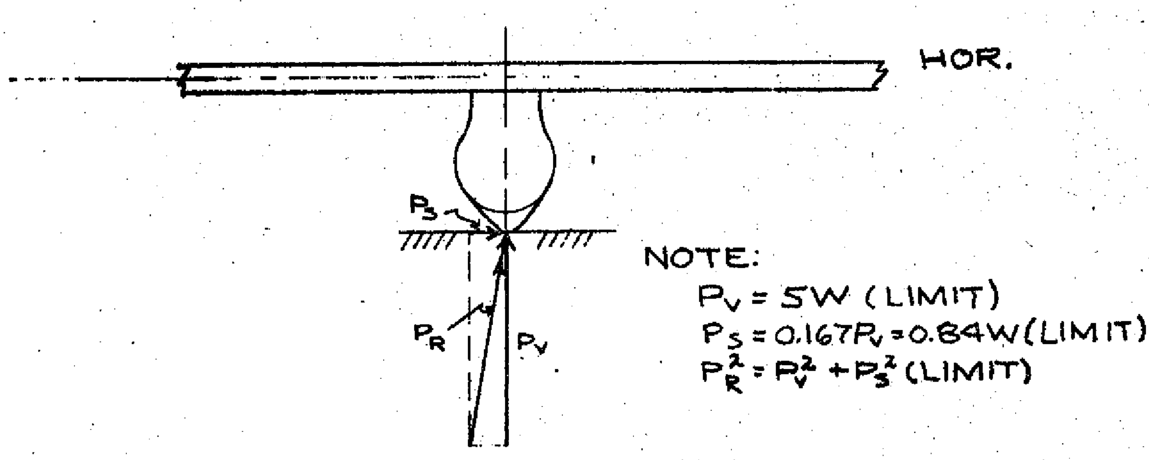


(b) For Skid Type Landing Gears

FIG. .2-12 - LEVEL LANDING
(Ref. CAAM 05.241)



(a) For Wheel Type Landing Gears



(b) For Skid Type Landing Gears

(Note: See Fig. .2-12 for Horizontal Components)

FIG. .2-13 - LEVEL LANDING WITH SIDE LOAD
(Ref. CAAM 05.242)

05.245 WING TIP LANDING

It may be assumed that a limit load of 150 pounds acts aft at the point of contact of one wing tip (or wing skid if one is used) and the ground in a direction parallel to the longitudinal axis. The unbalanced turning moment may be assumed to be resisted by:

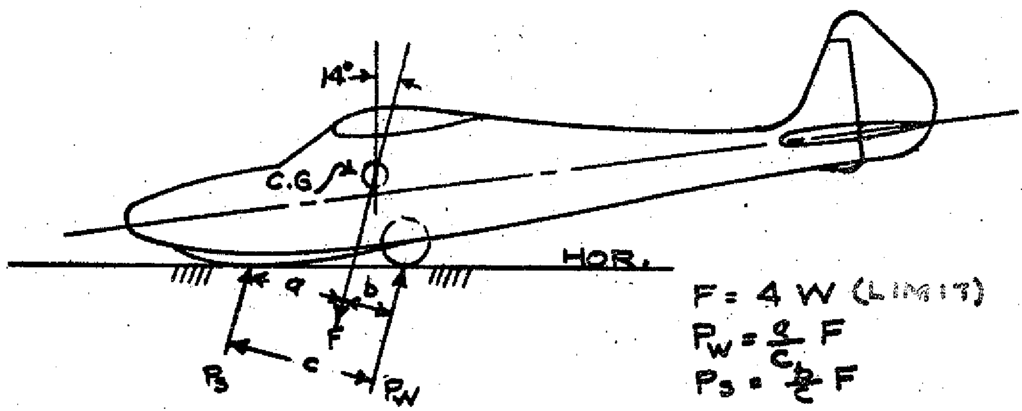
a. The methods shown in Fig. .2-15; or

b. The angular inertia of the glider.

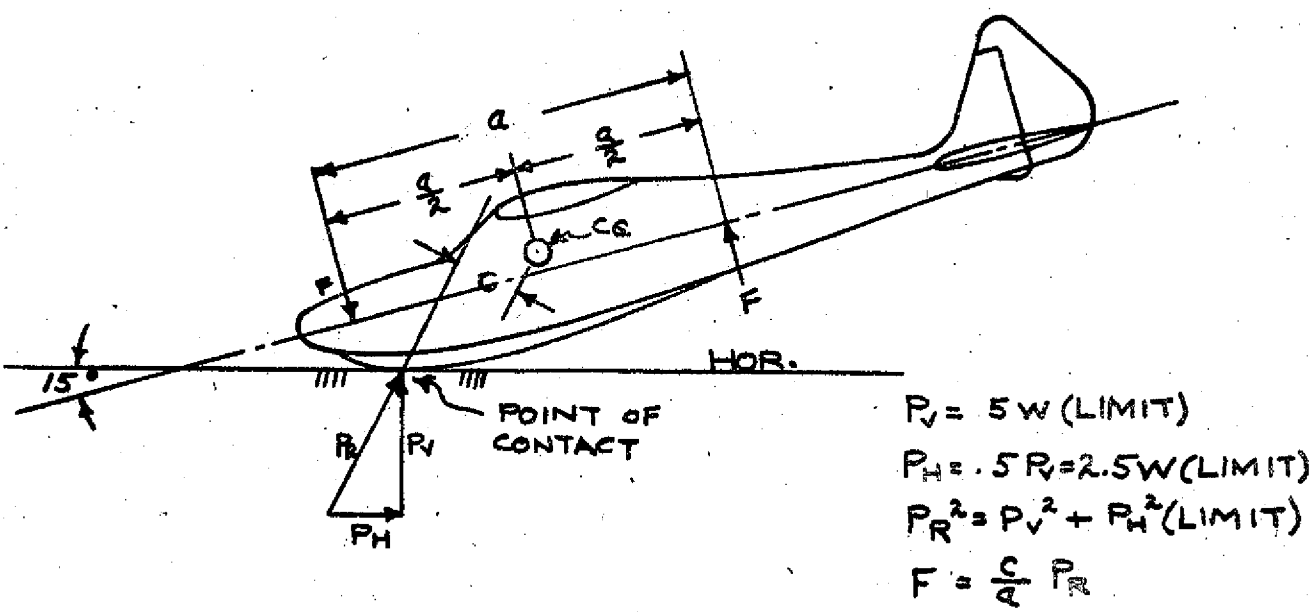
05.25 LAUNCHING AND TOWING LOADS

05.251 The loads specified in CAR 05.251 (a), (b), (c), and (d) are illustrated in Fig. .2-16. The effects of these loads need not be investigated aft of the front wing spar.

05.252 It can be assumed that a limit rearward acting chord load factor of 3.0 is developed in shock chord and winch launches.

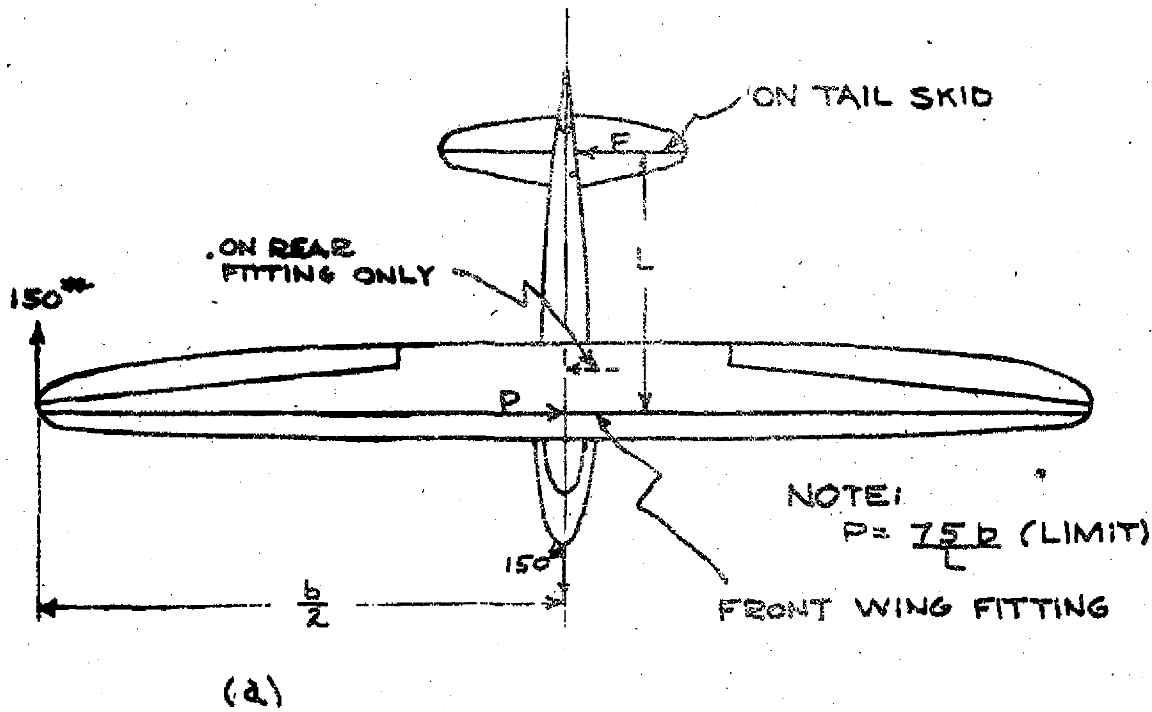


(a) For Wheel Type Landing Gears



(b) For Skid Type Landing Gears

FIG. .2-14 NOSE-DOWN LANDING
(Ref. CAAM 05.243)



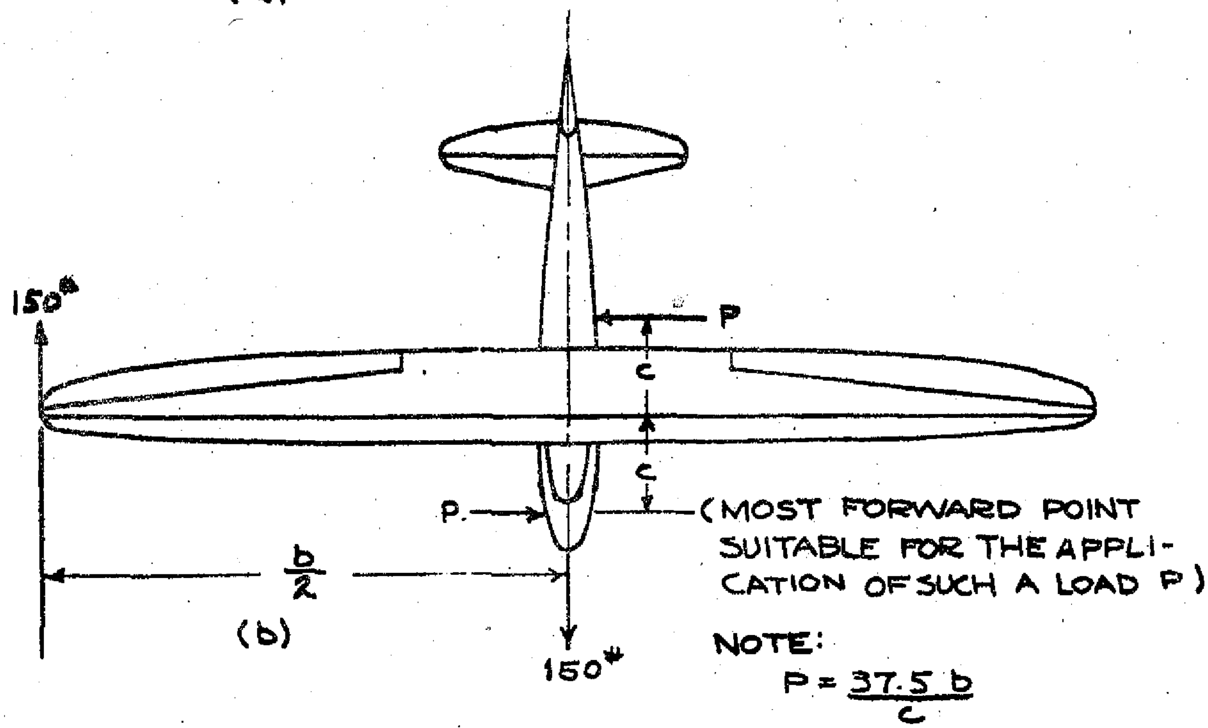
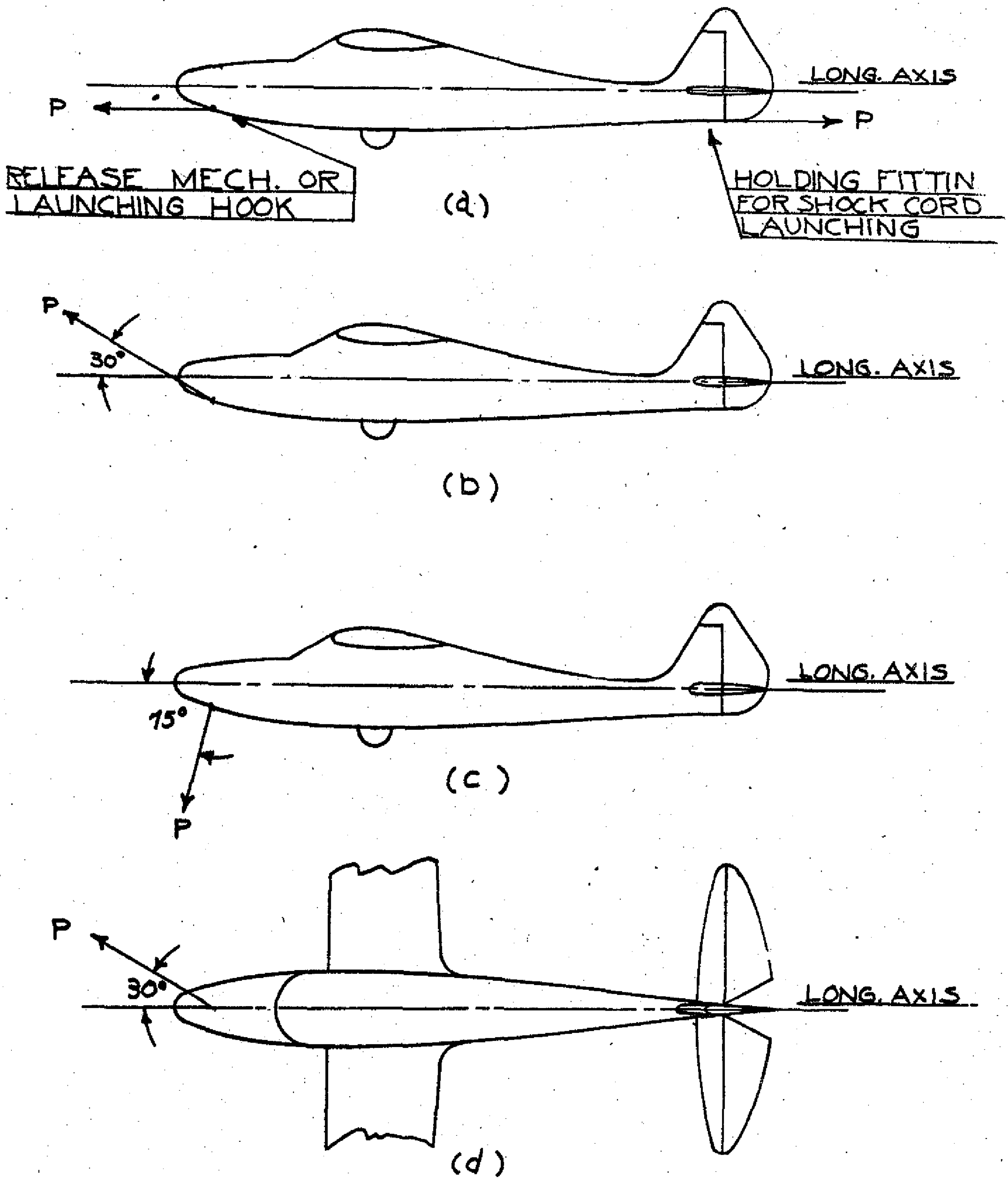


FIG. .2-15 - WING TIP LANDING
(Ref. CAAM 06.245)



Note: P = 1200 pounds or 3.0 times the gross weight, whichever is greater (limit).

FIG. .2-16 - LAUNCHING AND TOWING LOADS
(Ref. CAAM 05.251)

05.31 DETERMINATION OF LOADINGS

05.310 WINGS

05.3100 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:

$$y_f = \left[\left\{ C_N\ (r-a) + C_{M_a} \right\} q + n_2\ e\ (r-j) \right] \frac{C'}{144\ b} \qquad (.3\text{-}1)$$

$$y_r = \left[\left\{ C_N\ (a-f) - C_{M_a} \right\} q + n_2\ e\ (j-f) \right] \frac{C'}{144\ b} \qquad (.3\text{-}2)$$

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. .3-1 and are all expressed as fractions of the chord at the station in question.

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

q = dynamic pressure for the condition being investigated.
C_N and C_{M_a} are the airfoil coefficients at the section in question.
C' is the wing chord, in inches.
e is the average unit weight of the wing, in pounds per square foot, over the chord at the station in question. It should be computed or estimated for each area included between the wing stations investigated, unless the unit wing weight is substantially constant, in which case a constant value may be assumed.
n_2 is the net limit load factor representing the inertia effect of the whole glider 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 CAAM 05.218.

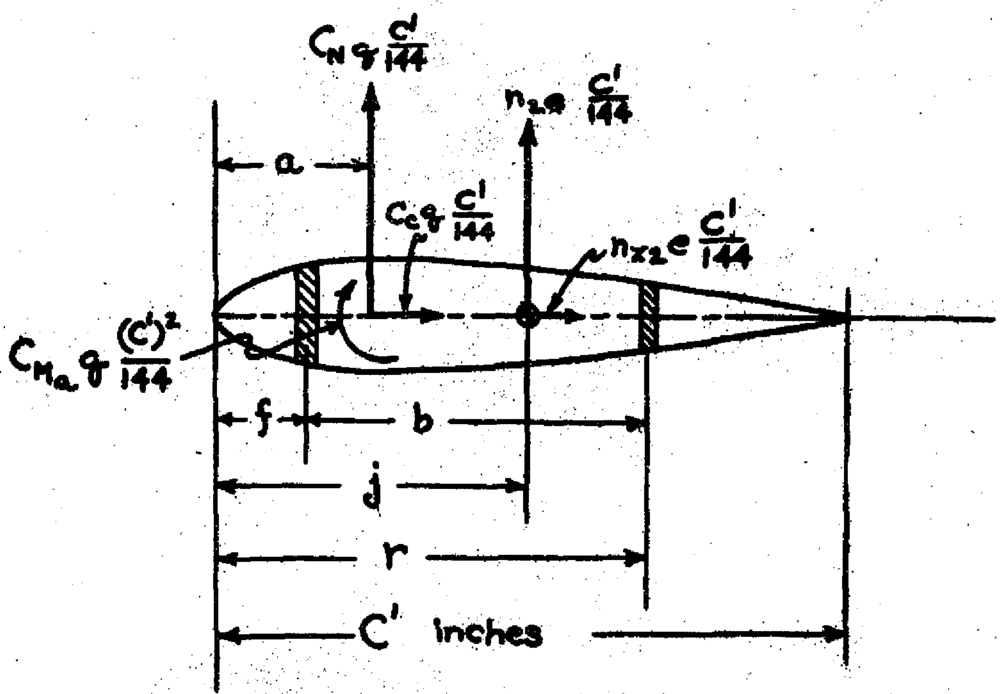
TABLE .3-I

COMPUTATION OF NET UNIT LOADINGS (CONSTANTS)

		Stations Along Span				
1	Distance from root, inches					
2	C'/144 = (chord in inches) /144					
3	f, fraction of chord					
4	r, fraction of chord					
5	b = r - f =(4) - (3)					
6	a, fraction of chord (a.c.)					
7	j, fraction of chord					
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)					

3. The computations required in using the above method are outlined in Tables .3-I and .3-II, 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.



ALL VECTORS ARE SHOWN IN POSITIVE SENSE

FIG. .3-1

UNIT SECTION OF A CONVENTIONAL 2-SPAR WING

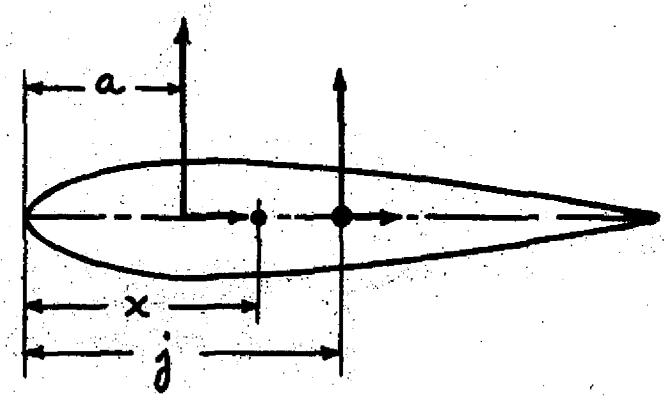


FIG. .3-2

SECTION SHOWING LOCATION OF ELASTIC AXIS

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. .2-8(c) is used for this purpose, together with the value of K_b obtained for this figure, as outlined in CAAM 05.217-C-3.

c. Item 15 provides for a variation in the local value of C_M.

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).

05.3101 DETERMINATION OF RUNNING CHORD LOAD

1. The net chord loading, in pounds per inch run, can be determined from the following equation:

$$y_c = \left[C_c\, q + n_{x2}\, e \right] C'/144 \qquad (.3\text{-}3)$$

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 glider in the chord direction. The value and sign are obtained in the balancing process outlined in CAAM 05.218. Note that when C_c is negative, n_{x2} will be positive.

e and C' are the same as in CAAM 05.3100.

2. The computations for obtaining the chord load are outlined in Table .3-II, Items 28 to 32. The following points should be noted:

a. The value of C_c, item 28, can usually be assumed to

TABLE .3-11

COMPUTATION OF NET UNIT LOADINGS (VARIABLES)

CONDITION

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

		(Refer also to Table .3-1)	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						

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 CAAM 05.123-A2 and Fig. .1-4.

05.3102 DETERMINATION OF RUNNING LOAD AND TORSION AT A GIVEN 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. .3-2, x denotes the location of the reference axis, expressed as a fraction of the chord. The net running load along the locus of the points x and the net running torsion about these points are found from the following equations:

$$y_x = (C_N\, q + n_2\, e)\ \frac{C'}{144} \qquad (.3\text{-}4)$$

$$m_x = \left[\left\{C_N\,(x-a) + C_{M_a}\right\} q + n_2\, e\,(x-j)\right] \frac{(C')^2}{144} \qquad (.3\text{-}5)$$

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 CAAM 05.3100. (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 .3-I and .3-II. 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 CAAM 05.3101.

05.311 CONTROL SYSTEMS

05.3110 FLIGHT CONTROLS

The minimum loads to be used for the design of flight control systems are covered in CAR 05.230-233 and CAAM 05.230-233.

05.3111 SECONDARY CONTROLS

The minimum loads to be used for the design of secondary control systems are covered in CAR 05.234 & 05.235 and CAAM 05.234 & 05.235.

05.312 FUSELAGES

05.3120 WEIGHT DISTRIBUTION

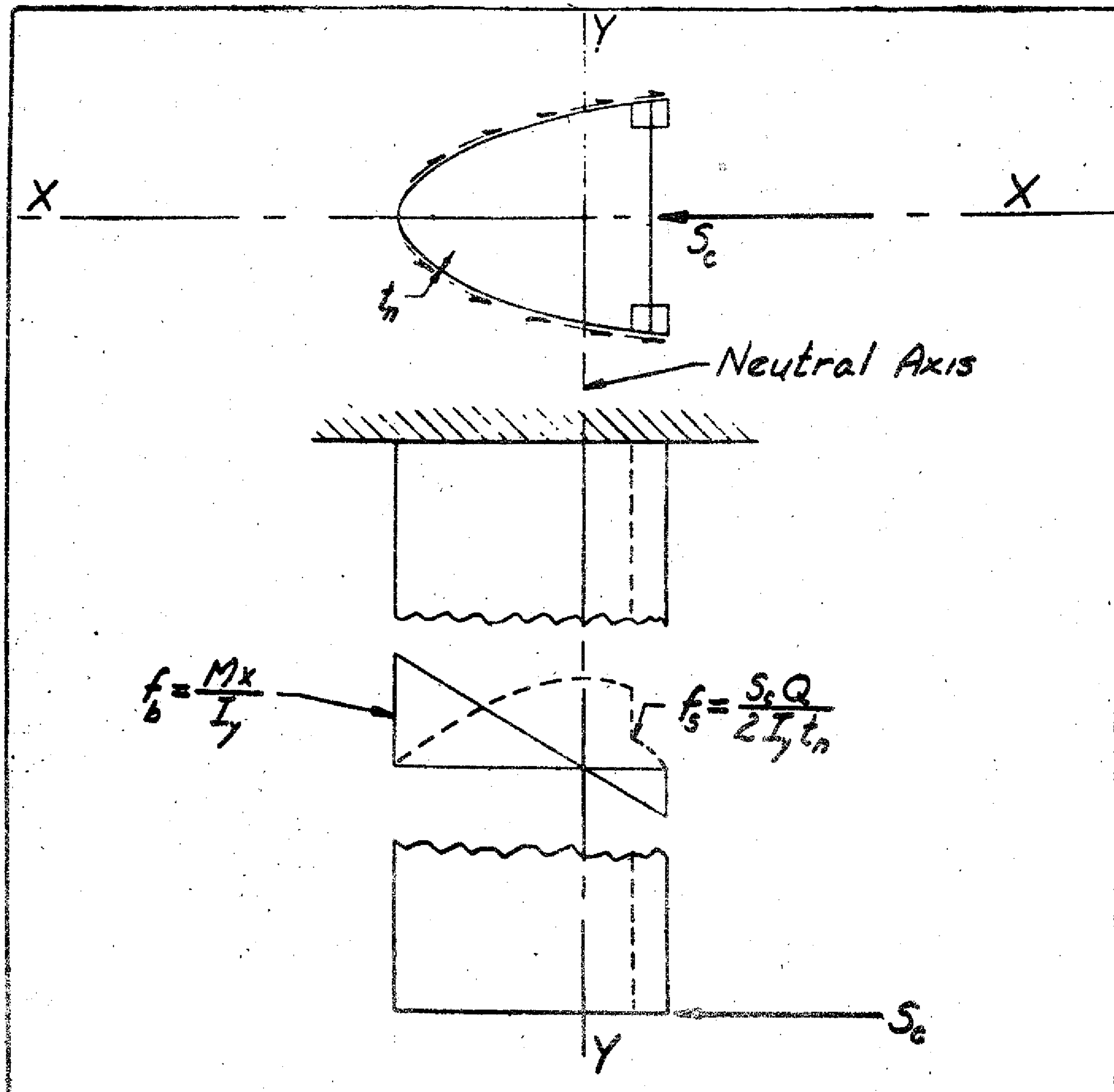
All major items of weight affecting the fuselage should be so distributed to convenient panel points that the true center of gravity of the fuselage and its contents is maintained. A suitable vertical division of loads should be included. The following rules should be followed in computing the panel point loads for conventional gliders:

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 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.

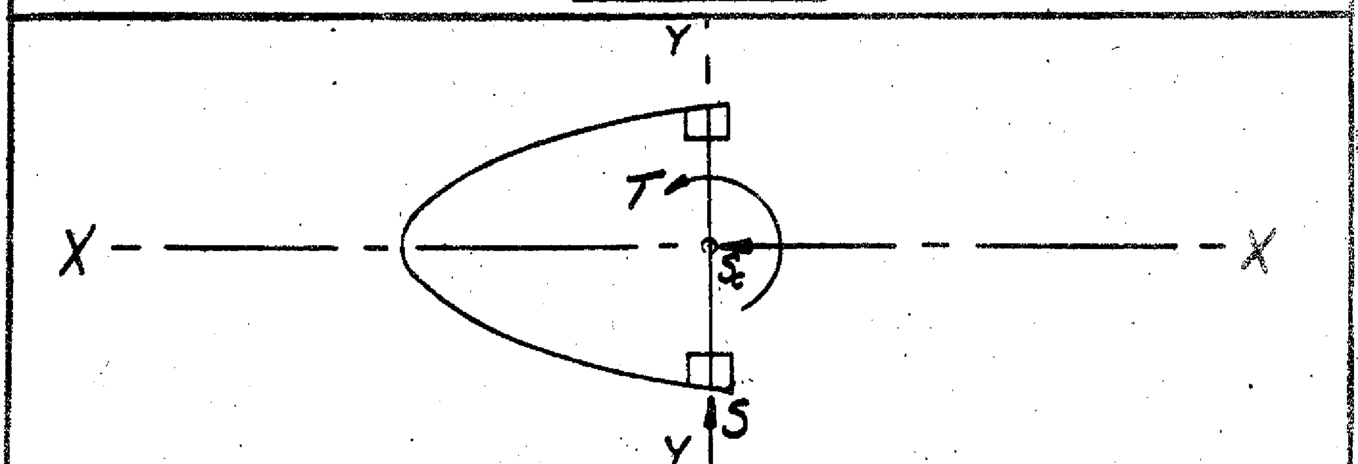
c. In all cases the moment of the partial panel loads due to any item about an origin near the nose of the fuselage should be equal to the moment of the item about that origin.

d. 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.



Y
X
X
S_c
t_n
Neutral Axis
$f_b = \frac{Mx}{I_y}$
$f_s = \frac{S_c Q}{2 I_y t_n}$
S_c
Y

FIG. 3-6



Y
T
X
X
S_c
S
Y

FIG. 3-7

05.32 STRUCTURAL ANALYSES

05.320 GENERAL

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 twenty-five cents.

05.321 PROOF OF WINGS

05.3210 GENERAL

Proof of wings by structural analysis only should be in accordance with CAR 05.32. The following points are pertinent:

1. Joint slippage in wood beams. When a joint in a wood beam is designed to transmit bending from one section of the beam to another or to the fuselage, the stresses in each part of the structure should be calculated on the assumption that the joint is 100 per cent efficient and also under the assumption that the bending moment transmitted by the joint is 75 per cent of that obtained under the assumption of perfect continuity. Each part of the structure should be designed to carry the most severe loads determined from the above assumptions.

2. Bolt holes. In computing the area, moment of inertia, etc., of wood beams pierced by bolts, the diameter of the bolt hole should be assumed to be one-sixteenth inch greater than the diameter of the bolt.

3. In computing the ability of box beams to resist bending loads only that portion of the web with its grain parallel to the beam axis and one-half of that portion of the web with its grain at an angle of 45° to the beam should be considered. The more conservative method of neglecting the web entirely may be employed.

4. Drag trusses. Drag struts should be assumed to have an end fixity coefficient of 1.0 except in cases of unusually rigid restraint, in which a coefficient of 1.5 may be used.

05.3211 DETERMINATION OF LOADING CURVES

1. The loading curves can be determined by the general methods given in CAAM 05.310, but for simplicity, a direct comparison of the Beam, Chord, and Torsion coefficients times the respective factors will give an indication of which conditions are going to be critical, especially for a single spar design. Thus, it may be possible to eliminate one or more conditions before plotting the actual curves.

2. The distribution of the wing dead weight can be estimated and shear and moment curves plotted. The dead weight curves times the dead weight load factor n_2 in the beam and chord directions are subtracted from the total respective limit beam and chord load curves to obtain the "net" limit load curves.

3. It will simplify the curves somewhat to figure and construct all net wing shear and moment curves to the root of the wing, even though it may be externally braced. The loads in the strut and its effect on the wing curves can be superimposed on the latter to determine the final curves.

4. In constructing shear curves from a curved running load, it is necessary to integrate the loading curve to obtain the shear, or to approximate the curve with a series of straight lines so that it can be calculated as a series of trapezoids. Moment curves are developed similarly from the shear curves.

05.3212 TWO SPAR WINGS

Methods of analysis for conventional two spar wings are contained in standard textbooks on airplane structures.

A WOOD SPARS

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

B METAL SPARS

1. 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 aircraft. If this is not practicable, the shear on the test beam should be relatively larger than in the aircraft. 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.

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.

3. 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.

05.3213 D-NOSE SPARS

A TWO FLANGE TYPE

1. Basic Principles and Assumptions.

Insofar as the structural analysis is concerned, the D-nose spar with two flanges may be considered as the combination of a beam (flanges and vertical web) and a torsion box (leading edge skin and vertical web). A typical structure of this type is shown in Figure .3-3. In the analysis it is assumed that the spar does not taper and that the structure is symmetrical, insofar as the two concentrated flanges are concerned, about a horizontal axis (a principal axis). It is further assumed that the structure is of such a nature that the engineering theories of bending (flexure formula), shear and torsion (membrance formula) are applicable.

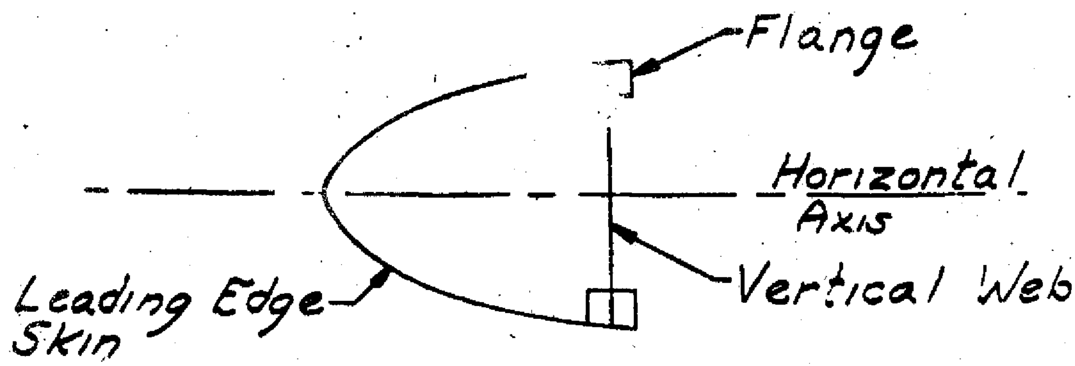


FIG .3-3: TYPICAL D-NOSE SPAR WITH TWO FLANGES

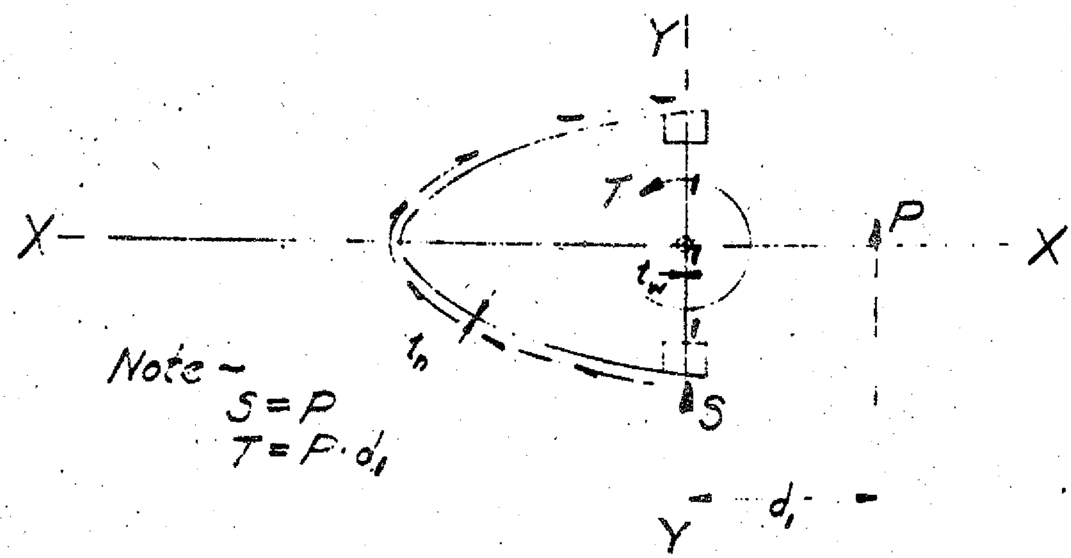


FIG .3-4

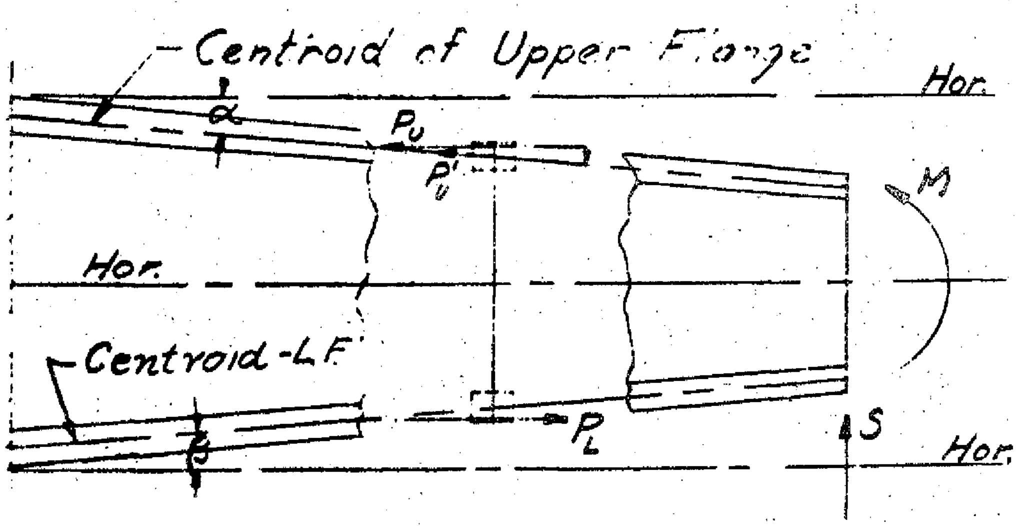


FIG .3-5

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2. Detailed Analysis for Beam Loads.

For the case of beam loading only, the spar will be subjected to a force P (see Fig. .3-4). For the purpose of practical structural analysis, it is desirable to replace the force P by a force S acting in the plane of the web and a torsional moment (or couple) acting about a point in the web. This conversion is illustrated in Figure .3-4.

a. Stress in Leading Edge Skin.

In this case, where the bending material is concentrated in two flanges (top and bottom of web), the leading edge skin will be stressed primarily by the torsional moment M. (Actually the leading edge skin adjacent to the flanges will be stressed by the bending loads.) The stress in the skin can be obtained from one of the formulas given below:

(1) If the structure and magnitude of the torsional moment are such that the skin does not wrinkle under load, the skin will be stressed in shear, and the following formula will apply:

$$f_{s_n} = \frac{T}{2At_n} \quad \text{(Membrane Formula)} \qquad (.3\text{-}6)$$

Where: T = the torsional moment in inch pounds.
A = the area enclosed by the leading edge skin and the web in square inches.
t_n = the thickness of the leading edge skin in inches.

(2) If the skin wrinkles under load, it will be stressed in tension (tension fields), and the following formula will apply:

$$f_{t_n} = \frac{T}{At_n} \qquad (.3\text{-}7)$$

For obvious reasons the skin should not wrinkle appreciably when stressed below the limit load.

b. Stress in Concentrated Flanges.

If the structure is not subjected to axial loads and does not taper in depth, the flanges will be stressed primarily by the bending load. The flange stress can be determined from the following formula:

$$f_b = \frac{My}{I_x} \qquad \text{(Flexure Formula)} \qquad (.3\text{-}8)$$

where M = the bending moment in inch pounds.
y = the distance from the neutral axis to the flange fiber being investigated in inches. (Obviously the outermost (or extreme) fiber will be most highly stressed).
I_x = the moment of inertia of the flanges about their neutral axis (the X-X axis) in inches to the fourth power.

If the flange material is concentrated, or nearly so, the above formula closely approaches:

$$f_b = \frac{M}{hA_f} \qquad (.3\text{-}9)$$

where h = the distance between the centroids of the two flanges (upper and lower) in inches.
A_f = the cross sectional area of the flange in question (upper or lower) in square inches.

c. Stress in Web.

The web will be stressed primarily by the shear S and the torsional moment T. (In certain cases it may be desirable to investigate the effect of bending at the junction of the web and the flanges.) The stress in the web can be determined from one of the formulas given below.

(1) If the structure and the magnitude of the loadings are such that the web does not wrinkle under load, the web will be stressed in shear and the following formula will apply:

$$f_{s_w} = \frac{SQ}{I_x t_w} + \frac{T}{2At_w} \qquad (.3\text{-}10)$$

where S = the shear in pounds.
Q = the static moment about the neutral axis of the section effective in bending above or below the point in question

in inches cubed. (For the remaining symbols, see a and b above.)

If the flange material is concentrated, the above formula becomes:

$$f_{s_w} = \frac{S}{ht_w} + \frac{T}{2At_w} \qquad (.3\text{-}11)$$

where h = the depth of the web in inches.

(2) If the web wrinkles under load, it will be stressed in tension (tension fields), and the following formula will apply:

$$f_{t_w} = \frac{2S}{ht_w} + \frac{T}{At_w} \qquad (.3\text{-}12)$$

d. Effect of Taper.

The effect of taper in spar depth is twofold, namely: (1) a decrease in the amount of beam shear resisted by the vertical web, and (2) a relatively slight increase in the axial loads in the flanges of the beam.

(1) Referring to Figure .3-5, it can be seen that the shear S' carried by the web of a tapered beam is expressed by the following formula:

$$S' = S - (P_U \tan a \quad P_L \tan \beta) \qquad (.3\text{-}13)$$

where S = the external shear at the section in pounds.

P_U = the horizontal component of the axial load in the upper flange due to bending in pounds. (It may be obtained by multiplying the stress determined from Formulas .3-8 or .3-9 by the cross sectional area of the upper flange.)

P_L = the horizontal component of the axial load in the lower flange.

(For a and 8, see Figure .3-5.)

(2) Although the effect of taper on the axial load in the beam flanges is usually so small that it can be safely neglected; it should be considered in the analysis of the critical sections of highly tapered beams. The true axial loads in the flanges, P_U' and P_L', can be closely approximated from the following formulas:

$$P_U' = P_U \sec \alpha \qquad (.3\text{-}14)$$

and

$$P_L' = P_L \sec \beta \qquad (.3\text{-}15)$$

For the meaning of the symbols, see Formula (.3-13) and Figure .3-5.

e. Effect of Axial Load.

If the spar is subjected to an axial load P in addition to the shear and bending moment, the flange stresses can not be determined from the formulas given above. The flange stresses can, however, be obtained from the following formulas:

$$f = \frac{M'y}{I_x} + \frac{P}{A} \qquad (.3\text{-}16)$$

where M' = the precise bending moment in inch pounds. (The precise moment includes the effect of secondary bending due to the axial load).
P = the axial load in pounds.
A = the cross sectional area of the flanges (upper and lower) in square inches.

(For the meaning of the other symbols, see Formula (.3-8)).

3. Detailed Analysis for Chord Loads.

A chord loading will subject the D-nose spar to both bending and shear stresses. The resulting stress distribution is shown in Figure .3-6.

a. Bending Stress.

The stress due to bending can be determined from the following formula:

$$f_b = \frac{Mx}{I_y} \qquad (.3\text{-}17)$$

where M = the bending moment about the Y-Y axis in inch pounds.

x = the distance from the neutral axis to the fiber being investigated in inches. (In general, the most forward portion of the leading edge skin (an external fiber) will be the most highly stressed).

I_y = the moment of inertia about the neutral axis (the Y-Y axis) in inches to the fourth power.

b. Shear Stress

The stress due to shear can be determined from the following formula:

$$f_s = \frac{SQ}{2I_y t_n} \qquad (.3\text{-}18)$$

where S = the shear in pounds.

Q = the static moment about the neutral axis of the section above or below the point being investigated in inches cubed.

t_n = the thickness of the leading edge skin in inches.

4. Combined Loadings.

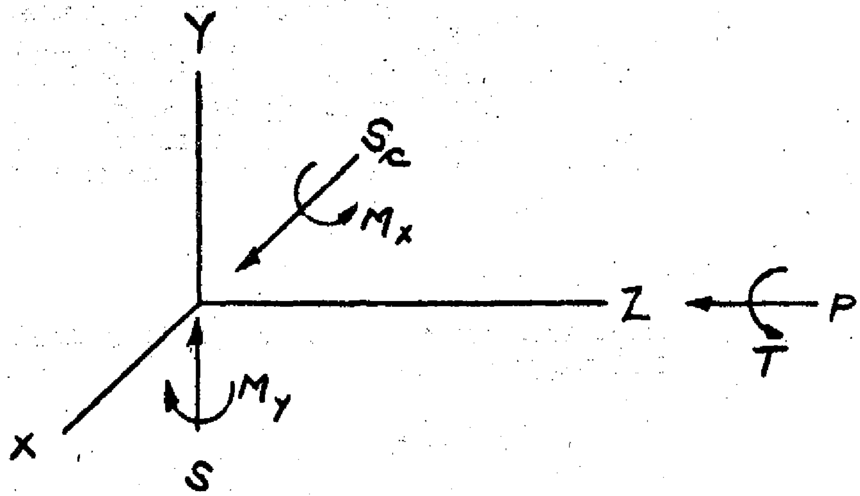
In the general case, a D-nose spar may be subjected to the loadings shown in Figure .3-7 together with an axial load P. The effects of these loadings and the pertinent formulas for determining the corresponding stresses can be determined from Table .3-III.

5. Limitations of Formulas.

The accuracy of the foregoing formulas will not be high in the vicinity of the root of the spar because the actual stress distribution will differ considerably from the assumed distribution. It is, therefore, recommended that high computed margins of safety be maintained in the root region. The theory also breaks down to some extent if stress concentrations are present. Therefore, the design should be of such a nature that serious stress concentrations will not exist.

TABLE .3-III

LOADINGS, TYPE OF STRESS AND PERTINENT FORMULA FOR D-NOSE SPARS



Component	Loading	Type of Stress	Formula
L. E. Skin	Beam (S, M_x & T)	Shear or Tension	(.3-6) or (.3-7)
" " "	Beam + Axial Load (S, M_x, T & P)	" "	" "
" " "	Chord (S_c & M_y)	Shear & Bending	(.3-17) and (.3-18)
Flanges	Beam (S, M_x & T)	Axial (due to bending)	(.3-8) or (.3-9)
"	Beam & Axial Load (S, M_x, T & P)	" (due to bending) and axial load	(.3-16)
"	Chord (S_c & M_y)	Axial (due to bending)	(.3-17)
Web	Beam (S, M_x & T)	Shear or Tension	(.3-10), (.3-11) or (.3-12)
"	Beam + Axial Load (S, M_x, T & P)	" "	(.3-10), (.3-11) or (.3-12)
"	Chord (S_c & M_y)	Axial (due to bending)	(.3-17)

B DETERMINATION OF ALLOWABLE STRESSES AND MARGINS OF SAFETY

1. Allowable Stresses.

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

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

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

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

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

2. Margins of Safety.

a. Leading Edge Skin.

The M.S. of the L.E. skin will be positive if

$$\left(\frac{f_b}{F_b}\right)^x + \left(\frac{f_s}{F_s}\right)^y \leq 1.0$$

compliance with yield requirements. Inasmuch as many control surfaces do not lend themselves to rigorous analysis, it is recommended that strength tests be considered for proving compliance with the ultimate load requirements.

The analysis and tests should include the horns, and should demonstrate compliance with multiplying factors of safety requirements contained in CAR 05.27.

05.3221 DEFLECTION OF HINGE POINTS

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.

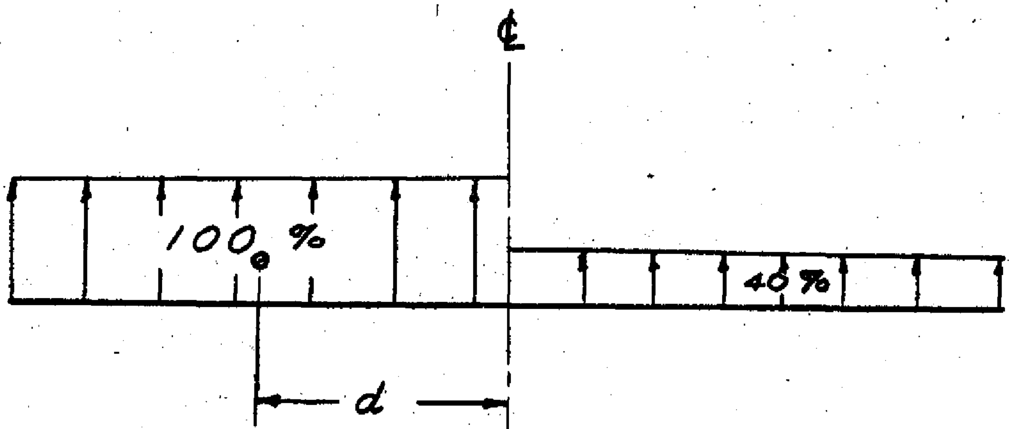
05.3222 RIGGING LOADS

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.

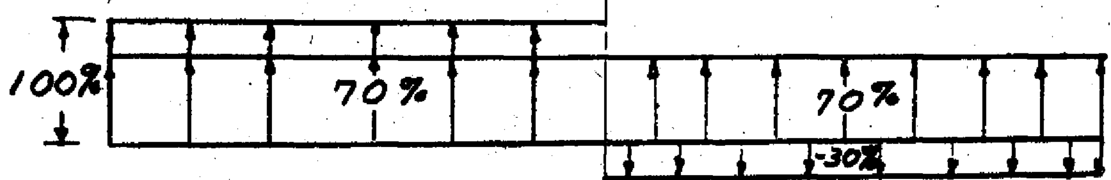
05.323 PROOF OF CONTROL SYSTEMS

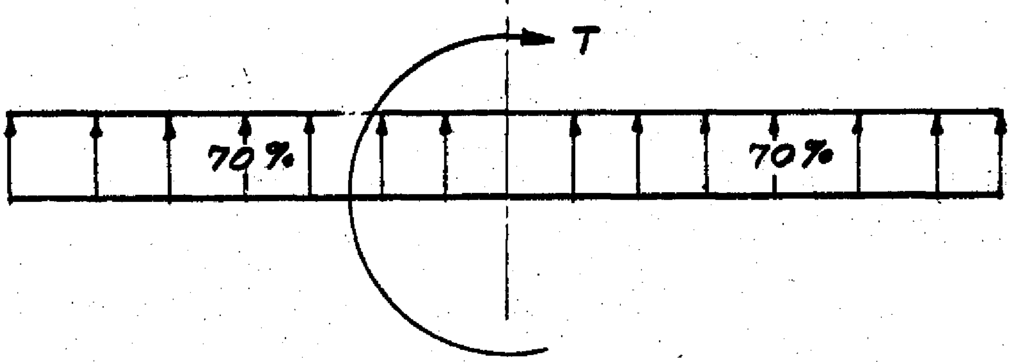
05.3230 GENERAL

Structural analyses of control systems will be accepted as complete proof of compliance with ultimate load requirements when the structure conforms with conventional types for which reliable analytical methods are available. Proof tests as defined in CAR 05.121 are required to prove compliance with yield load requirements.



Note:
d = centroid of loading.





Note:
T = .60W x d
where: W = the normal load on one panel.

FIG. .2-5

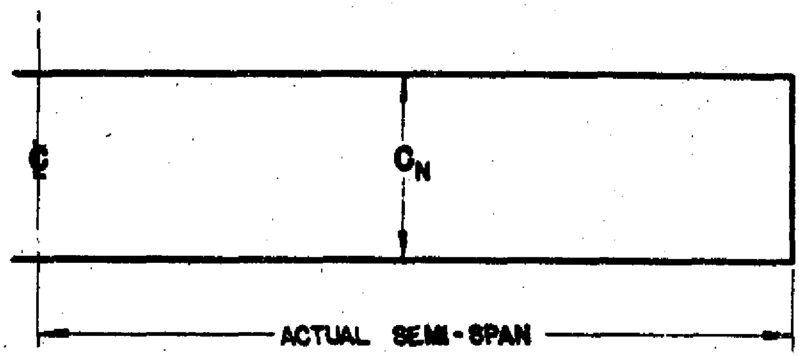


FIG. 2-6: SPAN DISTRIBUTION - NO TIP LOSS.

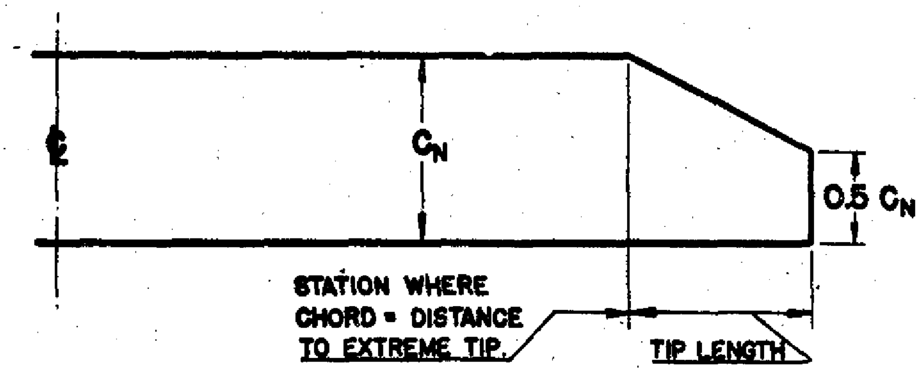


FIG. .2-7: SPAN DISTRIBUTION - WITH TIP LOSS

NOTE: Above figures apply only to wings which have mean taper ratios greater than 0.5 and without aerodynamic twist.

FIGS. .2-6 and .2-7 - SPAN DISTRIBUTION OF C_N FOR WINGS.

Army-Navy-Civil publication ANC-1 (1), "Spanwise Air Load Distribution" (obtainable from the Superintendent of Documents, Washington, D. C., at the nominal sum of 60 cents), in NACA Technical Reports Nos. 572, 585 and 631 and NACA Technical Note No. 606.

3. When the normal force coefficient is assumed to vary over the span, the values used should be so adjusted as to give the same total normal force as the design value of C_N acting uniformly over the span. (See CAAM 05.217-C for additional information).

4. When Figs. .2-6 and .2-7 are used the chord coefficient shall be assumed to be constant along the span.

5. For wings having aerodynamic twist (not geometric twist), it is very important that the effect of twist (wash-out) be considered in the investigation of the wings for the negative conditions (see CAAM 05.2134).

B CHORD DISTRIBUTION

1. The approximate method of chord loadings outlined in CAAM 05.3 for the testing of wing ribs is suitable for conventional two spar construction if the rib forms a complete truss between the leading and trailing edges. An investigation of the actual chord loading should be made in the case of stressed-skin wings if the longitudinal stiffeners are used to support direct air loads. In some cases it is necessary to determine the actual distribution, not only for total load but for each surface of the wing. If wind tunnel data are not available, the methods outlined in NACA Reports Nos. 383, 411, and 465 are suitable for this purpose. These methods consist in determining the "basic" pressure distribution curve at the "ideal" angle of attack and the "additional" pressure distribution curve for the additional angle of attack. These curves can be coordinated with certain values of C_L, so that the final pressure distribution curve can be obtained immediately for any C_L. Curves of this nature for several widely-used airfoils can be obtained directly from the NACA.

2. Leading Edge Loads. On high speed gliders the leading edge loads developed may be exceptionally severe, particularly the "down" loads which are produced by negative gusts when flying at the design gliding speed. The magnitude of such loads can be estimated, without determining the entire chord distribution, by the method outlined in NACA Report No. 413.

3. Effects of Auxiliary Devices. When a design speed higher than required is used in connection with wing flaps or other auxiliary devices, it will be necessary to determine the chord distribution over the entire airfoil. The effect of any device which remains operative up to V_g should be carefully investigated. This applies particularly to auxiliary airfoils, spoilers, and fixed slots.

C. DETERMINATION OF RESULTANT AIR FORCES

1. A general method is outlined below for determining the mean effective value of the normal force coefficient, the average moment coefficient, location of the mean aerodynamic center and value of the mean aerodynamic chord. These factors are needed in order to determine the balancing loads for various flight conditions (see CAAM 05.218). The most general case will be considered, so that certain steps can be omitted when simpler wing forms or span load distribution curves are involved.

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

3. For convenience and clarification, Table .2-I has been developed and the various curves obtained as a part of this method are illustrated in Figs. .2-8, .2-9 and .2-10. It should be particularly noted that when the area under a curve is referred to, the area should be expressed in terms of the product of the units to which the curve is drawn. The procedure is as follows:

   a. Fig. .2-8(a) illustrates the actual wing plan form, plotted to a suitable scale. This should agree with the definition of design wing area given in CAR 05.101.

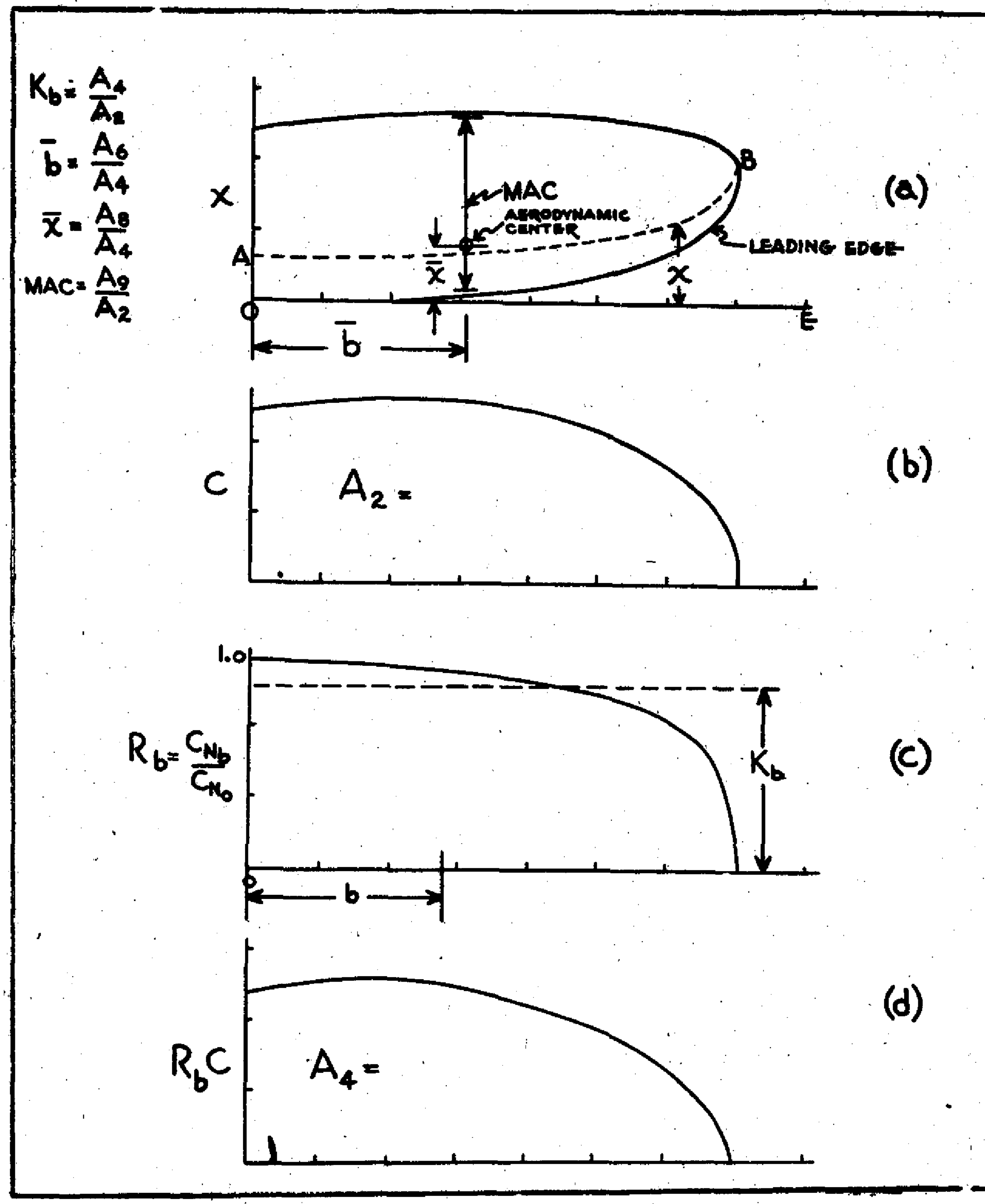


FIG. .2-8 - DETERMINATION OF EFFECTIVE NORMAL FORCE COEFFICIENT
(REF. CAAM 05.217-C3)

b. Fig. .2-8(b) shows the variation of wing chord, C, with span. The values of C are entered in Table .2-I as item (2). The area of the figure should be accurately determined and converted to the proper units. It should be one-half the value of design wing area.

c. Fig. .2-8(c) represents an assumed span distribution curve. The factor R_b represents the ratio of the actual C_N at any point to the value of C_{N_0} at the root of the wing. Values of R_b from this curve are entered in Table .2-I under item (3).

d. Fig. .2-8(d) is obtained by plotting R_bC (item 4) in Table .2-I against span. The ordinates of this curve are proportional to the actual force distribution over the span. The area under curve .2-8(d) should be accurately determined and expressed in the proper units. K_b, the ratio of the mean effective C_N to the value of C_{N_0} (at the root), is obtained by dividing the area under curve .2-8(d) by the area under .2-8(b), using the same units of measurement for each area. This value of K_b is indicated by the dotted line on curve .2-8(c).

e. To determine the location of the mean aerodynamic center along the span, Fig. .2-9(a) is drawn. The ordinates are obtained by multiplying the ordinates of curve .2-8(d) by their distance along the span, as shown in item 5, of Table .2-I. The area under curve .2-9(a) divided by the area under curve .2-8(d) gives the distance from the wing root to the chord on which the mean aerodynamic center of the wing panel is located. This distance is indicated on Fig. .2-8(a) by the dimension $\bar{b}$.

f. The locus of the aerodynamic centers of each individual wing chord is plotted on Fig. .2-8(a) as the dotted line A-B. In Table .2-I the distance "x" from the base line O-E to the line A-B is entered under item 6.

g. Fig. .2-9(b) is now plotted, using as ordinates the values of R_b Cx obtained from item 7 of Table .2-I. The area under curve .2-9(b) divided by the area under curve .2-8(d) gives the distance of the mean aerodynamic center from the base line O-E in Fig. .2-10(a). This distance is indicated as $\bar{x}$ on that figure.

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

K_b =

$\bar{b}$ =

$\bar{x}$ =

MAC =

TABLE .2-1

DETERMINATION OF RESULTANT AIR FORCES
(Ref. CAAM 05.217-C)

h. If it is assumed that the moment coefficient about the aerodynamic center of each individual chord is constant over the span, the magnitude of the mean aerodynamic chord is determined by means of Fig. .2-9(c). The ordinates for this curve are determined from item 8 of Table .2-I. The area under curve .2-9(c) divided by the area under curve .2-8(b) gives the value of the mean aerodynamic chord. By way of illustration, it is drawn on Fig. .2-8(a) so that its aerodynamic center coincides with the location of the mean aerodynamic center of the wing panel.

i. In cases in which wing flaps or other auxiliary high-lift devices are used over a portion of the span it is desirable to obtain the mean effective moment coefficient. This is the coefficient to be used for balancing purposes in connection with the mean aerodynamic chord previously determined under the assumption of a uniform moment coefficient distribution. In Table .2-1 under item 9 the local values of the moment coefficient about the aerodynamic center are entered. These are also plotted as Fig. .2-10(a) to illustrate a type of distribution which might exist.

j. Fig. .2-10(b) is plotted from the values indicated under item 10 of Table .2-I. The area under this curve divided by the area under curve .2-9(c) gives the mean effective value of the moment coefficient for the entire wing panel.

k. <u>In the case of twisted wings</u> a different span distribution exists for each angle of attack. The location of the resultant forces can, however, be determined in the above manner for any known span distribution.

05.218 BALANCING LOADS

A GENERAL

1. The basic design conditions (see CAAM 05.2134) must be converted into conditions representing the external loads applied to the glider before a complete structural analysis can be made. This process is commonly referred to as "balancing" the glider and the final condition is referred to as a condition of "equilibrium". Actually, the glider is in equilibrium only in steady unaccelerated (constant

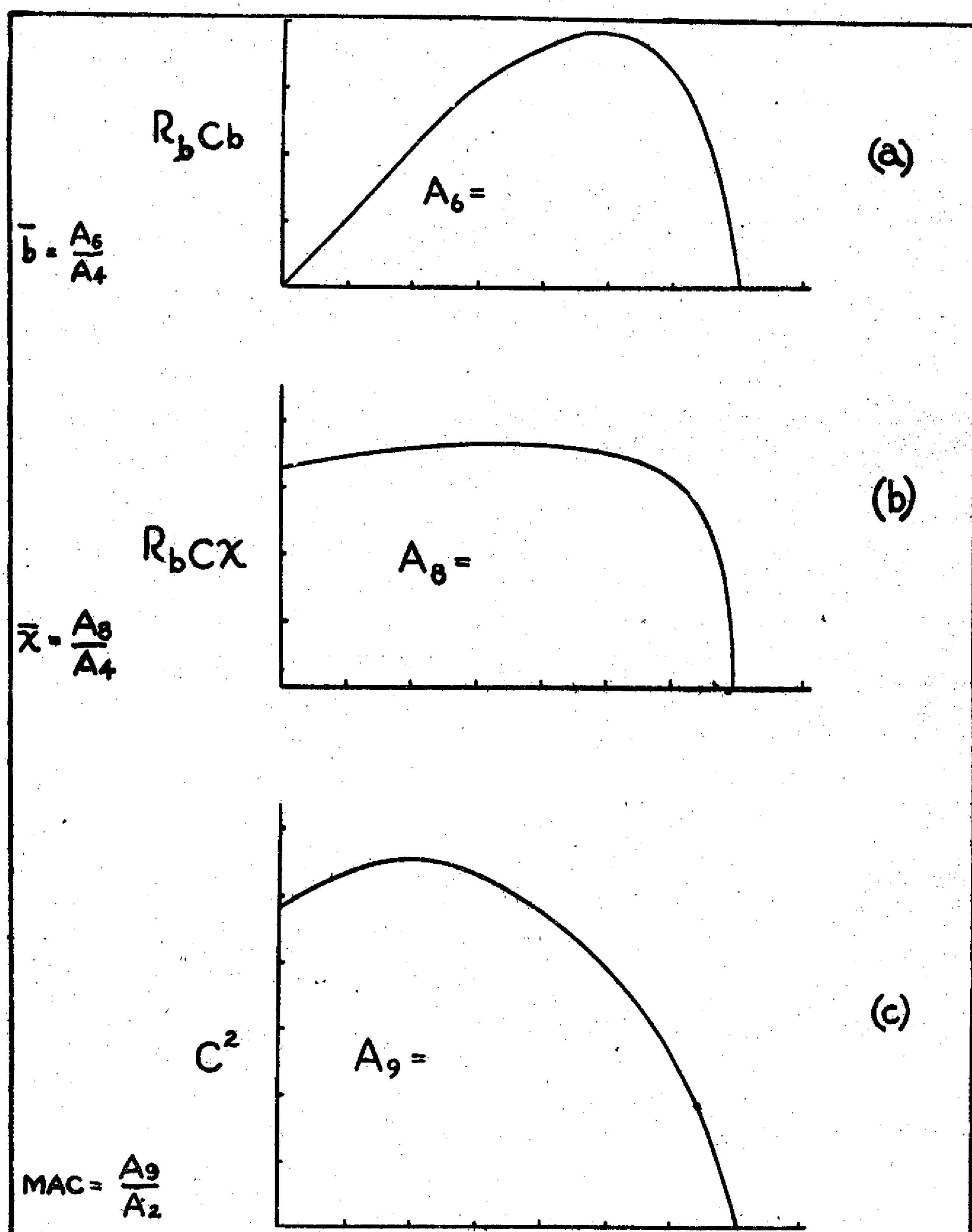


FIG. .2-9 – DETERMINATION OF MAC AND MEAN AERODYNAMIC CENTER
(Ref. CAAM 05.217-C3)

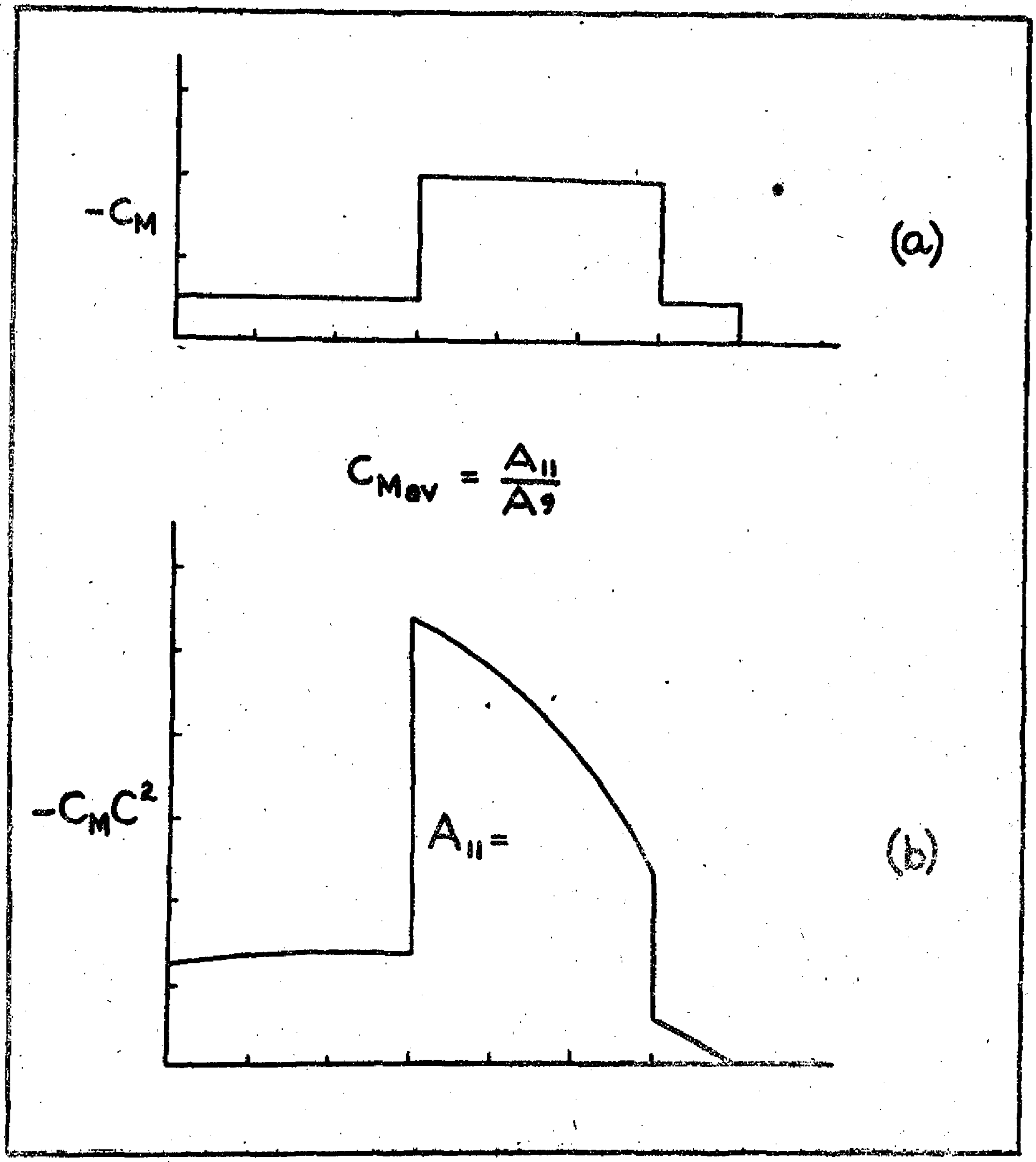


FIG. .2-1C – DETERMINATION OF MEAN EFFECTIVE MOMENT COEFFICIENT
(Ref. CAAM 05.217-C3)

speed) flight; in accelerated conditions both linear and angular accelerations act to change the velocity and attitude of the glider. It is customary to represent a dynamic condition, for structural 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 glider in a certain direction is "n" times the total weight of the glider, each item of mass in the glider is assumed to act on the glider 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 glider 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 glider. In general, such an analysis is not necessary except in certain cases for 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 glider 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 glider. 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 GLIDER

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

a. It is assumed that the limit load factors specified for the basic flight conditions (CAAM 05.2134) are <u>wing</u> load factors. A solution is therefore made for the <u>net</u> load factor acting on the whole glider. 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 glider.

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

c. In Fig. .2-11 the external forces are assumed to be acting at three points only. The assumption can generally be made that the fuselage drag acts at the center of gravity. When more accurate data are available, the resultant fuselage drag force can of course be computed and applied at the proper point. In cases where large independent items having considerable drag are present it is advisable to extend the set-up shown in Fig. .2-11 to include the additional external forces.

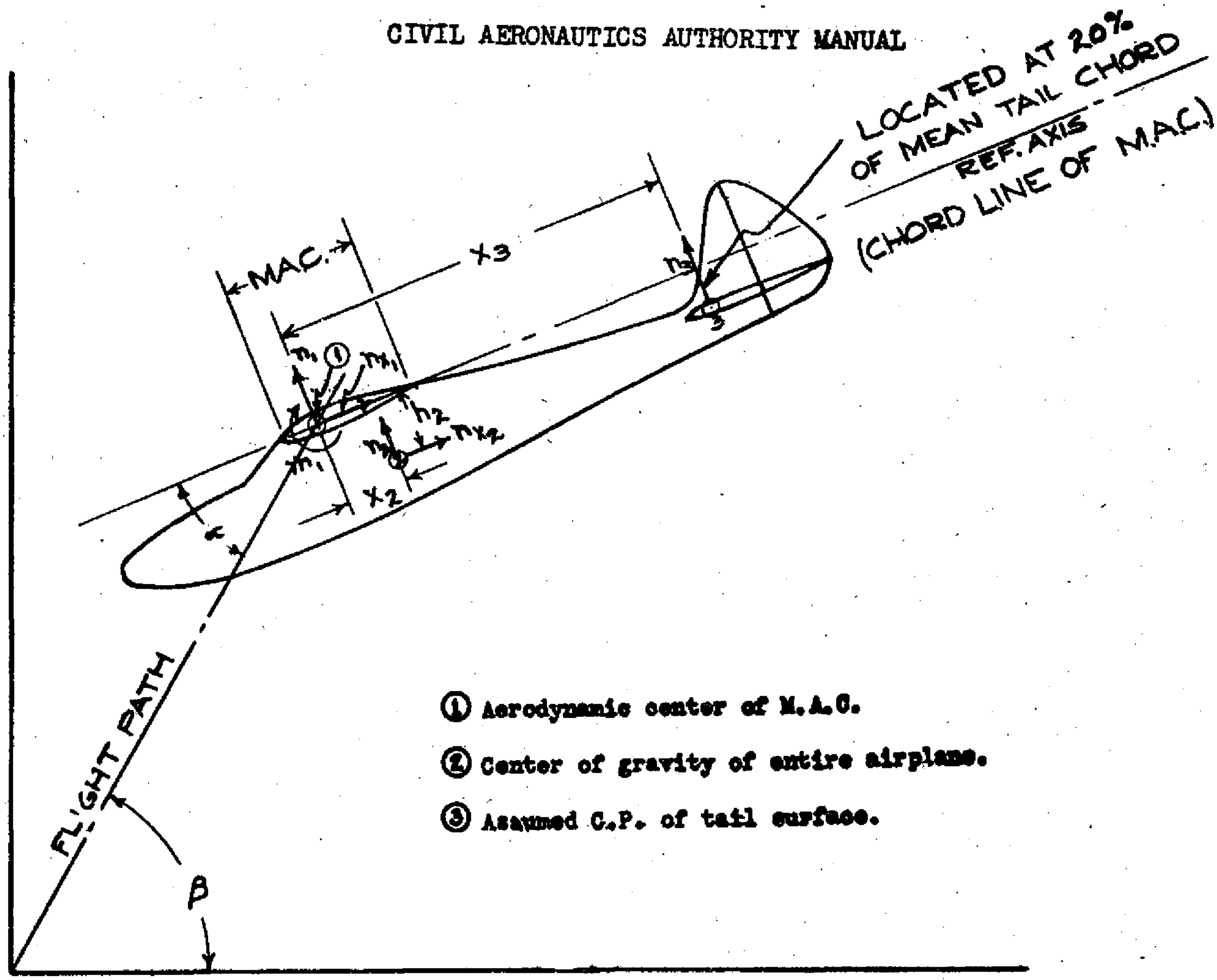
2. As shown in Fig. .2-11, a convenient reference axis is the basic chord line of the mean aerodynamic wing chord. (The basic chord line is usually specified along with the dimensions of the airfoil section). The determination of the size and location of the MAC is outlined in CAAM 05.217-C.

3. A tabular form will simplify the computations required to obtain the balancing loads for various flight conditions. A typical form for this purpose is shown in Table .2-II. In using Fig. .2-11 and Table .2-II the following assumptions and conventions should be employed:

a. <u>If known distances or forces are opposite in direction from those shown in Fig. .2-11, a negative sign should be prefixed before inserting in the computations</u>. For instance, in the case of a low-wing monoplane, h_2 will have a negative sign. 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 chord load acting at the tail surfaces may be neglected.



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. .2-11 - BASIC FORCES IN FLIGHT CONDITIONS
(Ref. CAAM 05.218-B)

4. Computation of Balancing Loads. In Table .2-II 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 design 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. .2-II. (Condition IV, Neg. L.A.A., will usually result in the largest balancing tail load.)

a. When the full-load center of gravity position varies appreciably the glider should be balanced for both extreme positions unless it is apparent that only one is critical. In general, only one center of gravity need be considered for single place gliders. In special 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 .2-II:

(4) 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 05.212 and CAAM 05.2134).

(8) Determine C_C as specified in CAAM 05.2134. See also Eq. 6, CAAM 05.1-C.

(10) The value of C_M' is specified in CAAM 05.2134. See also CAAM 05.217-C in cases involving wing flaps.

(12) The net tail load factor n_3 is found by a summation of moments about point (2) of Fig. .2-11, from which the following equation is obtained:

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. 05-1 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 $\bar{w}$ refers to the average limit pressure over the total effective area of the vertical surface. The total load acting is therefore equal to $\bar{w}$ 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.

05.224 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 05 apply only when the flaps are not used at speeds above a certain predetermined design speed. As noted in CAR 05.732, 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.

05.225 SPECIAL DEVICES

1. In lieu of wind tunnel data, it is recommended that spoilers and their attachment structures be designed for the limit loading obtained from the following formula:

$$w_{sp} = .0052\ V_{sp}^{2}$$

where: w_{sp} = the limit loading, psf.

V_{sp} = the indicated airspeed at which maximum operation of the spoilers is assumed, mph.

It should be assumed that the load is uniformly distributed over the surface. NACA Technical Memorandum No. 926 contains information pertaining to dive control brakes.

05.230 GENERAL

The control forces specified in CAR 05 are of an arbitrary nature; hence they may prove to be somewhat irrational in certain cases. In general, however, they represent simplified requirements which will result in satisfactory control systems. If he so desires, the designer may use a more rational loading for the design of the control system. The following loadings are considered satisfactory. The control systems may be designed for limit loads 25% greater than those corresponding to the limit loads specified in CAR 05 for the control surfaces to which they are attached, assuming the movable surfaces to be in that position which produces the greatest load in the control system, except that the loads should not be less than those listed below:

a. Elevator: 75 lbs. fore-and-aft

b. Rudder: 100 lbs. on one pedal only and
200 lbs. on each pedal simultaneously

c. Aileron: 50 lbs. laterally or as a part of a couple applied to the control wheel

The control forces specified should be applied to the entire control system including the control surface horns. The multiplying factor of safety of 1.20 (CAR 05.271) need not be applied to the fittings in the control system.

05.233 In regard to the distribution of the control force between the ailerons, the following assumptions are considered suitable:

a. For non-differential ailerons, 75% of the stick force or couple should be assumed to be resisted by a down aileron, the remainder by the other aileron; also, as a separate condition, 50% should be assumed to be resisted by an up aileron, the remainder by the other aileron.

b. For differential ailerons, 75% of the stick force or couple should be assumed to be resisted by each aileron in either the up or down position, or rational assumptions based on the geometry of the system should be made.

05.234 FLAP AND AUXILIARY CONTROL SYSTEMS

1. It is recommended that the following limit forces be used as minimum values for the design of flap and auxiliary control systems:

a. Flaps .... 50 lbs. (See note below)

b. Spoilers .... 50 lbs. (See note below)

c. Hand operated brakes .... 75 lbs.

d. Foot operated brakes ....100 lbs.

NOTE: The force used for the design of flap and spoiler control systems should not be less than 1.25 times the force corresponding to limit load used for the design of the surfaces.

2. 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.

05.235 The direction of applying the control force should correspond to the direction normally used by the pilot.

05.240 GENERAL

In so far as the requirements of CAR 05.24 are concerned, landing gears will be considered conventional if they consist of:

a. A single wheel or double co-axial wheels located on the bottom of the fuselage and directly below (or nearly so) the center of gravity of the glider, together with auxiliary skids attached to the bottom of the fuselage. One auxiliary skid running from the wheel forward to the nose, the other running aft to a point below the wing trailing edge (approximately). The rear auxiliary skid may be replaced or supplemented by a suitable tail skid (See Figures .0-2 and .2-14), or

b. A single main skid on the bottom of the fuselage which extends from the nose to a point below the wing trailing edge (approximately). This skid may be supplemented by a tail skid (see Fig. .2-14).

NOTE: Wing tip skids may be employed if desired.

05.241 LEVEL LANDING

The loading for this condition is illustrated in Fig. .2-12. As specified in CAR 05.241 the glider shall be assumed to be in a level attitude. However, as it is difficult to accurately define "level attitude", any <u>reasonable</u> attitude with the tail well off the ground will be considered satisfactory. If in the level landing condition the resultant load does not pass through the center of gravity, it will generally be acceptable to apply a balancing couple composed of an upward force acting near the nose of the fuselage and an equal downward 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 may be divided between the nearest panel points, if desired. These forces are illustrated in Fig. .2-12(a).

05.242 LEVEL LANDING WITH SIDE LOAD

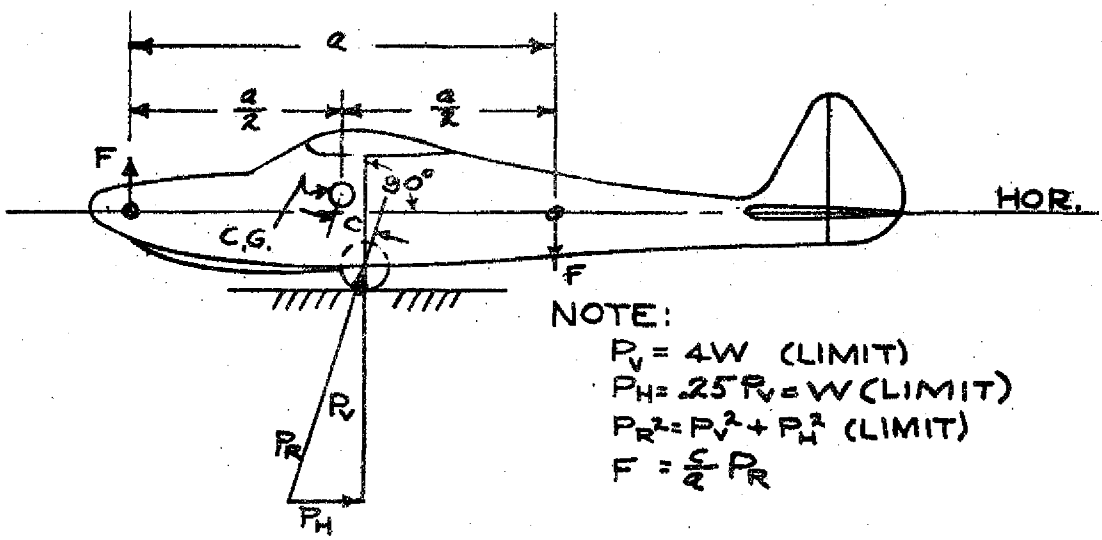
The loading for this condition is illustrated in Fig. .2-13. An acceptable method of balancing externally applied rolling moments about the longitudinal axis resulting from the side load is illustrated in Fig. .2-13(a). The forces resisting angular acceleration are assumed to be applied by the wing. 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 glider.

05.243 NOSE-DOWN LANDING

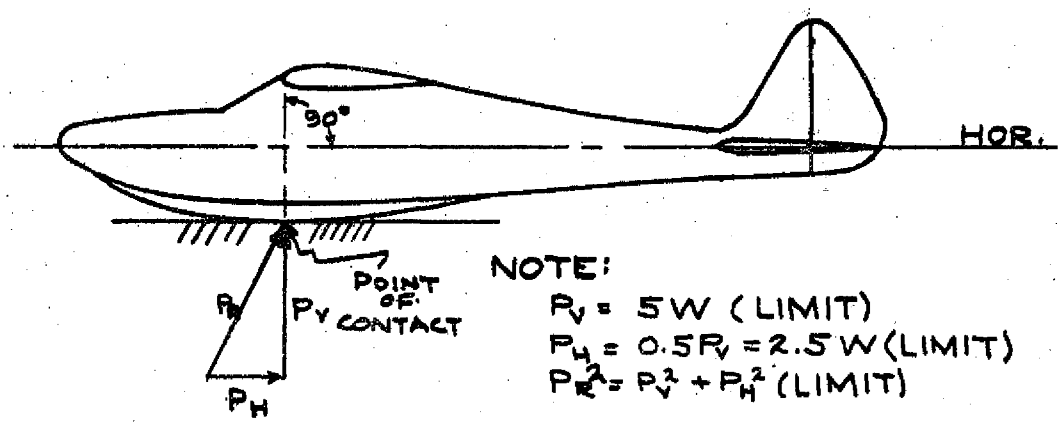
This condition is illustrated in Fig. .2-14.

05.244 HEAD-ON LANDING

It should be noted that the load factor specified in CAR 05.243 (4.0) is an <u>ultimate</u> load factor and not a <u>limit</u> load factor.

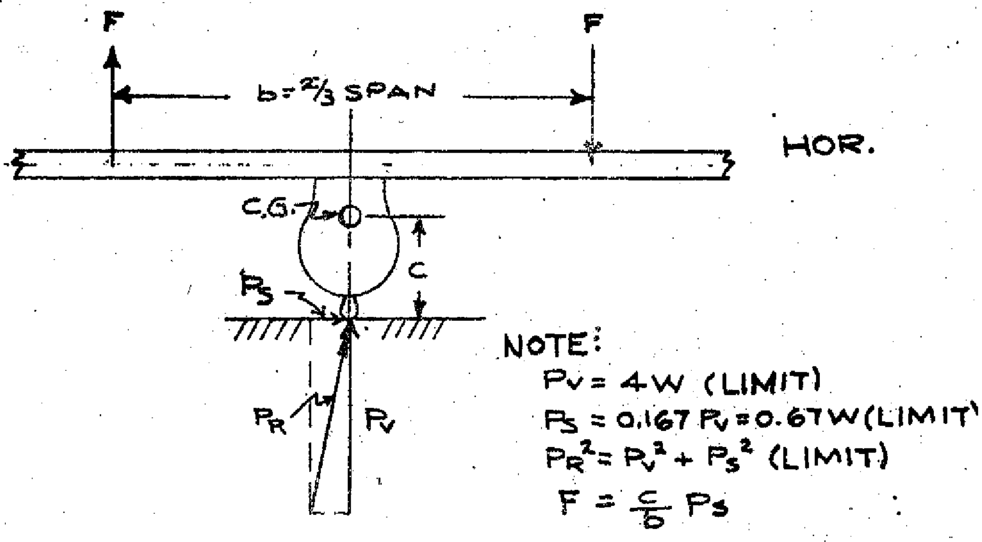


(a) For Wheel Type Landing Gears

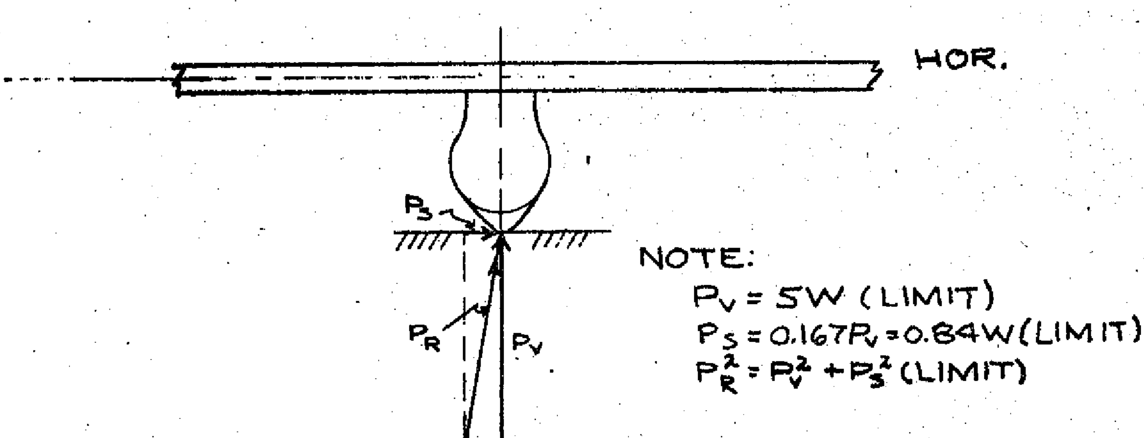


(b) For Skid Type Landing Gears

FIG. .2-12 - LEVEL LANDING
(Ref. CAAM 05.241)



(a) For Wheel Type Landing Gears



(b) For Skid Type Landing Gears

(Note: See Fig. .2-12 for Horizontal Components)

FIG. .2-13 - LEVEL LANDING WITH SIDE LOAD
(Ref. CAAM 05.242)

05.245 WING TIP LANDING

It may be assumed that a limit load of 150 pounds acts aft at the point of contact of one wing tip (or wing skid if one is used) and the ground in a direction parallel to the longitudinal axis. The unbalanced turning moment may be assumed to be resisted by:

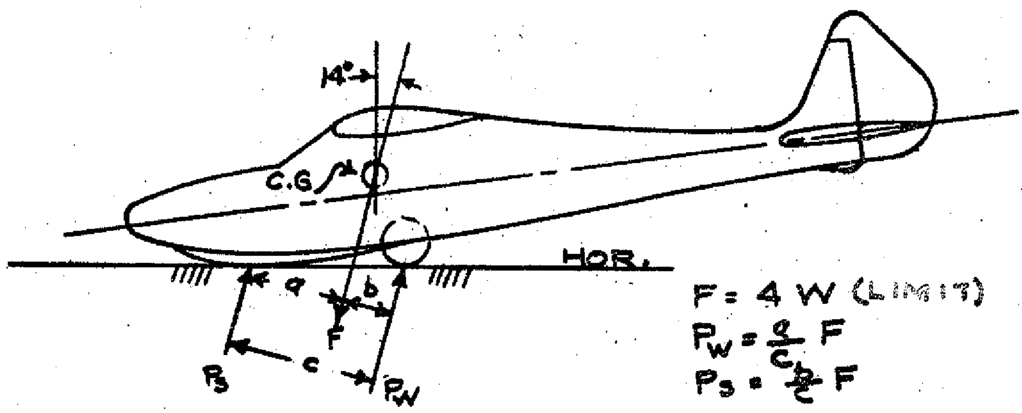
a. The methods shown in Fig. .2-15; or

b. The angular inertia of the glider.

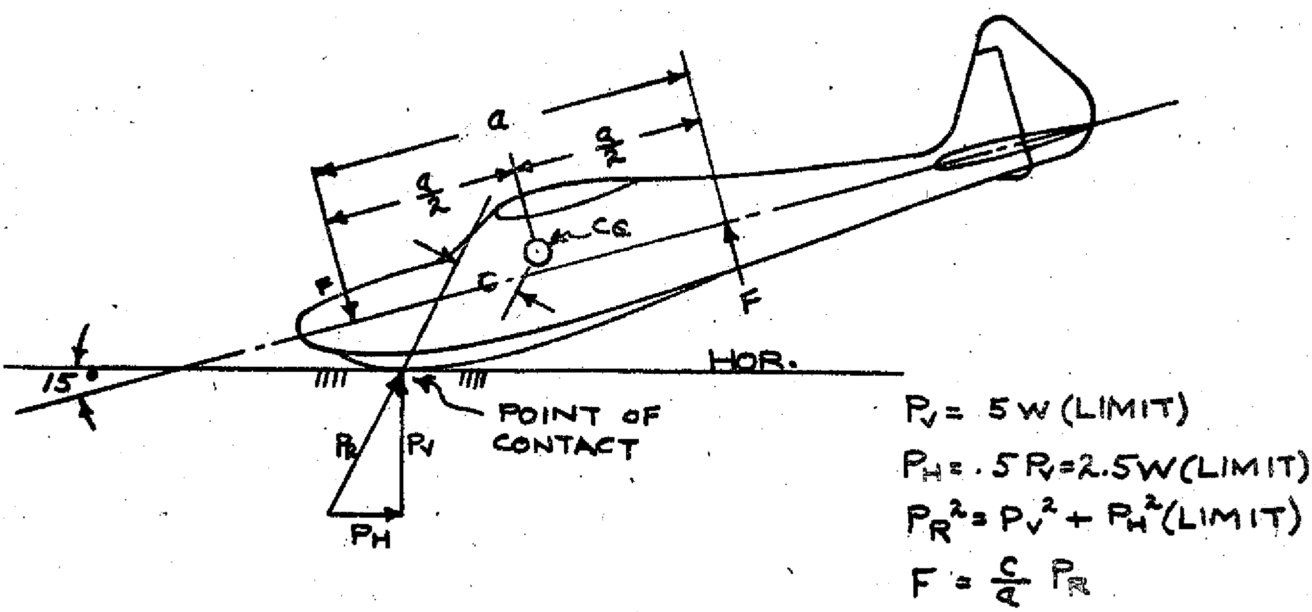
05.25 LAUNCHING AND TOWING LOADS

05.251 The loads specified in CAR 05.251 (a), (b), (c), and (d) are illustrated in Fig. .2-16. The effects of these loads need not be investigated aft of the front wing spar.

05.252 It can be assumed that a limit rearward acting chord load factor of 3.0 is developed in shock chord and winch launches.



(a) For Wheel Type Landing Gears



(b) For Skid Type Landing Gears

FIG. .2-14 NOSE-DOWN LANDING
(Ref. CAAM 05.243)

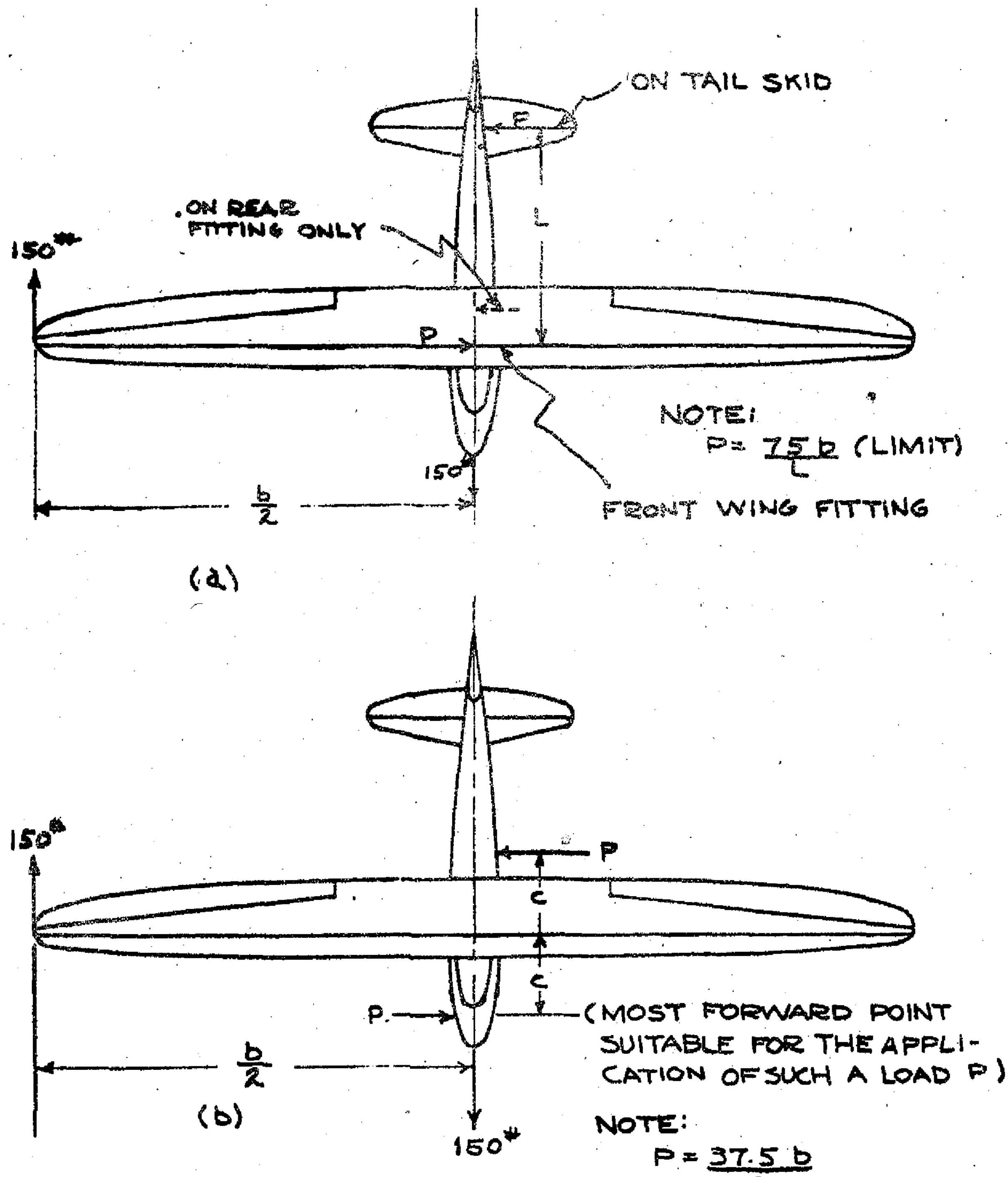
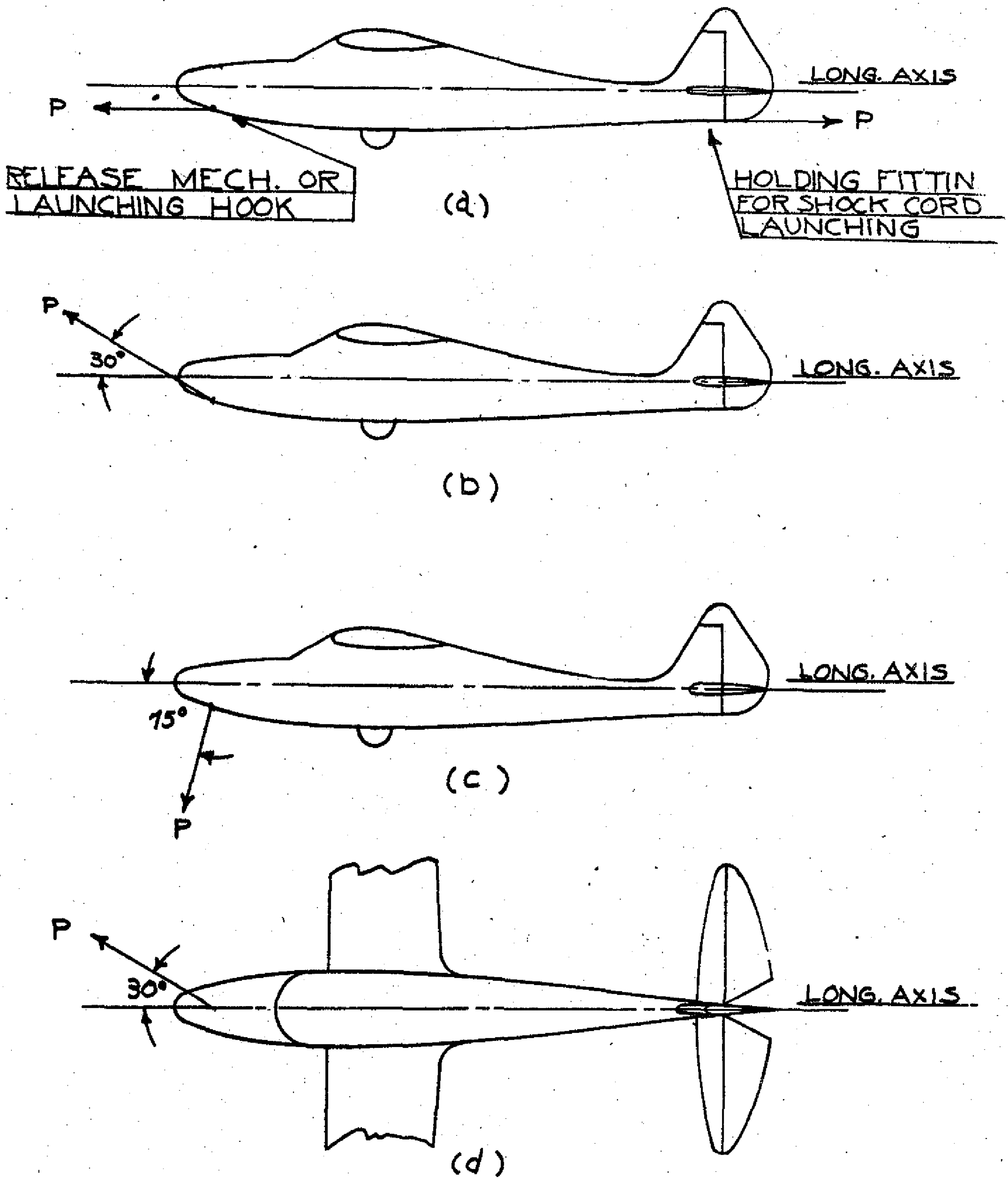


FIG. .2-15 - WING TIP LANDING
(Ref. CAAM 06.245)



Note: P = 1200 pounds or 3.0 times the gross weight, whichever is greater (limit).

FIG. .2-16 - LAUNCHING AND TOWING LOADS
(Ref. CAAM 05.251)

05.31 DETERMINATION OF LOADINGS

05.310 WINGS

05.3100 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:

$$y_f = \left[\left\{ C_N\ (r-a) + C_{M_a} \right\} q + n_2\ e\ (r-j) \right] \frac{C'}{144\ b} \qquad (.3\text{-}1)$$

$$y_r = \left[\left\{ C_N\ (a-f) - C_{M_a} \right\} q + n_2\ e\ (j-f) \right] \frac{C'}{144\ b} \qquad (.3\text{-}2)$$

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. .3-1 and are all expressed as fractions of the chord at the station in question.

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

q = dynamic pressure for the condition being investigated.
C_N and C_{M_a} are the airfoil coefficients at the section in question.
C' is the wing chord, in inches.
e is the average unit weight of the wing, in pounds per square foot, over the chord at the station in question. It should be computed or estimated for each area included between the wing stations investigated, unless the unit wing weight is substantially constant, in which case a constant value may be assumed.
n_2 is the net limit load factor representing the inertia effect of the whole glider 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 CAAM 05.218.

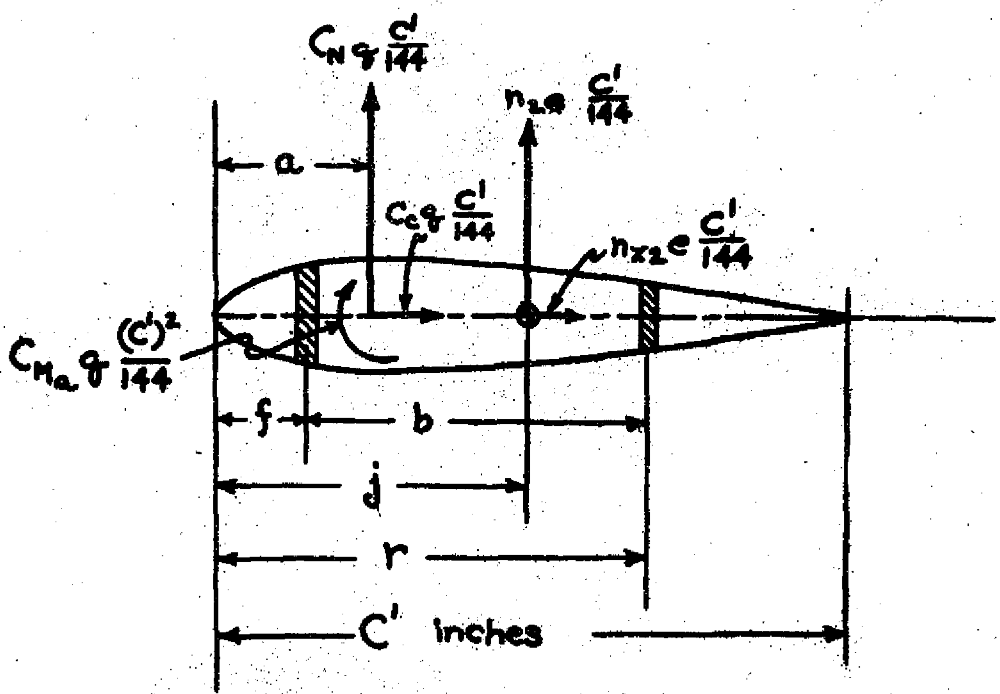
TABLE .3-I

COMPUTATION OF NET UNIT LOADINGS (CONSTANTS)

		Stations Along Span				
1	Distance from root, inches					
2	C'/144 = (chord in inches) /144					
3	f, fraction of chord					
4	r, fraction of chord					
5	b = r - f =(4) - (3)					
6	a, fraction of chord (a.c.)					
7	j, fraction of chord					
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)					

3. The computations required in using the above method are outlined in Tables .3-I and .3-II, 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.



ALL VECTORS ARE SHOWN IN POSITIVE SENSE

FIG. .3-1

UNIT SECTION OF A CONVENTIONAL 2-SPAR WING

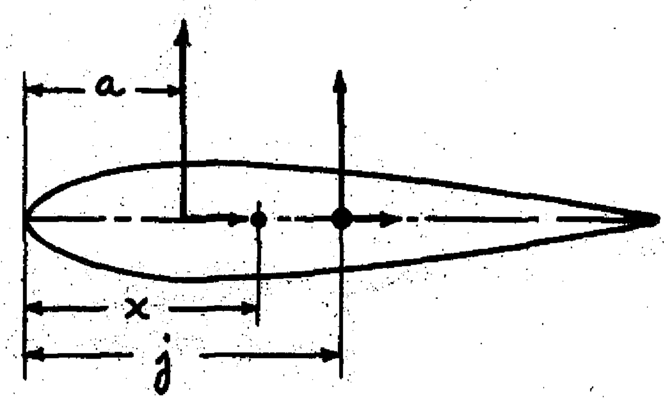


FIG. .3-2

SECTION SHOWING LOCATION OF ELASTIC AXIS

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. .2-8(c) is used for this purpose, together with the value of K_b obtained for this figure, as outlined in CAAM 05.217-C-3.

c. Item 15 provides for a variation in the local value of C_M.

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).

05.3101 DETERMINATION OF RUNNING CHORD LOAD

1. The net chord loading, in pounds per inch run, can be determined from the following equation:

$$y_c = \left[C_c\, q + n_{x2}\, e \right] C'/144. \qquad (.3\text{-}3)$$

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 glider in the chord direction. The value and sign are obtained in the balancing process outlined in CAAM 05.218. Note that when C_c is negative, n_{x2} will be positive.

e and C' are the same as in CAAM 05.3100.

2. The computations for obtaining the chord load are outlined in Table .3-II, Items 28 to 32. The following points should be noted:

a. The value of C_c, item 28, can usually be assumed to

TABLE .3-11

COMPUTATION OF NET UNIT LOADINGS (VARIABLES)

CONDITION

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

		(Refer also to Table .3-1)	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						

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 CAAM 05.123-A2 and Fig. .1-4.

05.3102 DETERMINATION OF RUNNING LOAD AND TORSION AT A GIVEN 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. .3-2, x denotes the location of the reference axis, expressed as a fraction of the chord. The net running load along the locus of the points x and the net running torsion about these points are found from the following equations:

$$y_x = (C_N\, q + n_2\, e)\, \frac{C'}{144} \qquad (.3\text{-}4)$$

$$m_x = \left[\left\{ C_N\, (x-a) + C_{M_a} \right\} q + n_2\, e\, (x-j) \right] \frac{(C')^2}{144} \qquad (.3\text{-}5)$$

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 <u>inches</u>.

The remaining symbols are explained in CAAM 05.3100. (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 .3-I and .3-II. 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 CAAM 05.3101.

05.311 CONTROL SYSTEMS

05.3110 FLIGHT CONTROLS

The minimum loads to be used for the design of flight control systems are covered in CAR 05.230-233 and CAAM 05.230-233.

05.3111 SECONDARY CONTROLS

The minimum loads to be used for the design of secondary control systems are covered in CAR 05.234 & 05.235 and CAAM 05.234 & 05.235.

05.312 FUSELAGES

05.3120 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 gliders:

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 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.

c. 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.

d. 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.

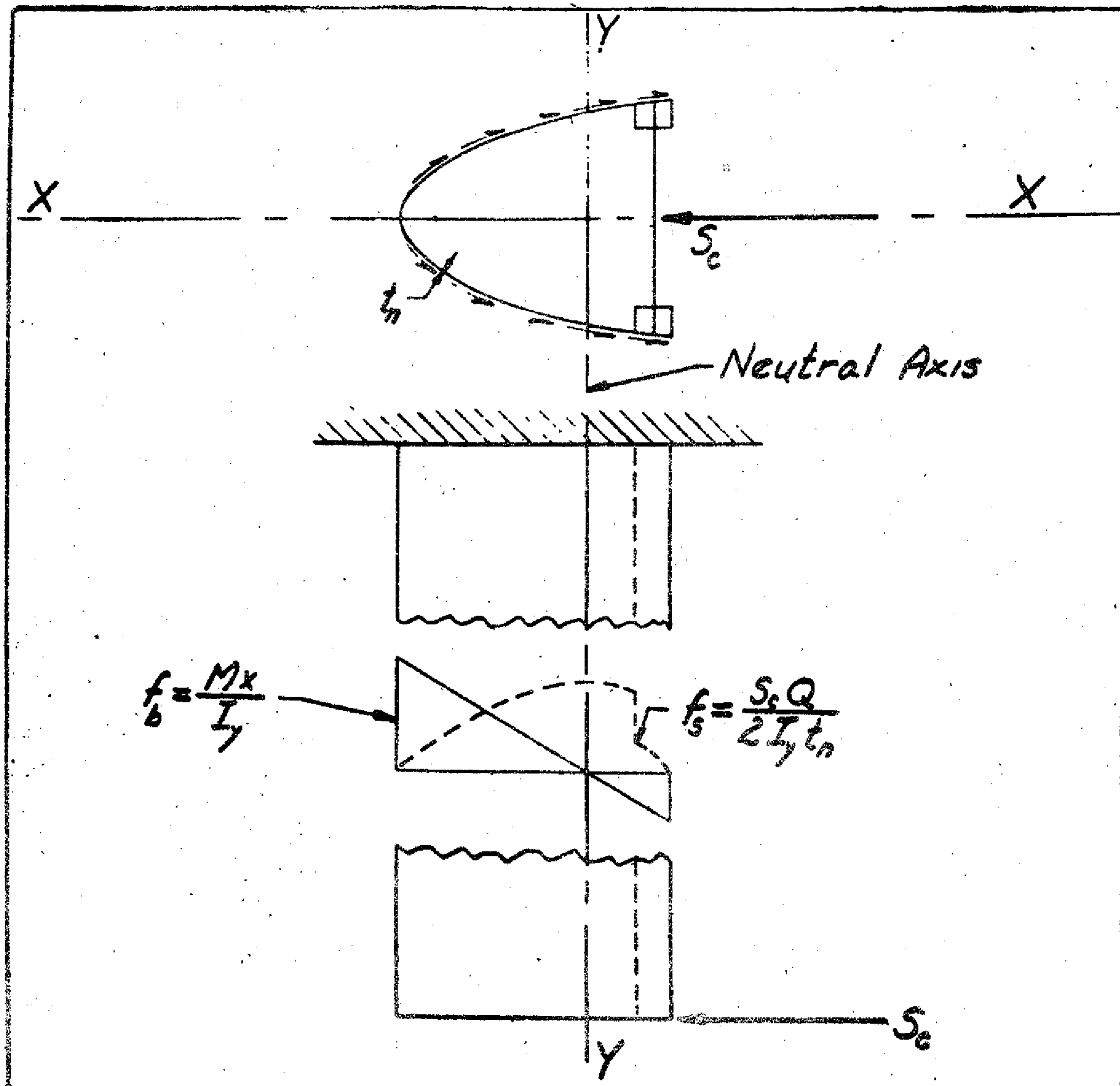

Y
X
X
S_c
t_n
Neutral Axis
$f_b = \frac{Mx}{I_y}$
$f_s = \frac{S_c Q}{2 I_y t_n}$
S_c
Y

FIG. 3-6

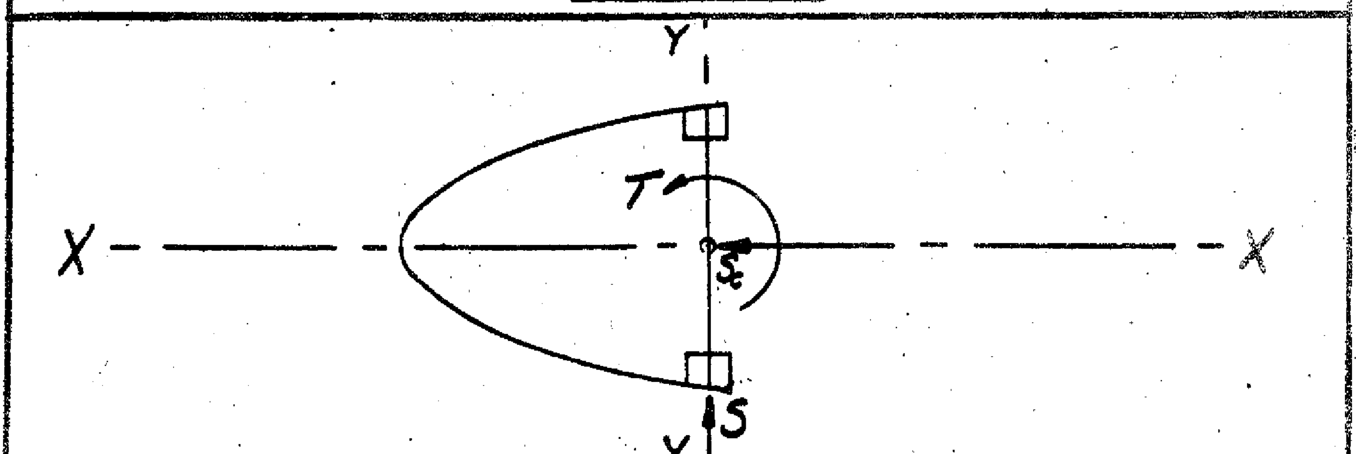

Y
T
X
X
S_c
S
Y

FIG. 3-7

05.32 STRUCTURAL ANALYSES

05.320 GENERAL

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 twenty-five cents.

05.321 PROOF OF WINGS

05.3210 GENERAL

Proof of wings by structural analysis only should be in accordance with CAR 05.32. The following points are pertinent:

1. Joint slippage in wood beams. When a joint in a wood beam is designed to transmit bending from one section of the beam to another or to the fuselage, the stresses in each part of the structure should be calculated on the assumption that the joint is 100 per cent efficient and also under the assumption that the bending moment transmitted by the joint is 75 per cent of that obtained under the assumption of perfect continuity. Each part of the structure should be designed to carry the most severe loads determined from the above assumptions.

2. Bolt holes. In computing the area, moment of inertia, etc., of wood beams pierced by bolts, the diameter of the bolt hole should be assumed to be one-sixteenth inch greater than the diameter of the bolt.

3. In computing the ability of box beams to resist bending loads only that portion of the web with its grain parallel to the beam axis and one-half of that portion of the web with its grain at an angle of 45° to the beam should be considered. The more conservative method of neglecting the web entirely may be employed.

4. Drag trusses. Drag struts should be assumed to have an end fixity coefficient of 1.0 except in cases of unusually rigid restraint, in which a coefficient of 1.5 may be used.

05.3211 DETERMINATION OF LOADING CURVES

1. The loading curves can be determined by the general methods given in CAAM 05.310, but for simplicity, a direct comparison of the Beam, Chord, and Torsion coefficients times the respective factors will give an indication of which conditions are going to be critical, especially for a single spar design. Thus, it may be possible to eliminate one or more conditions before plotting the actual curves.

2. The distribution of the wing dead weight can be estimated and shear and moment curves plotted. The dead weight curves times the dead weight load factor n_2 in the beam and chord directions are subtracted from the total respective limit beam and chord load curves to obtain the "net" limit load curves.

3. It will simplify the curves somewhat to figure and construct all net wing shear and moment curves to the root of the wing, even though it may be externally braced. The loads in the strut and its effect on the wing curves can be superimposed on the latter to determine the final curves.

4. In constructing shear curves from a curved running load, it is necessary to integrate the loading curve to obtain the shear, or to approximate the curve with a series of straight lines so that it can be calculated as a series of trapezoids. Moment curves are developed similarly from the shear curves.

05.3212 TWO SPAR WINGS

Methods of analysis for conventional two spar wings are contained in standard textbooks on airplane structures.

A WOOD SPARS

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

B METAL SPARS

1. 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 aircraft. If this is not practicable, the shear on the test beam should be relatively larger than in the aircraft. 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.

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.

3. 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.

05.3213 D-NOSE SPARS

A TWO FLANGE TYPE

1. Basic Principles and Assumptions.

Insofar as the structural analysis is concerned, the D-nose spar with two flanges may be considered as the combination of a beam (flanges and vertical web) and a torsion box (leading edge skin and vertical web). A typical structure of this type is shown in Figure .3-3. In the analysis it is assumed that the spar does not taper and that the structure is symmetrical, insofar as the two concentrated flanges are concerned, about a horizontal axis (a principal axis). It is further assumed that the structure is of such a nature that the engineering theories of bending (flexure formula), shear and torsion (membrance formula) are applicable.

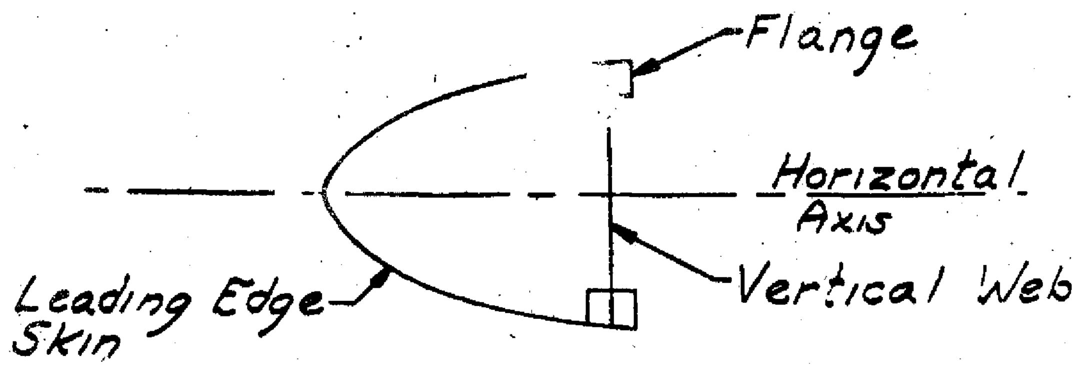


FIG .3-3: TYPICAL D-NOSE SPAR WITH TWO FLANGES

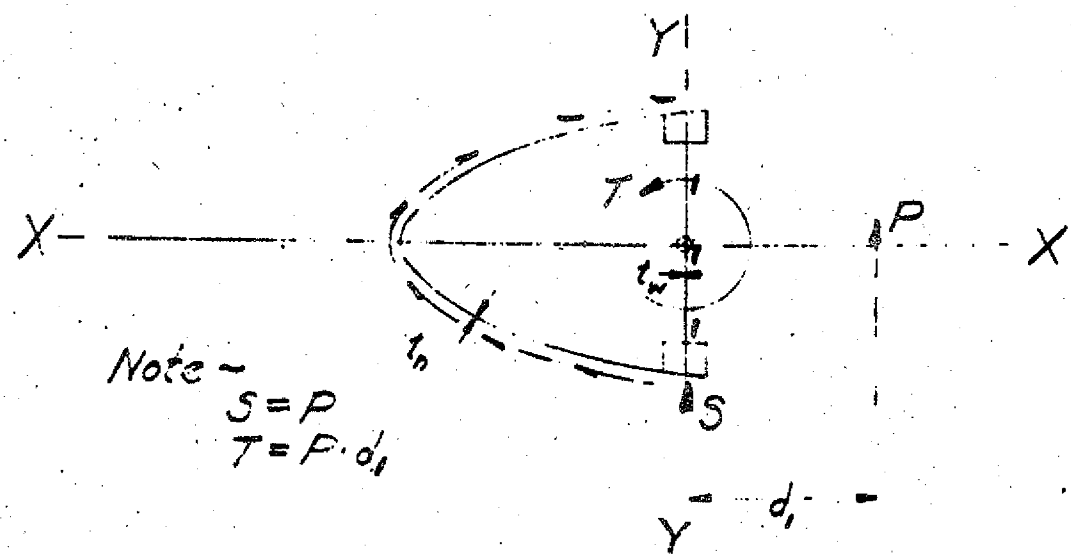


FIG .3-4

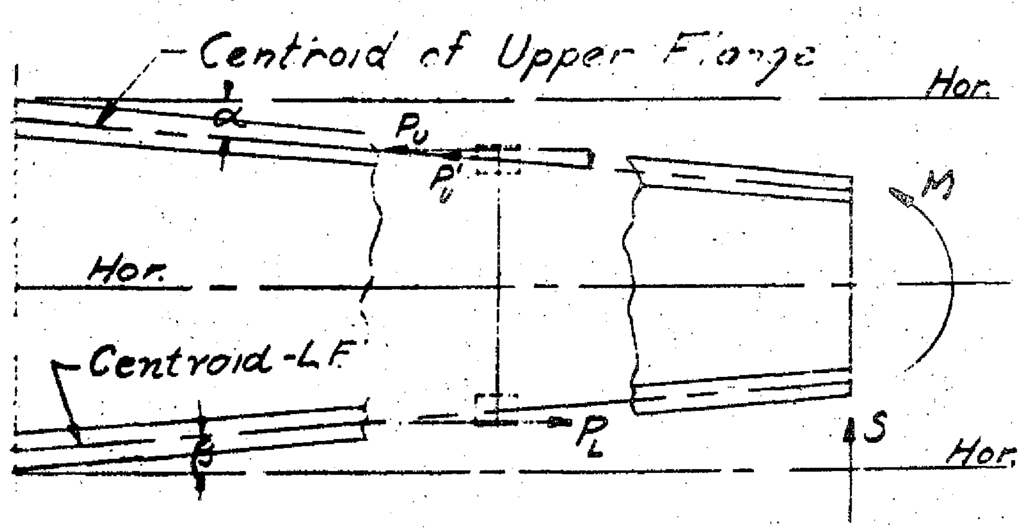


FIG .3-5

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2. Detailed Analysis for Beam Loads.

For the case of beam loading only, the spar will be subjected to a force P (see Fig. .3-4). For the purpose of practical structural analysis, it is desirable to replace the force P by a force S acting in the plane of the web and a torsional moment (or couple) acting about a point in the web. This conversion is illustrated in Figure .3-4.

a. Stress in Leading Edge Skin.

In this case, where the bending material is concentrated in two flanges (top and bottom of web), the leading edge skin will be stressed primarily by the torsional moment M. (Actually the leading edge skin adjacent to the flanges will be stressed by the bending loads.) The stress in the skin can be obtained from one of the formulas given below:

(1) If the structure and magnitude of the torsional moment are such that the skin does not wrinkle under load, the skin will be stressed in shear, and the following formula will apply:

$$f_{s_n} = \frac{T}{2At_n} \quad \text{(Membrane Formula)} \qquad (.3\text{-}6)$$

Where: T = the torsional moment in inch pounds.
A = the area enclosed by the leading edge skin and the web in square inches.
t_n = the thickness of the leading edge skin in inches.

(2) If the skin wrinkles under load, it will be stressed in tension (tension fields), and the following formula will apply:

$$f_{t_n} = \frac{T}{At_n} \qquad (.3\text{-}7)$$

For obvious reasons the skin should not wrinkle appreciably when stressed below the limit load.

b. Stress in Concentrated Flanges.

If the structure is not subjected to axial loads and does not taper in depth, the flanges will be stressed primarily by the bending load. The flange stress can be determined from the following formula:

$$f_b = \frac{My}{I_x} \quad \text{(Flexure Formula)} \qquad (.3\text{-}8)$$

where M = the bending moment in inch pounds.
y = the distance from the neutral axis to the flange fiber being investigated in inches. (Obviously the outermost (or extreme) fiber will be most highly stressed).
I_x = the moment of inertia of the flanges about their neutral axis (the X-X axis) in inches to the fourth power.

If the flange material is concentrated, or nearly so, the above formula closely approaches:

$$f_b = \frac{M}{hA_f} \qquad (.3\text{-}9)$$

where h = the distance between the centroids of the two flanges (upper and lower) in inches.
A_f = the cross sectional area of the flange in question (upper or lower) in square inches.

c. Stress in Web.

The web will be stressed primarily by the shear S and the torsional moment T. (In certain cases it may be desirable to investigate the effect of bending at the junction of the web and the flanges.) The stress in the web can be determined from one of the formulas given below.

(1) If the structure and the magnitude of the loadings are such that the web does not wrinkle under load, the web will be stressed in shear and the following formula will apply:

$$f_{s_w} = \frac{SQ}{I_x t_w} + \frac{T}{2At_w} \qquad (.3\text{-}10)$$

where S = the shear in pounds.
Q = the static moment about the neutral axis of the section effective in bending above <u>or</u> below the point in question

in inches cubed. (For the remaining symbols, see a and b above.)

If the flange material is concentrated, the above formula becomes:

$$f_{s_w} = \frac{S}{ht_w} + \frac{T}{2At_w} \qquad (.3\text{-}11)$$

where h = the depth of the web in inches.

(2) If the web wrinkles under load, it will be stressed in tension (tension fields), and the following formula will apply:

$$f_{t_w} = \frac{2S}{ht_w} + \frac{T}{At_w} \qquad (.3\text{-}12)$$

d. Effect of Taper.

The effect of taper in spar depth is twofold, namely: (1) a decrease in the amount of beam shear resisted by the vertical web, and (2) a relatively slight increase in the axial loads in the flanges of the beam.

(1) Referring to Figure .3-5, it can be seen that the shear S' carried by the web of a tapered beam is expressed by the following formula:

$$S' = S - (P_U \tan a \;\; P_L \tan \beta) \qquad (.3\text{-}13)$$

where S = the external shear at the section in pounds.

P_U = the horizontal component of the axial load in the upper flange due to bending in pounds. (It may be obtained by multiplying the stress determined from Formulas .3-8 or .3-9 by the cross sectional area of the upper flange.)

P_L = the horizontal component of the axial load in the lower flange.

(For a and 8, see Figure .3-5.)

(2) Although the effect of taper on the axial load in the beam flanges is usually so small that it can be safely neglected; it should be considered in the analysis of the critical sections of highly tapered beams. The true axial loads in the flanges, P_U' and P_L', can be closely approximated from the following formulas:

$$P_U' = P_U \sec \alpha \qquad (.3\text{-}14)$$

and

$$P_L' = P_L \sec \beta \qquad (.3\text{-}15)$$

For the meaning of the symbols, see Formula (.3-13) and Figure .3-5.

e. Effect of Axial Load.

If the spar is subjected to an axial load P in addition to the shear and bending moment, the flange stresses can not be determined from the formulas given above. The flange stresses can, however, be obtained from the following formulas:

$$f = \frac{M'y}{I_x} + \frac{P}{A} \qquad (.3\text{-}16)$$

where M' = the <u>precise</u> bending moment in inch pounds. (The precise moment includes the effect of secondary bending due to the axial load).
P = the axial load in pounds.
A = the cross sectional area of the flanges (upper and lower) in square inches.

(For the meaning of the other symbols, see Formula (.3-8)).

3. Detailed Analysis for Chord Loads.

A chord loading will subject the D-nose spar to both bending and shear stresses. The resulting stress distribution is shown in Figure .3-6.

a. Bending Stress.

The stress due to bending can be determined from the following formula:

$$f_b = \frac{Mx}{I_y} \qquad (.3\text{-}17)$$

where M = the bending moment about the Y-Y axis in inch pounds.

x = the distance from the neutral axis to the fiber being investigated in inches. (In general, the most forward portion of the leading edge skin (an external fiber) will be the most highly stressed).

I_y = the moment of inertia about the neutral axis (the Y-Y axis) in inches to the fourth power.

b. Shear Stress

The stress due to shear can be determined from the following formula:

$$f_s = \frac{SQ}{2I_y t_n} \qquad (.3\text{-}18)$$

where S = the shear in pounds.

Q = the static moment about the neutral axis of the section above or below the point being investigated in inches cubed.

t_n = the thickness of the leading edge skin in inches.

4. Combined Loadings.

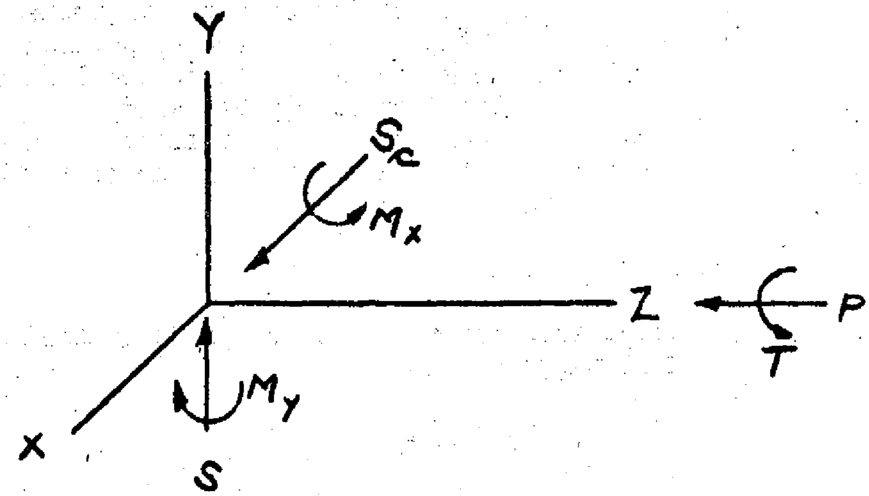
In the general case, a D-nose spar may be subjected to the loadings shown in Figure .3-7 together with an axial load P. The effects of these loadings and the pertinent formulas for determining the corresponding stresses can be determined from Table .3-III.

5. Limitations of Formulas.

The accuracy of the foregoing formulas will not be high in the vicinity of the root of the spar because the actual stress distribution will differ considerably from the assumed distribution. It is, therefore, recommended that high computed margins of safety be maintained in the root region. The theory also breaks down to some extent if stress concentrations are present. Therefore, the design should be of such a nature that serious stress concentrations will not exist.

TABLE .3-III

LOADINGS, TYPE OF STRESS AND PERTINENT FORMULA FOR D-NOSE SPARS



Component	Loading	Type of Stress	Formula
L. E. Skin	Beam (S, M_x & T)	Shear or Tension	(.3-6) or (.3-7)
" " "	Beam + Axial Load (S, M_x, T & P)	" "	" "
" " "	Chord (S_c & M_y)	Shear & Bending	(.3-17) and (.3-18)
Flanges	Beam (S, M_x & T)	Axial (due to bending)	(.3-8) or (.3-9)
"	Beam & Axial Load (S, M_x, T & P)	" (due to bending) and axial load	(.3-16)
"	Chord (S_c & M_y)	Axial (due to bending)	(.3-17)
Web	Beam (S, M_x & T)	Shear or Tension	(.3-10), (.3-11) or (.3-12)
"	Beam + Axial Load (S, M_x, T & P)	" "	(.3-10), (.3-11) or (.3-12)
"	Chord (S_c & M_y)	Axial (due to bending)	(.3-17)

B DETERMINATION OF ALLOWABLE STRESSES AND MARGINS OF SAFETY

1. Allowable Stresses.

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

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

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

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

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

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

2. Margins of Safety.

a. Leading Edge Skin.

The M.S. of the L.E. skin will be positive if

$$\left(\frac{f_b}{F_b}\right)^x + \left(\frac{f_s}{F_s}\right)^y \leq 1.0$$

compliance with yield requirements. Inasmuch as many control surfaces do not lend themselves to rigorous analysis, it is recommended that strength tests be considered for proving compliance with the ultimate load requirements.

The analysis and tests should include the horns, and should demonstrate compliance with multiplying factors of safety requirements contained in CAR 05.27.

05.3221 DEFLECTION OF HINGE POINTS

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.

05.3222 RIGGING LOADS

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.

05.323 PROOF OF CONTROL SYSTEMS

05.3230 GENERAL

Structural analyses of control systems will be accepted as complete proof of compliance with ultimate load requirements when the structure conforms with conventional types for which reliable analytical methods are available. Proof tests as defined in CAR 05.121 are required to prove compliance with yield load requirements.

Analyses or individual load tests should be conducted to demonstrate compliance with the multiplying factor of safety requirements specified in CAR 05.27 control system joints subjected to angular motion.

Operation tests (see CAR 05.342) are required in addition to the proof tests and analyses.

05.3231 CRITICAL LOADINGS

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. The proof of control systems should include the most severe loading conditions for all parts of the system.

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, wing flap, and tab control systems.

05.324 PROOF OF LANDING GEARS

05.3240 GENERAL

Structural analyses of landing gears will be accepted as complete proof of compliance with the load requirements when the structure conforms with conventional types for which reliable analytical methods are available.

05.3241 WHEELS AND TIRES

In wheel type landing gears, the ultimate strength of wheels and the bursting strength of tires should not be less than the <u>ultimate</u> loads to which they are subjected.

05.3242 SHOCK ABSORPTION

There are no definite requirements regarding the energy absorption characteristics of glider landing gears. On heavy gliders either air-wheels or shock absorbing skids should be used.

05.325 PROOF OF FUSELAGES

05.3250 GENERAL

Proof of fuselages by structural analysis only should be in accordance with CAR 05.32.

05.3251 SHEAR AND BENDING MOMENT DIAGRAMS

1. It will be found an advantage in analyzing all types of fuselages to construct shear and bending moment curves for all the critical design conditions.

2. Determine the distribution of the fuselage weights from the weight (see CAAM 05.032, e and f) and balance calculations and construct the unit shear and bending curves about the center of gravity.

3. For each critical condition, multiply the unit shears and moments by the load factor perpendicular to the fuselage axis, and balance with the previously determined wing and tail reactions. The net fore and aft acceleration can be applied at the C.G., as the distribution of these components individually at the various panel points is unnecessarily complicated.

4. Select the critical condition for shear and the critical condition for bending for each fuselage panel, whether monocoque or truss, and find the internal panel loads by taking a section through the panel near the joint and solving for the loads in the members.

05.3252 TRUSSED TYPE FUSELAGES

A GENERAL

Methods of analysis for conventional trussed type fuselages are covered in standard textbooks on airplane structures.

B TORSION

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.

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

2. For the structure aft of the rearmost bulkhead the tail load may be represented by a side load acting at the center of the tail post and a couple equal to this load times its vertical distance from the center of pressure of the vertical tail. The side load may be assumed to be divided equally between top and bottom trusses. For the structure forward of the rearmost bulkhead the tail load may be represented by a side load acting at the center of the tail post and a torque acting at the rearmost bulkhead equal to the tail load times the vertical distance from the center of pressure of the vertical tail to the center of this bulkhead. This side load may be assumed to be divided equally between top and bottom trusses. The assumption may be made that the torque (not the forces composing the equivalent couple) is divided equally between the horizontal and vertical trusses. The couples acting on the bulkhead and resisted by the top, bottom, and side trusses can then be readily obtained. Stress diagrams should be drawn for the trusses to obtain the loads in the members. The longeron loads should be taken from the diagrams for the horizontal trusses 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.)

3. 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.

4. 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.

05.3253 STRESSED-SKIN TYPE FUSELAGES

A GENERAL

1. The strength of skin-stressed fuselages is affected by a large number of factors, most of which are difficult to

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account for in a stress analysis. The following are of special importance:

a. Effects of cut-outs.
b. Behavior of 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.

B WOOD "MONOCOQUE" FUSELAGES

The following comments can be used as a guide:

1. Wood "monocoque" fuselages may be treated as beams, the longerons forming the flanges and the sides being the webs. The allowable stresses for the longerons can be taken from charts for beam allowables, provided the member is properly supported against column failure by the covering. If not, the member should be checked as a column between frames. If the member is continuous, the coefficient of restraint C can be taken as 1.5. Neglecting the effect of the covering outside the longerons, especially if the fuselage is curved on top and bottom, will give conservative results, as the plywood covering can carry some compression. The amount it can carry depends on the thickness, bulkhead spacing, and curvature. The most satisfactory method is to test a suitable section in order to determine the allowable in case it is desired to take this material into account.

2. The shear stress in the sides can be calculated in the same manner as for a beam, using the same beam section pro-

perties as used for finding the bending stress.

3. The torsional shear caused by the eccentricity of the fin and rudder load above the axis of the fuselage can be calculated by the same method as for a D-nose wing spar (see Formula .3-6). The allowable shear stresses will vary with curvature and bulkhead spacing, but in general can be determined by the method used for D-nose wing spars.

4. An approximation of the loads in the bulkhead vertical members can be made by assuming the adjacent bay is braced by a shear diagonal instead of by being completely covered with plywood and solving for the load in the vertical as in a truss type fuselage.

C METAL "MONOCOQUE" FUSELAGES

The following comments can be used as a guide:

1. Metal "monocoque" fuselages can be treated as beams in bending, utilizing the full section material unless the covering is very light and unsupported at the extreme fibers.

2. An approximate method is to disregard all the skin material and calculate the moment of inertia of the section on the basis of the longitudinal stringers only. In this case the allowable stress should be taken as that for the stringer as a column.

3. The most accurate method is to calculate the "effective width" of skin which can be counted on to be acting as a unit with each stringer. The stringer-skin unit is assumed to be a column between bulkheads. For details on this method refer to ANC-5.

4. The shear in the covering is calculated from the static moment of the section as with any beam. The bulkhead members are not likely to be critical in compression unless the shear stress is so high that the side of the fuselage becomes a "tension field". The stress at which a panel will become a tension field depends upon the size and shape of the unsupported panels, and the curvature, as well as the thickness. The stress at which a panel will start to wrinkle is usually judged from experience and substantiated by static tests. If computations show that the panel will wrinkle before reaching the ultimate design load, the bulkheads and riveting should be designed to suit this condition.

5. When the sheet wrinkles to form tension fields, the allowable stress for the side panel may be taken as 0.7 of the ultimate tensile strength.

05.326 PROOF OF FITTINGS AND PARTS

05.3260 GENERAL

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 05-3) 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.

05.33 COMBINED ANALYSIS AND TESTS

05.330 GENERAL

Structural analyses are acceptable for proving compliance of conventional structures with the <u>ultimate</u> load requirements. In all cases, certain <u>proof</u> and <u>operation</u> tests are required by CAR 05.342. For unconventional structures, for which reliable analysis methods have not been developed, it is necessary to resort to combined analysis and tests or <u>strength</u> tests only to prove compliance with the <u>ultimate</u> load requirements.

05.331 D-nose wing spars and "monocoque" fuselages are considered as unconventional structures, and hence usually can not be substantiated by analysis methods alone. Such structures can, however, be substantiated by combined analysis and tests (see CAAM 05.3213).

05.332 Before deciding on whether combined analysis and tests or strength tests alone are most advantageous for substantiating unconventional structures, it is recommended that an investigation of the work involved be conducted. The following points should be considered in such an investigation:

1. Components which have been strength tested cannot be used in certificated gliders unless it can be shown that no damage has occurred in any part of the structure or that such damage has been properly repaired.

2. Unless test results are reduced to correspond to the minimum mechanical properties of the materials used (which requires additional tests), the test loads must equal or exceed 115 per cent of the ultimate design loads for structures other than those actually used in the glider.

3. In certain cases, the cost of structural analysis may be less than the cost of strength tests. This is especially true when extra components must be constructed for test purposes.

4. Gliders may be built sufficiently over-strength so that tests on the entire structure may be conducted without causing failure or permanent set of any part. It is recommended that a certain amount of checking be made by an engineer in order to safeguard the builder, but no analysis need be turned in to the Authority if complete strength tests are conducted.

05.34 LOAD TESTS

05.343 TEST LOADS, APPARATUS AND METHODS

05.3430 GENERAL

1. Purpose. The definition of a "strength" test states that it is used for determining the ability of the structure to withstand its "ultimate" load. Ordinarily, the stress analysis determines this satisfactorily, but in some cases an analysis cannot be relied on, or the manufacturer may not wish to submit a complete stress analysis. "Strength" tests are therefore specified in such cases as an available method for determining whether the structure will fail before it reaches the required "ultimate" load.

2. Destruction tests. When a static load test is carried to the point where the maximum carrying capacity of the structure is reached, the test is usually referred to as a destruction test. A "strength" test is not necessarily a destruction test, as it is not always necessary to carry the loading to the point of maximum capacity in order to meet the requirements.

3. "Strength" tests in lieu of stress analysis. "Strength" tests may be conducted in lieu of a stress analysis, provided

that the test program is submitted for approval before the test is conducted. One reason for submitting the program in advance is to be sure that special provision has been made to test all members or portions of the structure requiring excess factors of safety. A certain amount of calculation is necessary in any case, in order to determine the magnitude and distribution of the test loads, as well as the critical condition for which tests should be made. Since "strength" tests in lieu of stress analysis must be used as a basis for the approval of all structures to be built from the drawings for which approval is desired, the question of material variations from standard and possible discrepancies between the test specimen and drawings is of considerable importance.

05.34300 MATERIAL TESTS

1. Standard properties. Drawings which are to be approved as a basis for a type certificate will specify certain minimum guaranteed material properties, usually by reference to existing standard specifications (S.A.E., Army, etc.). The manufacturer is not required to substantiate the strength characteristics of the materials used when a standard specification is available. In cases where some new material is to be used, the manufacturer will be required to submit test data substantiating the properties to be assumed as minimum.

2. Stress-strain diagrams. In general, the most useful data are obtained from a stress-strain diagram obtained in a tension test and such diagrams should be obtained in all cases where new materials are used. This diagram permits the determination of the following important characteristics:-

a. Ultimate tensile stress.
b. Yield point in tension.
c. Modulus of elasticity (E).

Further information on stress-strain diagrams can be obtained from ANC-5.

3. Special materials tests. Special tests may be required in order to account for factors difficult to evaluate in a stress analysis. The effects of welding after heat-treatment are difficult to predict in some cases. Stress-concentration caused by poor detail design will often reduce the allowable stresses considerably below standard values. Most metals show marked reductions in allowable stresses after being subjected to large alternating stresses for some time.

05.34301 METHODS OF CORRECTING TO STANDARD

1. General. A knowledge of the types of failures which occur in structures subjected to static load tests is essential in order that proper correction factors may be applied to reduce the results to standard conditions (for material having the minimum guaranteed strength properties). It is apparent that a built-up structure is subject to failure in many different ways and at different places. In general it is only necessary to derive correction factors for the particular portion in which the failure occurred. When the total load sustained is 15% greater than the "ultimate" load required, no material corrections are necessary.

2. Method of applying. The correction should be made by multiplying the test load sustained at failure, by the ratio of standard strength of the material to the strength of a specimen taken from the structure. The particular strength property involved will depend largely on the nature of the failure, but in general it is desirable to obtain a stress-strain diagram for the material specimen. A chemical analysis might be required if there is doubt as to the actual material used in the test structure.

05.34302 TEST PROCEDURE

1. Jigs. Tests of tail surfaces, wings and other units of that character may be conducted by mounting the surface to be tested to a specially built framework, using the regular attachment fittings of the unit being tested. The jig or framework should conform to the glider structure as far as possible. In cases where the attachment of a component to the fuselage involves the distribution of concentrated loads into a thin-walled structure, it is highly desirable to test the surfaces while attached to the actual structure, or to the portion affected. In such cases special care should be taken to obtain net deflections of the surface tested. Metal members resting heavily on wood supports will crush the wood and produce erroneous deflection readings, hence are to be avoided.

2. Loading schedule. A loading schedule should be prepared, showing the load distribution to be used and giving the values of the loads to be applied at each stage of the loading process. When the load is to be applied by means of bags of shot or by weights, it is desirable to weigh each increment of loading in advance and to assign it to a marked space on the floor, so that no confusion will result. The loads can be divided into suitable increments of about one sixth (16.7%) of the required

"ultimate" load. In the usual case, such increments will be one quarter (25%) of the required "limit" load, so that the "proof" test load will have been reached at the fourth increment. The "ultimate" load for a "strength" test will then be reached at the sixth increment. After reaching the "ultimate" load, the size of the increments should be reduced so that the second additional increment will produce 115% of the "ultimate" load. However, if the structure should show signs of failing at any time the loading increments should be accordingly reduced.

3. Supports. It is always advisable to support the unit being tested by means of jacks while the loads are being laid on. A safety framework or blocking should be provided in all cases so that the structure will not move far after failure. This not only protects workmen and observers but also permits an accurate determination of the point of initial failure and may permit continuation of the test after local reinforcement if that seems desirable.

4. Deflection sticks should be attached at various points of the test specimen and a level should be provided for reading the scales, which should preferably be graduated in tenths of an inch.

5. Procedure in "proof" tests. It is advisable to apply at least a part of one increment and remove it again before measuring the initial deflections. After each increment is applied, the jacks should be lowered for a period of at least 1 minute before readings are taken. When the total "yield" (see CAR 05.121) load has been loaded on to the structure and readings have been obtained, the entire load should be removed, preferably one increment at a time. The deflection readings at zero load should then be obtained.

6. Procedure in "strength" tests. The procedure outlined for the "proof" test, in the preceding paragraph should be followed exactly until the "proof" test is completed. Note that the "strength" test, if conducted first, would not permit the determination of the permanent set caused by the "limit" load. After the "proof" test the loading should be continued beyond the "limit" load in accordance with the loading schedule. As the "ultimate" load is approached, the structure should be carefully observed and any unusual behavior noted. The increments should be reduced if any signs of failure are observed. If the structure should fail locally before reaching the "ultimate" load, it is permissible to reinforce the failed portion when that is possible and to resume the test. The details of

such reinforcement, when used, should be carefully noted. If it is obvious that material correction factors will be small, it is not necessary to proceed to the 115% overload. If the test speciment cannot be used after test, it is desirable to continue the test to destruction, that is, to the point at which no further loads can be held by the structure. When failure begins during the lowering of the jacks, it is advisable to remove some of the load before completely removing the support, in order that the minimum load causing failure can be determined as closely as possible.

7. Procedure in "strength" tests in lieu of analysis. In such cases the procedure is the same as for "strength" tests except that it will often be necessary to prove that specified extra loads (higher factors of safety) can be carried by certain portions of the structure. Additional design conditions may also have to be investigated. Whenever possible, adequate photographs should be taken and samples of the material secured and tested.

8. Check of test structure. The representative of the Authority will check the test specimen against copies of the drawings submitted to the Authority for approval. The representative will note the numbers of the drawings to which the structure corresponds. In the case of "strength" tests in lieu of stress analysis a very thorough check of the structure to be tested will be made against the drawings. After the test, the portions which failed will be further checked for dimensions and strength properties.

04.34303 MANUFACTURER'S TEST REPORT

1. Preparation. In all cases the manufacturer making a test is required to prepare a complete report.

2. Completeness of report. In general a report, to be complete, should include:

a. Adequate photographs of the test set-up and the part under test, unless sufficiently clear drawings and explanations are shown.
b. Photographs of failed parts or other points of special interest.
c. Records of deflections and other observations and readings taken. Such data are frequently best presented in the form of curves, but in any case the actual readings should be included in tabular form.

d. Date of test, identification number of report, serial and model number of glider and signatures of responsible manufacturer's personnel.
e. All information necessary for the particular test involved.
f. Any other observations which the representative of the Authority feels to be necessary in order to furnish a complete record.

3. In addition to the above items, the following points should be covered in test reports:

a. Critical conditions. The test report should contain computations or references to substantiate the choice of the test conditions and the loadings required.
b. Loading schedule. The loading schedule used in the test should be included in the report.
c. Set-up. A complete description of the test set-up should appear in the report, accompanied by photographs or sketches. References should be made to drawing numbers for the parts being tested.
d. Procedure. A chronological account of the procedure should be presented, giving details and noting all items of an unusual character. The behavior of the structure at various loading stages is of considerable interest, particularly in the case of stressed-skin construction.
e. Test data. All deflection readings should be included, preferably in tabular form. Deflections of the jig should be included.
f. Corrections to standard. When such corrections are necessary, the data for the specimen tests should be included in the report.
g. Disposal of tested structure. A statement should be included as to the disposition of the test specimen, that is, whether it is intended to be used in the glider assembly.

05.3431 WING TESTS

05.34310 RIB TESTS

1. Selection of Ribs.

a. Wing of uniform chord. At least two ribs should be tested for each of the two loading conditions. This applies to a straight wing in which the ribs are all identical, except at the extreme tip and except possibly at the strut points, and means two identical standard ribs.

b. Tapered wing. For tapered wings the selection of test ribs requires the exercise of judgement to insure that the tests made are sufficient to prove the strength of all ribs. It is usual for a manufacturer, however, to use the same member sizes and truss arrangement throughout a series of ribs of varying size. In those cases the largest rib of each series should be used for test.

c. Most tapered wings will incorporate either two or three similar series of ribs. If two are used, tests should be made on two each of the largest rib in each series (a total of four ribs). If there are three series, it will probably be best to test two identical ribs of the largest size used and one each of the largest in each of the other series.

d. In general, no more than a total of four ribs need be tested.

2. Test Loadings.

a. The rib tests required shall 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 air 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.

b. The leading edge portion of the rib may be very severely loaded in conditions II and IV. An investigation of the maximum down loads on this portion should be made when V_g exceeds 125 mph. (See CAAM 05.217-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.

c. The following loadings are acceptable for two-spar construction when the rib forms a complete truss between the leading and trailing edges. (See CAAM 05.217-B1 for other cases.)

(1) For the high angle of attack condition ribs having a chord length of 60 inches or greater should be subjected to 16 equal loads so arranged as to be applied at 1.0, 3.0, 5.0, 7.3, 9.9, 12.9. 16.2, 19.9, 24.1, 28.9, 34.2, 40.4, 47.5, 56.5, 72.0 and 90 per cent of the chord. The sum of these loads should equal the total load carried by the rib, computed as specified above. 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.

(2) For the medium angle of attack condition 16 equal loads should be used on ribs of chord of 60 inches or greater, 8 equal loads for chords less than 60 inches. In either case the total load should be computed as specified above. When 16 loads are used, they should be applied at 9.34, 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.

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

e. When the covering is to be attached separately to the two chords of the rib, the loading specified in paragraph c 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.

f. For ribs attached directly to the spar of single spar wings, the above load distribution may be used.

3. <u>Test Methods.</u>

a. Standard Procedure.

(1) The ribs should be attached to short spar sections by the method used in the actual glider. The spar sections should be supported in such a manner that

they will not prevent free deflection of the rib. It is satisfactory to mount the spars so that their edges rest directly on the supporting structure but they must not be restrained from rolling or twisting.

(2) Any type of testing apparatus or method which applies the loads correctly is acceptable.

(3) To simulate the lateral bracing effect given a rib in the actual wing assembly, it is permissible to employ vertical guide blocks along the sides of ribs which are tested singly. These guide blocks should leave the ribs free to deflect in the direction in which the load is being applied, should have faces bearing against the rib which are not wider than one half inch, and, for metal covered wings, should be spaced at least eight inches apart. For fabric covered wings these lateral supports should not be closer than twice the stitch spacing, or the length of the individual rib chord members, or eight inches, whichever is the greater. In any case the lateral supports should simulate, as closely as practicable, the actual conditions represented in the glider.

(4) In order to avoid local failures of a type not likely to be encountered in flight, it is permissible to use small blocks not more than one inch long to distribute the load at the loading points.

05.34311 WING TORSION TEST

1. Test Methods.

a. Wing Mounted on Jig.

(1) Set-up. The wing is mounted on a heavy timber or steel jig by means of the main wing fittings in such a way that the chord is vertical. A beam is clamped to the wing near the wing tip so that a torque may be delivered to the wing. Figure .3-8 (A) shows the nature of this set-up. When the set-up is not made close to a wall, a framework should be erected along the span and close enough to the wing to act as a reference in determining deflections. This framework must be rigid and secure to prevent movement or displacement. The length of the beam, measured from the chord of the wing to the point of attachment of the load-

ing platform, (distance L in figure), should be approximately equal to the mean aerodynamic chord of the wing. The wing should be uncovered.

(2) Loading. The load on the platform should be increased in suitable increments until the wing is twisted an amount sufficient to accurately determine a definite curve as discussed below. The loading should be carried to the point where the wing shows evidence of appreciable stress, but never far enough to injure the wing.

(3) Deflection measurements. The amount of rotation of the wing should be determined by taking deflection readings at two points along the chord for several locations reasonably spaced along the span. (See Figure .3-8(A)). The distance between points on the chord should be far enough apart to indicate appreciable deflections for the most rigid of wings.

b. Wing Mounted on Glider.

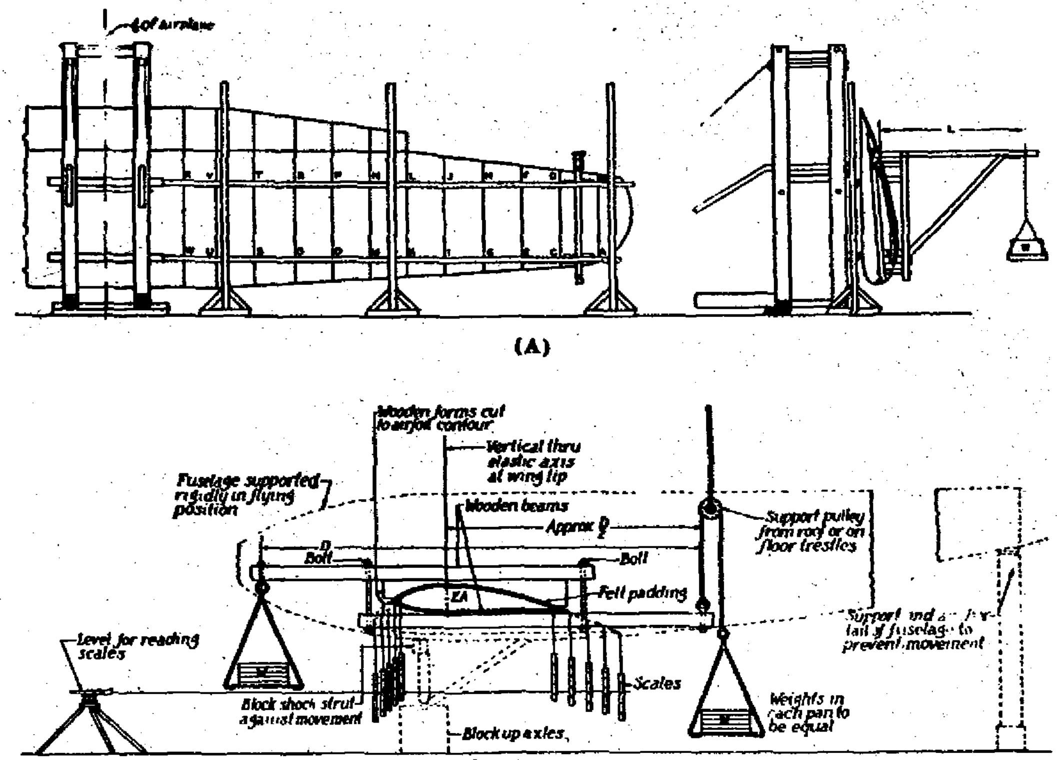
(1) Set-up. When the wing is mounted on a glider the set-up shown in Figure .3-8(B) should be used. In this case it is necessary to block the wheels and shock struts in order to insure rigidity of supports. The wing should be uncovered.

(2) Loading. The procedure in testing is similar to that already outlined. However, two platforms are used, being equally loaded. This results in the application of a torque load only to the wing.

(3) Deflection measurements. Deflections are taken at desired points as before. A "Wye" level and suspended scales provide an easy and accurate means of obtaining deflections. A zero reading for each point is taken prior to application of any load, and deflection readings are taken at the same points when loaded.

c. Test Report.

(1) Recorded data. The data to be recorded are: The loads 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.



SET-UP FOR TORSIONAL TEST OF WING

FIG. 3-8

(2) Interpretation of results. Having obtained the leading and trailing edge deflections or a corresponding set of data, the angle of twist of each section of the wing for a given torque, or platform load, is calculated and plotted against the distance from the wing tip.

θ = angle of twist in degrees

$$= \tan^{-1} \frac{\text{(Leading edge deflect. + trailing edge deflect.)}}{\text{Chord}}$$

or

$$\theta = \frac{57.3}{\text{Chord}} \text{ (Leading edge deflect. + trailing edge deflect.)}$$

Plotting the angle of twist will reveal any inaccuracies in the data and facilitate checking results.

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

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

where

C_{TR} = coefficient of torsional rigidity, lb. in.^{2}

$d\theta$ = angle of twist in degrees in length dL (in inches) caused by torque of M in. lb.

C_{TR} is computed for several points along the span and plotted against the distance from the tip. This curve shows the variation of torsional rigidity of the wing throughout its span, and will be used for purposes of comparison with wings similarly tested.

05.34312 WING PROOF AND STRENGTH TESTS

1. Test Loads. The loads to be used, and their distribution over the wing will depend upon the particular condition being tested for. In general, when the tests are made to prove the

strength of the entire wing, there will be four tests made corresponding to the four basic flying conditions of positive high angle of attack, negative high angle of attack, positive low angle of attack, and negative low angle of attack. In some cases, particularly for most cantilever wings, it will be possible to omit the least critical of the negative conditions and thereby reduce the number of test conditions to three.

2. Test Methods.

a. General. CAAM 05.34302 outlines the general structural test procedure, which applies also to wing tests.

b. Mounting. The wing will almost always be mounted on a jig and care must be taken to insure that the method of attachment duplicates the method used in the actual glider.

c. Chord component. The chord of the wing will not be horizontal, except in some cases, but will be inclined so the load laid on the wing will also load the drag system. The angle of inclination will be such as to produce the correct chord component as determined from the stress analysis.

d. Deflections. Numerous deflection measurements should be taken along the span either at the leading and trailing edges or at the front and rear spars. The points of support should be observed also to see whether or not they move under load. Measurements should be made at each increment of load and these values should later be plotted in curve form in the manufacturer's report, to show the elastic behavior of the wing under load.

3. Rigidity.

a. There are no established criteria for permissible bending deflection in wings.

4. Test Report.

a. In addition to the applicable portions of CAAM 05.34303 the manufacturer's report should include all items peculiar to his particular test, such as computations showing the chord component of the test loads and curves of deflections along the span.

05.3432 CONTROL SURFACE TESTS

05.34320 TAIL SURFACES AND AILERON TESTS

1. Test Loads.

a. Kinds of tests. Tests on tail surfaces and ailerons may be either "proof" tests or "strength" tests. In either case the load over the surface is distributed in the same manner.

b. Load distribution-tail surfaces. The horizontal and vertical tail surfaces are each required to be tested for both of the conditions illustrated in Figures CAR 05-1 and CAR 05-3. The magnitude of the load in each case depends upon which of the specified conditions is critical.

c. Ailerons. Ailerons are tested for the load distribution shown in Figure CAR 05-6.

d. Balance area. When there is no balance portion on the movable surface, the unit loading at the hinge line will be constant over the span. When a balance portion of constant chord is used on a tail surface having a variable chord, the unit loading on the balance portion will not be constant over the span, generally being lower near the tip.

2. Test Methods.

a. Horns. Control surface tests should include the horn or fitting to which the control system is attached. Control surface horns should be held by tubes or straps and never by flexible cables unless the test is purely a control system test because cable stretches excessively.

b. Mounting. The control surfaces may be mounted on the glider as in actual operation provided that cables are eliminated and that the fuselage or wing is either supported rigidly or its movement at the attachment fittings of the control surfaces is accurately measured. If the surfaces are mounted on a jig, the jig should be so constructed as to duplicate exactly the attachment conditions applying in the glider.

c. Fabric covering. When the unit tested is a fabric-covered surface the fabric should be installed as for service except that only the primer coats of dope should be applied, in order to leave the covering slack. The load should be applied directly to the covering.

d. Brace wires. During the test all brace wires must be rigged so that they are at least as tight as they would ordinarily be in service. If any wires become slack during the test that fact should be noted in the report and a statement given concerning the approximate load at which it occurred.

e. Load application. The test load can usually best be applied by means of bags of sand or lead shot, so distributed as to represent the required loading.

f. In the case of a test for the so-called "balancing" condition, note that all the load acts in one direction on the fixed surface.

3. Test Report.

(See CAAM 05.34303.)

05.34321 FLAP TESTS

1. Test Methods and Loads.

a. Wing flaps will almost always be tested on a jig because of the difficulty of mounting a complete wing and flap assembly in inverted position. In all cases, however, the test should include the supporting brackets and their means of attachment to the wing. The test will be similar to that for an aileron or elevator except that the load distribution over the surface will usually be uniform instead of decreasing toward the trailing edge, as is the case for those other surfaces. If it is necessary or advisable to test the flap supports as installed in the wing, that may be done without inverting the wing by running cables from the flap hinges up and over pulleys to a loading platform.

2. Test Report.

(See CAAM 05.34303).

05.3433 CONTROL SYSTEM TESTS

1. Test Loads.

a. Operating test. The controls will be operated from the pilot's seat when the system is sustaining half of the "limit" load specified in CAR 05.23 for the design of the system in question.

b. Test for strength. The test loads for a "proof" test or a "strength" test are derived either from the control surface design loads or from the specified control system design loads. This is explained in CAAM 05.230, in which an alternate loading is described.

2. Test Methods.

a. Arrangement for test. A control system test should be conducted only upon a fully installed system in the actual aircraft. The load may be applied in either of the following ways:

(1) The control system for the main surfaces may be rigidly secured at the normal point of contact with the pilot's hand or foot and the actual surfaces loaded.
(2) The control system for the main surfaces may be secured as in (1) and the load applied to the outer extremities of the system by some other means, such as cables leading over pulleys to a loading platform.
(3) The control system for any adjustment device such as stabilizer, trailing edge tabs or wing flaps is usually required to be self-locking in which cases no additional fixation should be permitted during tests.
(4) Other methods may be proposed but will not be approved unless they lend themselves equally as well as the above to the operation tests.

b. The type of blocking used at the control wheel, the top of the stick, or the rudder pedal should be such that it can readily be removed and put back with the system under load or that it will not interfere with limited movement of the controls. (A compression member for the stick or column to lean against in the neutral position during loading is a good installation because the stick or column can be pulled back for the operation test without disturbing the set-up. For the rudder controls a tension cable or wire, leading forward from one pedal, is sometimes convenient).

c. The test load should be applied, if practicable, in the direction which produces the most critical load in the system. That is, push-pull tubes should be loaded in compression rather than tension and brackets and fittings should be loaded in the worst direction if a probable variation in strength between the two possible directions is indicated.

d. Operating test. The purpose of the operating test is to determine that the controls are free from binding, jamming or excessive friction or deflection at the specified operating test loads.

e. Test for strength. The purpose of a control system test for strength is to ascertain that all those parts of the system which are difficult to analyze, such as brackets, pulleys, and fairlead supports, have sufficient strength and rigidity for service. Careful observation during the test and a thorough inspection of all parts of the system after test are therefore essential. When a control system is carrying the "proof" load no deflection of the brackets or any other part of the system and its supports (except cables) should be apparent upon visual inspection.

3. Rigidity.

a. Extra-flexible cable, as used in control systems, will stretch under load even though of adequate size and prestretched. This is objectionable but does not impair the airworthiness of a system unless the deflection is excessive.

4. Test Report.

(See CAAM 05.34303.)

05.3434 FUSELAGE TESTS

1. Test Loads.

a. The test loads for a fuselage are usually the torsion and bending loads corresponding to the "ultimate" or "limit" loads for the tail surface structure. The horizontal tail surfaces, being symmetrically placed, introduce straight bending loads into the fuselage structure, which are resisted by the wings and the weights forward of the center of gravity. The loads from the vertical tail cause a bending moment, acting sideways, and a twisting or torsional moment, which is resisted by the wings. In landing conditions the fuselage will be loaded by inertia loads and the wheel or skid reaction.

b. It is customary to perform two separate tests, one for each type of loading. For the bending test the loads are so chosen and placed, if possible, as to represent the most severe loading condition for all parts of the fuselage. If a fuselage appears to have a weak top structure it is tested for upward acting tail loads but it is otherwise usual to test for the downward acting loads. For the torsion test there is only one condition to consider, that is, the fin and rudder load.

2. Test Methods.

a. General. CAAM 05.34302 outlines the general structural test procedure, which applies also to fuselage tests.

b. Bending test. For the bending test the fuselage is mounted in a horizontal position and is held in place only by its wing attachment fittings. It is either right side up or upside down depending upon the loading conditions being tested. Tail surface loads are applied through the stabilizer attachment fittings and loads representing weights in the fuselage, if used, are laid inside or hung at their proper locations.

c. Torsion test. For the torsion test the fuselage is mounted on its side, with the longitudinal axis horizontal, and held only by the wing attachment fittings. If the fin is in place on the fuselage, the test load is laid directly on it, distributed so as to locate the center of pressure of the load in its proper place. Otherwise some means of applying the correct torsion, shear and bending loads through the fin attachment fittings must be devised. Unless the fin is of cantilever construction it will be necessary for this test to have both the fin and stabilizer mounted on the fuselage.

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

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

Reference is made to CAR 05.244 which affects the strength of the wing root fittings and surrounding structure against unsymmetrical loads.

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

e. Deflections. Deflection readings should be taken at each increment of load, at several points including the wing attachment fittings (to ascertain whether or not they move), the rear end of the fuselage and, for the torsion test, the tip of the fin.

05.400 MATERIALS AND WORKMANSHIP

1. Materials and processes. 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. 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 to secure 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 a check of margins of safety. Plus tolerances are limited by assembly considerations.

05.4000 MATERIALS OF CONSTRUCTION

1. Wood.

Wood has had general use as a material for glider construction. For the most part those woods used in wood airplanes are used in glider construction.

a. Aircraft spruce is used for spars. This should be selected for grain and strength in accordance with the pertinent Army or Navy specification (see Aircraft Airworthiness Section Report No. 15). Spruce should be used for wood drag compression members.

b. Bass-wood is a satisfactory wood for glider ribs. It has ample strength, excellent gluing qualities and is quite stable with regard to moisture content. Only clear sap-free white basswood should be selected.

c. White pine when of a good quality also makes a good rib material. Spruce, when especially selected for straightness of grain, is also satisfactory.

d. Ash is used for fuselage skis, bent wing-tip parts and wing-tip skids. White ash is best and should be dry and well seasoned.

2. Plywood.

Plywood is extensively used in both wing and fuselage construction of gliders. When used for stressed parts like in nose covering of single or D-spar wings and monocoque fuselages birch or mahogany and spruce should be used. Spar webs and fairing may be made of mahogany plywood. All plywoods should preferably be either resin or blood albumen glued material and should be fresh stock.

3. Steel.

Both alloy and SAE 1025 sheet steel are used for fittings and fastenings on wood constructed gliders. Only good quality rust-free stock should be used and it should be protected from corrosion by zinc or cadmium plating, metallizing, red oxide primer, or equivalent. Steel tubing is replacing wood as a fuselage material to a large extent and it probably affords the best pilot protection due to its ability to absorb considerable energy in failing. Due to the relatively low forces encountered in fuselage design mild steel and seam welded tubing are satisfactory for the most part. Where high stresses and extreme shock absorbing qualities govern the design, however, chrome molybdenum steel (SAE 4130) tubing should be used. Owing to the difficulty of protecting from corrosion, gauges lighter than .028" should not be used.

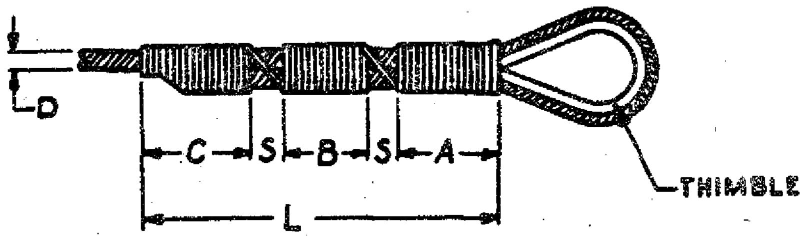
4. Cables and Wires.

Cable and wire where used should be of aircraft quality. Solid wire should be used only in drag systems. Sizes smaller than 16 gauge (.051" diameter) should not be used. The usual ferrule and thimble fastening is satisfactory if soldered with non-corrosive flux and well cleaned and varnished or primed. The wire should not be over heated in the soldering operation. Where used for tail or wing stays, or supports, 19 wire stranded cable with a tinned copper wire wrapped and soldered serving is satisfactory. Only 7 x 7 or 7 x 19 cable should be used in control systems. Where slight bends around pulleys of only 30° or less are encountered, 7 x 7 construction is satisfactory. In all other cases 7 x 19 cable should be used for controls. Unless stressed above 50% of the rated strength, the served and soldered fastening will be satisfactory. See Figures .4-1, .4-2 and .4-3 for cable fittings and splices.

5. Aluminum Alloys.

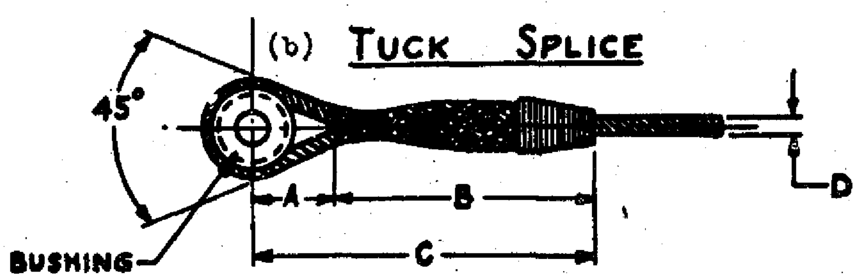
The aluminum alloys have had some use in present day

(a) SERVED AND SOLDERED SPLICE



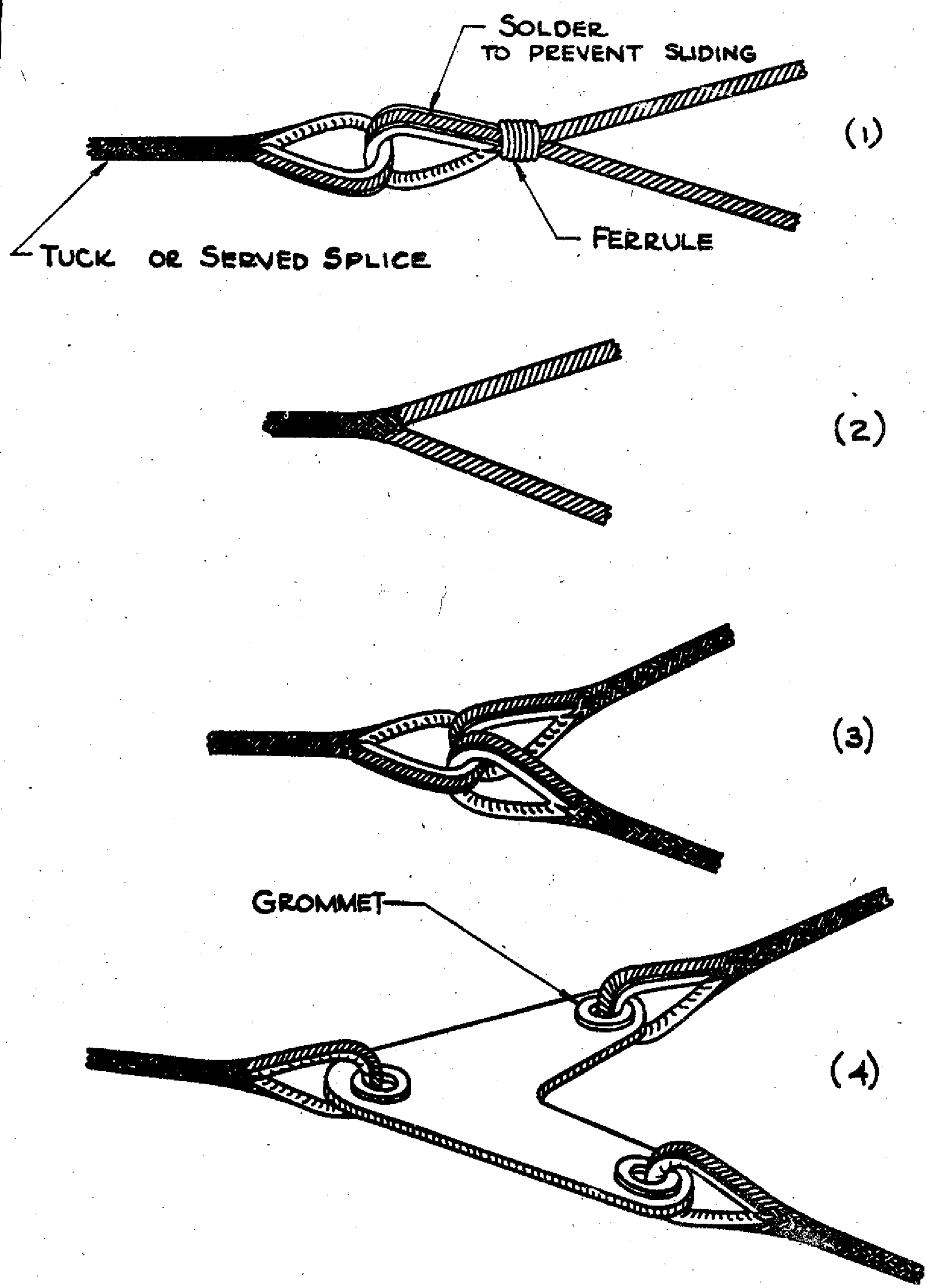
D	L	A	B	C	S	DIA. OF WRAPPING WIRE
1/16	1 3/4	1/2	1/2	1/2	1/8	.035
3/32	2	9/16	9/16	5/8	1/8	.035
1/8	2 1/2	3/4	3/4	3/4	1/8	.035
5/32	3	7/8	7/8	1	1/8	.035

(b) TUCK SPLICE



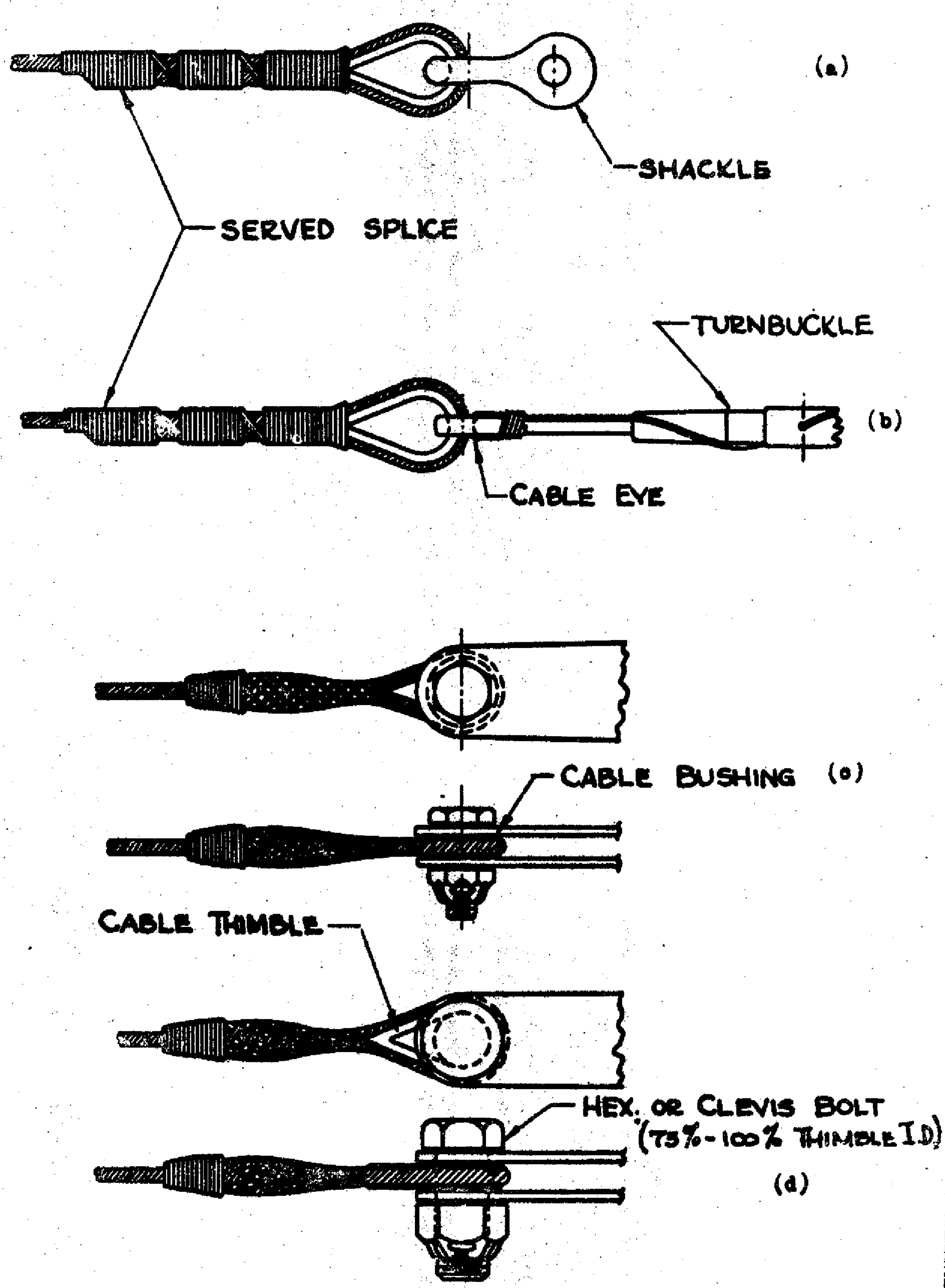
DIMENSIONS AFTER STRETCHING			
DIA. OF CABLE	A	B	C
1/16	9/16 $^{+0}_{-1/8}$	1 1/2 $^{+0}_{-3/8}$	2 1/16 $^{+0}_{-1/2}$
3/32	9/16 $^{+0}_{-1/8}$	1 5/8 $^{+0}_{-3/8}$	2 3/16 $^{+0}_{-1/2}$
1/8	5/8 $^{+0}_{-1/8}$	2 $^{+0}_{-3/8}$	2 5/8 $^{+0}_{-1/2}$
5/32	11/16 $^{+0}_{-1/8}$	2 3/8 $^{+0}_{-3/8}$	3 1/16 $^{+0}_{-1/2}$

STANDARD CABLE SPLICE ENDS FIG. .4-1



CABLE INTERSECTIONS FIG. .4-2

GE27



CABLE END ATTACHMENTS FIG. .4-3

glider construction for fairing and gap covers as well as for primary structures. It should only be worked as a structural material by those trained in its use and then with the same care as applied to airplanes. It is not advisable to use it for fittings in wood or composite construction gliders unless properly protected from corrosion (see CAAM 05.4103).

6. Fabric.

The light weight glider fabrics of about 2 oz. weight are satisfactory for gliders with design gliding speeds up to 70 miles per hour. For higher speed gliders the light airplane fabrics weighing 2.6 oz. per sq. yd. should be used.

05.4001 PROCESSES

1. General.

Wood working tools should be sharp and kept in good condition so as not to injure the wood fibers. In ripping and cross cutting wood parts, sharp fine tooth saws should be used.

2. Splices.

Splices in structural wood members when necessary should have a 12 to 1 slope or greater and surfaces fitted for perfect uniform contact before gluing. The surfaces should be formed with a planer, if possible. The dimensions and type of splice should be similar to those given in CAAM 01.33. Care should be taken in clamping glued splices to use thick "cushion" blocks of the proper slope and size so as to produce clamping action perpendicular to the line of the splice, uniformly distributed, and not such that the pieces tend to slip past each other. A finished splice in wood or plywood should show no change in cross-section at the splice.

3. Glued Joints.

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

4. <u>Nailing Strips</u>.

In applying plywood, nailing strips may be used to produce the gluing pressure so that the nails may be removed after the glue sets. These strips may be thin hard-wood, plywood, or fibre. In cases where a joint in plywood is to be made, a piece of thin aluminum or wax paper under the nailing strip will be found to produce a very clean joint. All plywood joints should be scarfed or beveled at a slope of 12 to 1 or greater so as to leave no ridge at the joint.

5. <u>Jigs</u>.

Wherever several similar shaped parts like ribs are to be glued up from wood pieces, jigs should be used. These may be nailing jigs or may be clamping jigs. Care should be taken, where bent pieces are used, to hold the shape by temporary pieces well beyond the end of the used part of the curve so that there is no tendency for the curve to flatten out when removed from the jig. Where complicated curved parts like wing-tips are to be made and the bend is too severe for a single piece, several laminations may be glued together in a jig.

6. <u>Metal Forming</u>.

a. Steel sheet used for stressed fittings should be handled with special care to avoid scratching or marring the surface in any way. Lines should be drawn with a pencil and not a sharp instrument. All bends should be made around a block which has had the corner rounded off to a radius of at least twice the thickness of the metal (See CAAM 01.33). During the bending operation, sheet should be held in a vice which has the jaws covered with copper, aluminum or brass so as to prevent marring. Hammering should be done through a hard-wood block rather than on the bare metal. Holes may be punched provided they are reamed out at least .010" to get them up to size. When drilling stacked sheets, the burr should be removed from each sheet so that the hole is clean and surface smooth. Where cuts to form corners are made, a 1/8 inch diameter hole should be drilled in the corner and the cut made to the hole, not past it.

b. High strength aluminum alloy sheet must be worked with caution to avoid any marring, scratching, or sharp bends. The strong aluminum alloys, such as 17 ST and 24 ST, should not be bent to a radius less than four times the thickness of the sheet (Reference CAAM 01.33). The bend should be preferably at right angles to the direction of the grain. Bends can be made over soft material such as hardwood.

c. Soft aluminum alloys, 2S, 3S and 4S, are used for formed non-structural parts, and are available in varying degrees of hardness. These alloys can be readily formed and welded. When the forming is severe, the metal may be annealed occasionally to prevent undue strain due to the cold working. This may be done by covering the part with a layer of soot by means of a welding or blow torch, then applying enough heat with a neutral flame to burn off this layer. Riveting may be done with A 17-S rivets when heat treating facilities are not available. Torch welding of aluminum or aluminum alloys is not permissible for structural parts.

7. Welding.

a. Welding is of two types: (a) fusion welding where steel is joined by fusing the two pieces of steel together and filleting with steel rod, and (b) brazing (brass or bronze welding) where a fillet of brass or bronze is used to join two pieces of steel together. True welding should be done only by one who has had considerable instruction and practice. Steel welding is recommended for edge welds and highly stressed fittings, as at wing roots and strut attachments. In either type of welding, the weld should be stressed in shear, never in bending or direct tension.

b. In glider work, brazing will be found to be a satisfactory method of joining thin wall tubes. There is, when it is properly done, no danger of burning or injuring the thin wall tube.

c. Only Oxweld 25 H.S. brass rod or its equivalent should be used, with the proper flux. Anyone with a knowledge of the principles of soft soldering should be able to braze with very little instruction. No attempt should be made to braze joints which have not been abraded to a bright, scale-free finish. This is particularly true of places which have been previously welded. Steel welding must not be done under any condition at or near a place where brass has been used previously. Excessive or long heating of a brazed joint is to be avoided. A white deposit around a brazed joint indicates excessive heat.

d. While tube joints should have a <u>close fit</u> for brazing, a small gap can be filled with brass without impairing the strength of the joint.

8. <u>Protective Finishes (See CAAM 05.4013)</u>.

a. Steel tubing and sheet must be cleaned to bare metal by either sand blast or non-corrosive cleaner and protected by red oxide, aluminize coating, zinc chromate primer, metalizing or other accepted rust-proofing method. Sand blasting should not be used on material equal to or less than .028" in thickness. Wherever primers are likely to come in contact with doped fabric they should be further protected with dope proof paint.

b. Internal aluminum alloy fittings are best finished by anodizing, priming and enameling. Thorough cleaning followed by priming and varnishing may be used. In places where corrosion may be severe, assemblies may be dipped or smeared with a coating of warm mixture of 70% pure beeswax with 30% rust-preventative grease. External surfaces of "Alclad" aluminum alloy or pure aluminum need not be finished. A good coating of wax will offer temporary protection against tarnishing, of the bare metal.

9. <u>Covering and Doping</u>.

The fabric covering of gliders owing to their great area and relatively light construction may prove destructive to primary structure. Fabric tension under excessive doping is very high and great care must be used to brace against it.

a. Fuselages. In covering steel tube fuselages, those tubes to which the fabric must be secured should be wrapped with gauze bandage and given a coat of clear dope. This dope should dry hard before the fabric is applied. The fabric can then be cemented on by an application of another coat of dope and brushing the fabric down in place.

b. Wood-fabric joints. Wood parts should be protected with "lionoil", thinned spar varnish or equivalent. Where fabric is to be cemented to plywood or wood parts the varnish coat should be omitted entirely. Such surfaces should have a heavy coat of dope and allowed to dry hard before covering. Rib cap-strips should be especially carefully coated so that the fabric will be well fastened to the ribs. Where fabric ends on plywood and cementing is depended on for fastening, there should be at least 1" contact on cement joint. The edge of the fabric should be covered with 1-1/2" pinked tape with half the tape on the fabric and half on the plywood. Only glider fabric, light airplane cloth, and balloon cloths can be cemented to wood parts as they are thin and porous enough for the dope to penetrate. Grade A fabric is not suitable for this type of attachment.

c. Wings. Long unsupported ribs may require bracing to prevent them from turning over under the fabric. This can be accomplished by cross-bracing with fabric tape cemented to the rib cap-strips. The ribs should be ribstitched with a good linen thread on about 3 inch centers (See CAAM 01.33 for recommended sewing practices). It is also advisable to cement the fabric to the ribs.

All places where two pieces of fabric are sewed together should be covered with at least 1-1/2" pinked tape after the first coat of dope. Where sanding is resorted to to obtain smooth finishes care should be used not to sand over rib corners or other places where cutting through is likely to result. Places on doped surfaces subjected to trailer or ground wear should have an extra patch of grade A fabric and several extra coats of dope. All fabric covered surfaces should be provided with drain grommets.

After the fabric has been applied and the fastening joints are dry it is advisable to go over the whole surface with a moist sponge to remove all wrinkles and even up the undoped fabric tension so that when the dope is applied the dope tension will be uniform in all directions.

d. Doping. The recommended doping practice is to apply one coat of clear dope, then one coat of aluminum, and then the coat of color or pigmented dope. This may be followed by a coat of one part thinner, one part clear dope and one part pigmented dope, and for best results should be applied with a spray gun. It will be found satisfactory to apply all coats of dope to the light fabrics with a spray gun provided a gun is used which will handle the dopes without thinning. In the event grade A fabric is used, it is recommended that the first two coats of dope be brushed on to obtain good impregnation of the dope in the fabric, and that an additional coat of aluminum dope be used.

05.4010 GLUING

1. Gluing may be used except in cases where inferior joints might result or where proper protection from moisture cannot be shown.

2. High grade casein and synthetic resin glues are satisfactory. 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 fabrication of acceptable joints. The recommendations of the glue manufacturer should be strictly followed.

05.4011 TORCH WELDING

Torch welding of primary structural parts should be used only for ferrous materials and for such other materials shown to be suitable therefor. For large diameter tubing of less than .035" wall thickness brazed joints may prove more satisfactory than welded joints.

05.4012 ELECTRIC WELDING

1. Electric arc, spot or seam welding may be used in the primary structure when specifically approved by the Authority for the application involved. Requests for approval of use of electric welding should be accompanied by information as to the extent to which such welding is to be used, drawings of the parts involved, apparatus employed, general methods of

control and inspection and references to test data substantiating the strength and suitability of the welds obtained.

05.4013 PROTECTION

1. All members of the primary structure should be suitably protected against deterioration or loss of strength in service due to corrosion, abrasion, vibration or other causes. This applies particularly to design details and small parts. All exposed wood structural members should be given at least two protective coatings of varnish or approved equivalent. Built-up box spars and similar structures should be protected on the interior by at least one coat of varnish or equivalent and adequate provisions for drainage should be made. Due care should be taken to prevent coating of the gluing surfaces.

2. 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. 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.

3. Two methods of specifying protective coating are in general use. In one, the various operations or code symbols are listed on the pertinent detail or assembly drawing, while in the other method, a specification listing the operation and the numbers or classes of the parts to be so treated is prepared. Data submitted to the Authority need cover only the minimum protection to be employed.

05.4014 INSPECTION

1. Inspection openings of adequate size should be provided for such vital parts of the glider as require periodic inspection.

2. Points most frequently in need of inspection are main fittings, control linkages, cables at sharp bends, and drag wires. Satisfactory inspection of these and other points can only be carried out if the size and location of openings are such as to give adequate accessibility not only for inspection, but also for servicing and replacement. The inspection of drag wires is expedited if, in lieu of other

means, grommets of about 3/8 inch inside diameter have been installed in the fabric, through which a hook may be inserted to check wire tension or possible breakage.

05.402 JOINTS, FITTINGS AND CONNECTING PARTS

1. In each joint of the primary structure the design details should be such as to minimize the possibility of loosening of the joint in service, progressive failure due to stress concentration, and damage caused by normal servicing and field operations. (See CAR Table 05-3 for multiplying factors of safety required.)

2. 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.

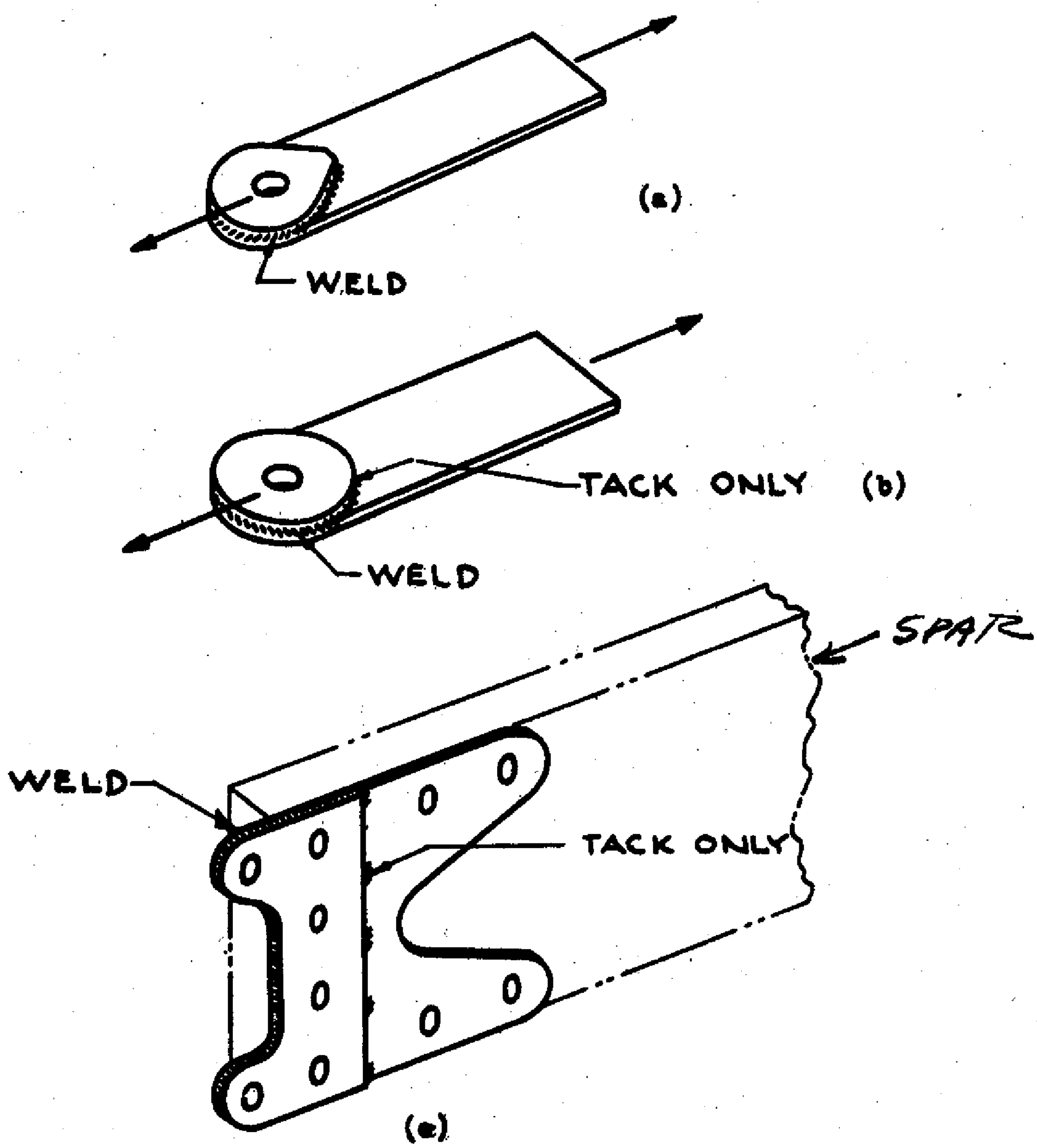
b. Adequate allowance for flexibility of parts being joined.

c. Specifying proper surface condition, i. e., a rough turning job on a highly stressed part invites cracking and failure.

Typical fitting reinforcements are shown in Figure .4-4.

3. Welded fittings subjected to high concentrated loads should be arranged so that the major part of the load is carried through shear welds. Special care should be taken to avoid welds directly around the sections of lugs, etc., which are in bending.

4. Important fittings such as a strut and wing root connection where the bolts (or pins) are often removed should be reamed for a tight fit at assembly to prevent undue wear in service. At some points it may be desirable to design the lugs with enough clearance and margin so that the fittings can be reamed out if necessary to take the next larger



WELDED FITTING REINFORCEMENTS FIG. .4-4

size bolt. It is suggested that in some cases the fittings may be equipped with removable bushings which may be replaced when wear becomes excessive.

5. Tapped fittings should not be used unless they can be replaced in case the threads become damaged or stripped.

6. 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 compact fittings. This sometimes results in a shear failure of the wood along the grain, even though the design load in the tension direction is small. To reduce the possibility of such failures bolt spacings and end margins should be in accordance with Figure 2-4 of ANC-5.

7. When 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.

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

05.4020 BOLTS, PINS AND SCREWS

1. In general, bolts at fitting points should be stressed in shear, not in combined shear and bending.

2. All bolts, pins and screws in the structure should be of uniform material of high quality and of first-class workmanship. Machine screws should not be used in the primary structure unless specifically approved for such use by the Authority. The use of an approved locking device or method is required for all bolts, pins and screws.

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

4. Restrictions on the use of Elastic Stop nuts are as follows:

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

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

c. They should not be used with bolts drilled for cotter pins.

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 Authority.

5. 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 Authority.

05.4021 WOOD SCREWS

The use of wood screws in the primary structure is not advisable except in special cases when the suitability of the particular application must be proved to the satisfaction of the Authority.

05.4022 EYE BOLTS

Special eye bolts and similar special bolts should have a fillet between the head and the shank of at least 1/4 the diameter of the bolt when used in control surfaces or at other locations where they might be subjected to bending or vibration.

05.4023 CASTINGS

1. Castings used in the primary structure should incorporate

the multiplying factor of safety specified in CAR 05, Table 05-3 and should be of such material and design as to insure the maximum degree of reliability and freedom from defects. The Authority reserves the right to prohibit the use of castings where such use is deemed to be unairworthy.

2. 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 Figure .4-5. 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.

3. As with other glider 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 physical tests of test coupons cast with the part are also used. Steel castings with smooth surfaces may be inspected by magnafluxing. X-raying provides a good means of thoroughly inspecting castings if the results are properly interpreted, i. e., by an expert.

05.403 TIE-RODS AND WIRES

1. The minimum size of tie-rod recommended for use in primary structure is No. 6-40. The corresponding minimum size of single-strand hard wire is No. 16 (0.051 inch diameter).

2. 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

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.

(Continued on next page.)

FIGURE .4-5 SUGGESTED CASTING PRACTICE

ALLOY	MINIMUM FILLET RADIUS(3)	MINIMUM SECTION(3) (Webs, etc.)	MAXIMUM RATIO OF ADJACENT SECTIONS(4)	REMARKS(1)(2)
Steel	1/4" (1/2" prefer-red)	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 05, Table 05-3. 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.

FIGURE .4-5 SUGGESTED CASTING PRACTICE (Continued) 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.

3. Hard wire (single strand) should never be used for lift and landing wires because of its unreliable nature. 19 strand cable, 7 x 7, or 7 x 19 construction with ends spliced, or served and soldered should be used instead. The ends should be made over standard cable thimbles or cable bushings, and turnbuckles supplied for adjustment.

05.4030 WIRE TERMINALS

The assumed terminal efficiency of single-strand hard wire should not be greater than 85 per cent.

05.4031 WIRE ANCHORAGES

A fitting attached to a wire or cable up to and including the 3,400-pound size should have at least the rated strength of the wire or cable, and the multiplying factor of safety for fittings (CAR 05, Table 05-3) is not required in such cases. In the case of fittings to which several tie-rods or wires are attached, this recommendation applies separately to each portion of the fitting to which a tie-rod or wire is attached, but does not require simultaneous application of rated wire loads. The end connections of brace wires should be such as to minimize restraint against bending or vibration.

05.4032 COUNTER WIRE SIZES

In a wire-braced structure, the wire sizes should be such that any wire can be rigged to at least 10 per cent of its rated strength without causing any other wire to be loaded to more than 20 per cent of its rated strength. As used here "rated strength" refers to the wire proper, not the terminal. See also CAR 05.274 and CAR 05.275.

05.404 GENERAL FLUTTER PREVENTION MEASURES

1. The Authority reserves the right to require special provisions against flutter in any case when such provisions appear to be necessary.

2. The general principles of flutter prevention should be observed on all gliders. 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, a relatively high degree of mass balance of control surfaces, high frictional damping, and adequate wing fillets and fairing. Features tending to create aerodynamic disturbances, such as sharp leading edges on movable surfaces, should be carefully 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. 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 any improvement in the anti-flutter characteristics of the particular portion of the glider involved and may even be eliminated entirely in some cases. Not all of the previously named aids to flutter prevention are necessary in combination, as the desired result can often be achieved by utilizing only certain features to a sufficient degree.

05.41 DETAIL DESIGN OF WINGS

05.410 GENERAL

1. The more elementary types of gliders logically call for inexpensive wing construction, with simple parts easy to assemble. The two spar wing with single drag-bracing is popular. This is varied by using double drag bracing of wood or wire, which tends to distribute the loads more uniformly between spars, but complicates the analysis unduly unless its effect is (conservatively) neglected. In this case the structural efficiency is not of primary importance, so a solid spar gives economy and simplicity as compared to the built-up types, such as the "box" or "I" beam. This is especially true for the externally-braced straight wings usually employed. When greater structural efficiency is required, as in the intermediate classes of gliders, or advanced trainers, some benefit in weight saving will be obtained by using a built-up single-spar construction. This helps at the wing tips especially, where weight saving is of greatest benefit to lateral maneuverability.

2. Where the wing is straight and spars parallel, assembly is very quick and easy. As soon as a tapered wing is used, with

spars converging, and often bent, the greatest merits of the two-spar type of construction are lost. A tapered wing presupposes design for greater efficiency, therefore, it is of higher aspect ratio and calls for a more efficient beam. The single built-up spar with stressed leading edge cover on the wing is practically universal on advanced designs, both in cantilever and braced wings. Some straight wings also use this type of structure.

3. Such construction requires careful workmanship both in laminating the spar and in attaching the nose cover over the supporting nose-ribs. Special arrangements have been devised for attaching L.E. covers.

4. The single-spar wing lends itself to metal as well as wood construction. The main reasons for not using metal have been the special skill required to work it, and the fact that the structure is usually not highly enough loaded to permit the most efficient use of the material. The minimum gages of material usable from the standpoint of handling, workability, and corrosion resistance tends to exceed that needed to carry the design loads. With a high aspect ratio, thin wing, having a high design wing loading, it is likely that metal construction will be as light and efficient as wood. The labor to assemble would be somewhat greater than for corresponding wood construction.

5. It is necessary on the single spar wing to flare out the structure at the root to provide a reasonably wide base for carrying the drag and torsion loads into the attachment fittings. A secondary member is usually run back from the spar at an angle connecting into the rear root fitting at a chordal point corresponding approximately to the rear spar position in a two spar wing. The nose covering is carried back to this member; to give structural stability, two pins at the root, one forward, and one aft are necessary for an externally braced wing. Full cantilever wings require horizontal pins at the top and bottom of the spar, or one vertical pin full depth at this point, and another pin aft to take out the drag and torsion reactions. Some wings have another pin at the leading edge to increase the rigidity, but this is not necessary in order to get a completely stable structure. It is desirable that the two lower horizontal pins be in line fore and aft so when the third pin or strut is removed the wing will hinge down for ease of assembly or knock-down.

05.4100 TORSIONAL STIFFNESS

1. It is essential that the wing structure have adequate stiff-

ness in order to insure freedom from flutter and other undesirable characteristics. This is particularly important with reference to wing torsional stiffness. Fabric covered wings, in particular, may be critical in this respect. When question as to adequate torsional stiffness arises it is customary to check the deflection characteristics by the application of a torque couple near the wing tip and by measuring the resulting angular deflections along the span. It is then possible to determine the coefficient of torsional rigidity C_{TR} of the wing. A typical test procedure is given in CAAM 05.3.

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

05.411 WING BEAMS

1. Provisions should be made to reinforce wing beams against torsional failure, especially at the point of attachment of lift struts, brace wires and aileron hinge brackets.

05.4110 SOLID WOOD SPARS

Solid wood spars should be made of the best grade spruce. They may be tapered in depth or thickness, but should not be thinner than approximately 1/4 inch at any point.

05.4111 BUILT-UP SPARS

1. Of the built-up types of spars, the box with smooth plywood faces is the most convenient for attaching ribs and has half as many flanges to make as the "I" type. In either type, blocking must be provided at all points where they are fittings attached, etc. Such blocking must be tapered off at the ends to avoid concentration of stress in the flanges. Intermediate verticals are provided in some cases to increase the allowable stress in the webs. These verticals need not be filleted at the ends unless they also carry a concentrated load to be distributed into the spar web. On box spars the attached rib verticals provide this stiffening effect on the webs. CAAM 01.33 shows details of spar construction.

2. Top and bottom spar flanges are usually proportioned according to the relative bending loads in each direction so as to give uniform margins of safety.

3. Spar webs are made of spruce, mahogany, or birch plywood. These are usually of three-ply construction, but special two-ply and 45 degree constructions are available in spruce and mahogany, providing somewhat higher allowable stresses. The face grains of 3-ply should be laid vertically on the spar. Splices should be vertical, preferably over a stiffner.

4. Laminated spars may be spliced in either plane, and splices in the various laminations should be spaced well apart. Splices in solid spars, if any, should be in the vertical plane, as shown in CAAM 01.33.

05.4112 LEADING EDGE

1. Ordinarily three-ply plywood leading edge material is laid with the face-grain spanwise. The best arrangement, however, is to use material having face-plies at 45 degrees, center-ply running spanwise. In any case, the material must be securely fastened at each rib in order to develop its full strength. Splices in the nose covering should be made over a rib. A minimum thickness of 1/16 inch for spruce or mahogany is recommended.

2. On designs where the leading edge covering comes to the spar, the latter can be built full depth of the wing, with a saving of weight over the type which is run under the rib cap strips and then built up flush with spacer strips between ribs. The nose cover serves to gusset the nose ribs to the spar, and an additional gusset with face grains running chordwise should be provided to attach the cantilevered tail ribs. Sufficient glued area must be provided to the face of the spar to carry the vertical shear from the ribs. A smoother covering can be obtained by scalloping the L.E. covering back past the spar between ribs, to keep the fabric from pulling abruptly over the sharp edge of the spar.

05.4113 RIBS

1. Rib spacing under a stressed torsion-nose depends on the thickness of the plywood and the curvature. Usual practice is about six inch spacing with 12 inches for tail ribs. This can be increased where the limit stress is very low, but sufficient ribs should be provided to hold the surface true and prevent "oil cans". Roughly, the maximum spacing for nose ribs should not be greater than the nose rib chord, preferably less.

2. It is recommended that a leading edge spanwise strip be provided under the cover for support and for drag load in the

single-spar wing. The greatest cross-sectional dimension of the strip should be fore and aft.

3. Nose ribs should be stiff enough to allow the nose covering to be clamped down hard for gluing. The lightest practical method of getting the required stiffness is to build the rib up out of approximately square material and carry the joint gussets the full length of the cap strips so as to increase the bending stiffness of the caps between supports. Nose-rib caps must be heavier in order to avoid splitting when the covering is to be nailed on than when the gluing work is clamped only.

4. End ribs and corner ribs should be braced or specially designed to provide stiffness against fabric tension loads. A double set of ribs two or three inches apart and connected with a plywood cap top and bottom is effective, or a rib with members four or five times normal width will usually provide the necessary stiffness.

05.4114 TRAILING EDGES

Typical laminated trailing edges are shown in Figure .4-6. Bent up metal trailing edges are also used, but are more difficult to attach satisfactorily to the ribs. In any case, trailing edges must be husky enough to withstand considerable rough handling when setting up the glider.

05.4115 WING TIP BOWS

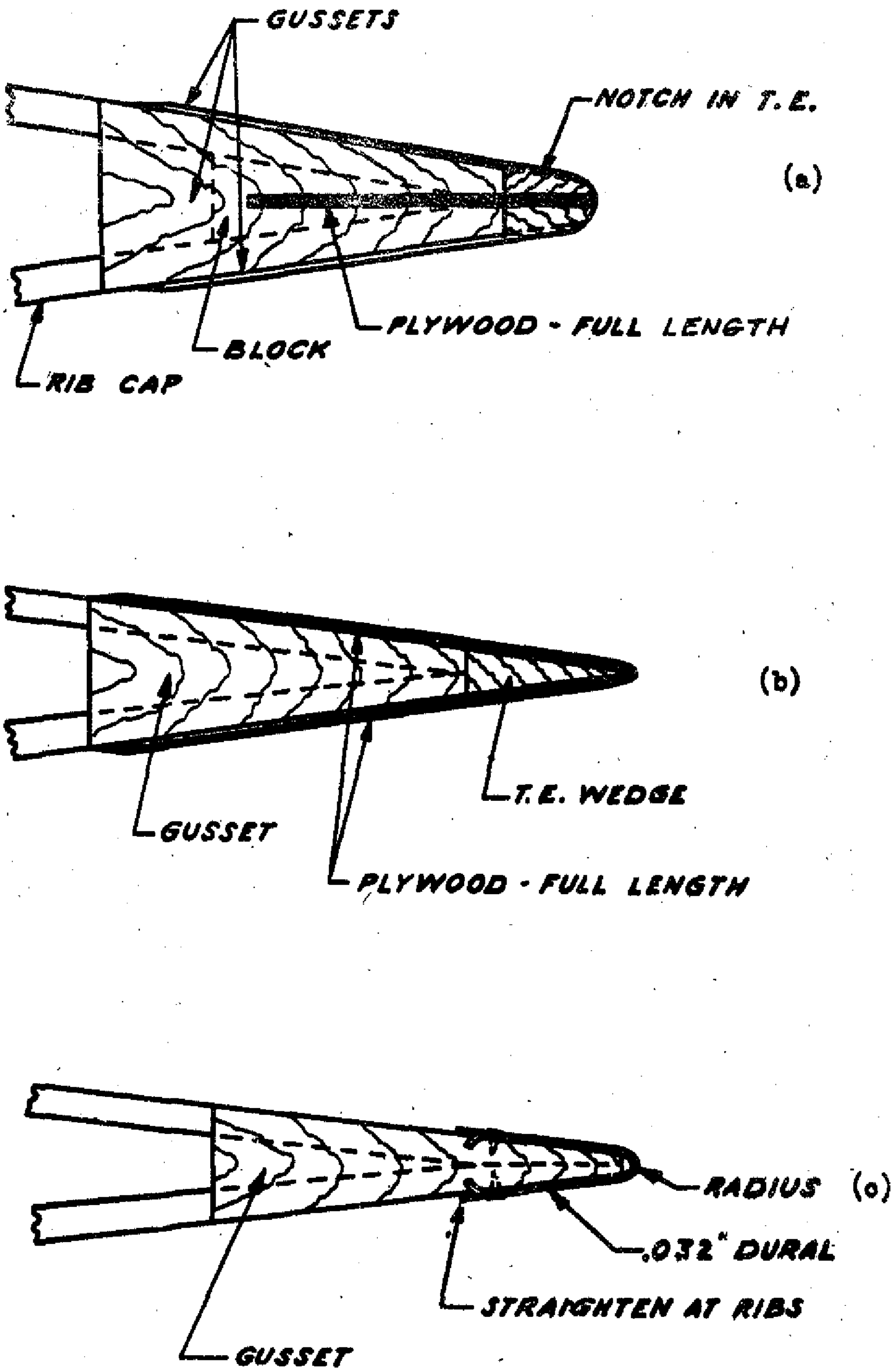
Wing tip bows are frequently made of wood, laminated, or steel tubing attached by welded clips bolted to the spars. The lower surface of tips should be covered with metal or plywood for protection.

05.4120 STRUTS

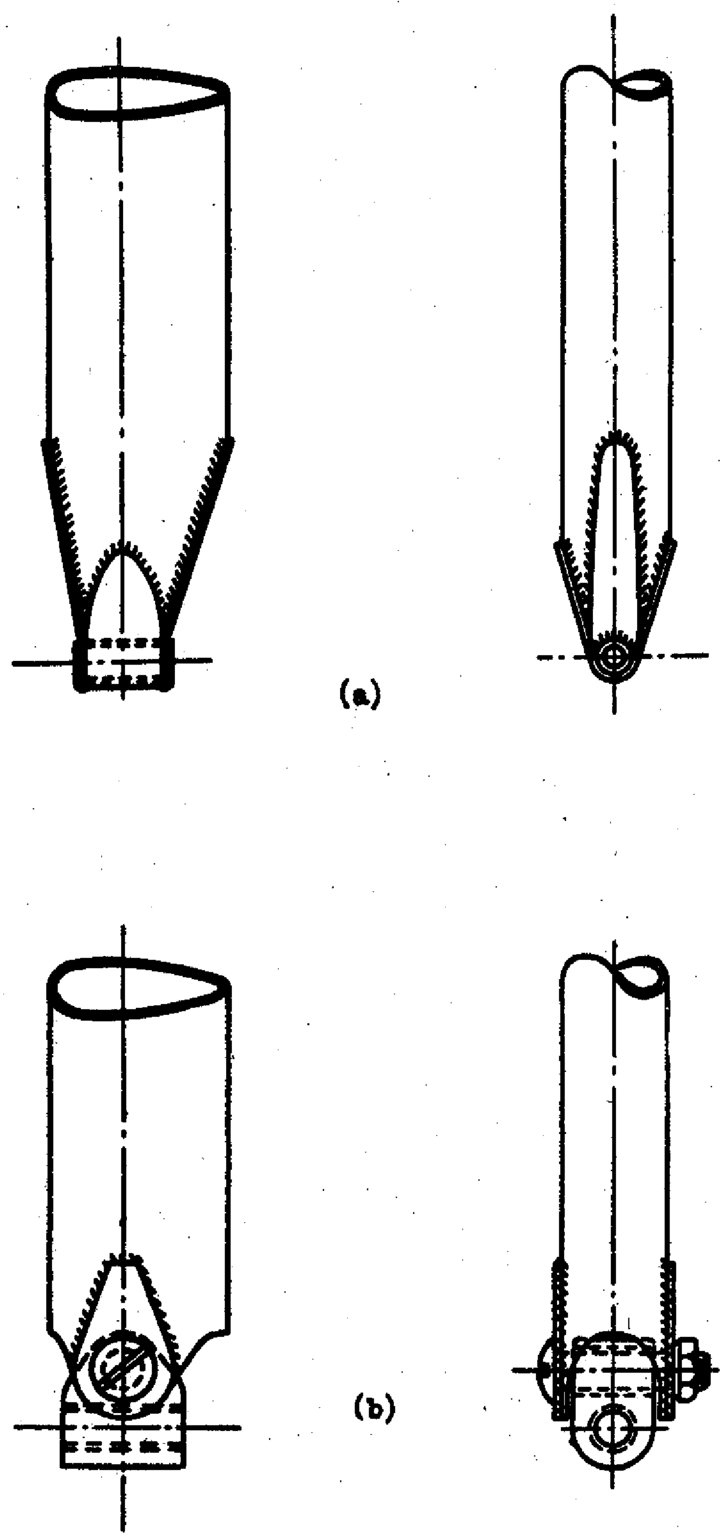
1. Struts may be wood, or metal tubing, either round with fairing or steamline in shape. Either steel or aluminum alloy is frequently used.

2. The wooden struts are usually tapered and are made up solid or of square or rectangular section with plywood fairing. Wood struts must be carefully designed to carry the tension loads from the metal end fittings into the wood.

3. Steel struts have the necessary end fittings welded on so that the load is carried through the weld in shear, while with aluminum alloy the fittings are riveted or bolted into the end



TRAILING EDGE CONSTRUCTIONS FIG. 4-



TYPICAL STRUT ENDS FIG. .4-7

of the strut. A typical end-fitting for a streamline steel strut is shown in Figure .4-7. For a single strut, it is desirable to provide universal end fittings similar to Figure .4-7.

4. Care should be taken in the design of the attachment of struts and wires to avoid eccentric loads tending to roll the spar or bend it in the weak direction. This can be avoided on two-spar wings by providing a deep drag strut near the strut point capable of stabilizing the spars. On single-spar wings it is desirable to keep the strut axis under the spar axis, but in case there is a fore and aft component of load at the attachment point, a nose rib should be provided to distribute this into the nose covering.

5. In connection with single spar braced wings employing fairly high cambered airfoils (which have a large C P travel), it is wise to provide for some freedom at the strut attachment to allow for flexing of the wing without straining or introducing dangerous secondary loads into the strut. A serious secondary load could be imposed on the strut when flying at a low angle of attack into a "down" gust unless the wing is exceptionally rigid or freedom to flex is provided by means similar to a universal joint. Ball socket joints are not recommended.

05.4121 JURY STRUTS

When clamps are used for the attachment of jury struts to lift struts, the design should be such as to prevent misalignment or local crushing of the lift strut.

05.415 WIRE BRACING

1. External bracing. When streamline wires are used for external lift bracing they should be double. (See CAR 05.273).

2. Wire-braced gliders. If glider wings are externally braced by wires only, the right and left sides of the bracing should be independent of each other so that an unsymmetrical load from one side will not be carried through the opposite wires before being counteracted, unless the design complies with the following conditions:

a. The minimum true angle between any external brace wire and a spar is 14°.

b. The counter (landing) wires are designed to remain in tension at least up to the limit load.

c. The landing and flying wires are double.

3. Drag truss. Fabric-covered wing structures having a cantilever length of overhang such that the ratio of span of overhang to chord at root of overhang is greater than 1.75 should have a double system of internal drag trussing spaced as far apart as possible, or other means of providing equivalent torsional stiffness. In the former case counter wires should be of the same size as the drag wires (See also CAR 05.275.)

4. Multiple-strand cable should not be used in drag trusses unless such use is substantiated to the satisfaction of the Authority.

05.414 WING FITTINGS

1. In designing wing-root fittings, care should be taken to box or brace the extending flat ears at the attachment bolts or pins so that the drag loads will not induce appreciable bending stresses in the ears. This also applies to fittings at strut or wire attachment fittings that may have side load components as well as compression and tension loads. This can usually be accomplished by welding a shear plate on to form a three-sided box, or by bracing with an external web on one or more sides of the fitting. If clevis pins are used in place of bolts to connect the two parts of a fitting, the fitting should be designed accordingly. Particular care should be made to provide the extra rigidity in fittings that would normally be provided by the clamping action of bolts and nuts. It should be pointed out that the load distribution in a particular fitting may differ when the attaching bolts and nuts are replaced by clevis pins and safety pins.

2. Bolts for attaching fittings to spars are passed directly through the spar, or through suitable bushings to increase the bearing area in the wood. Care should be taken not to weaken the spar by too close a spacing of bolts, or reduction of the effective section moment of inertia below the critical value. If necessary, the spar can be padded out locally by laminating to increase the spar width. Ash or maple is sometimes used for this when the stress is high as it gives greater bearing strength under the bolts. Unless ash or maple is used under fittings, soft wood should always be built up locally with a layer of birch plywood to prevent crushing under the fittings and bolts,

and to prevent splitting of the spar.

3. Care should be taken to have all bolts in wood spars stressed only axially and/or in shear, wherever possible, as bending on a bolt is more likely to split the wood. Bolts passing through wood should always be provided with the large bearing washers where there is no fitting to serve this purpose. For grouped bolts, it is desirable to provide a single plate on the back side to distribute the load over a large area, rather than to provide separate washers for each bolt. Examples of good and bad fitting designs are given in Figure .4-8.

05.415 FABRIC COVERING

1. The fabric (see CAAM 05.4000-6) should be well finished with dope to provide a smooth surface in keeping with the desired high performance.

2. It is well known that a rough surface creates a large increase in skin friction drag. Since the drag of a well designed high performance glider is mostly of this type and form drag is reduced to a minimum, it is important to get the best surface finish possible. Means of obtaining a high finish is dealt with in CAAM 05.4001-9.

3. All handholes through the fabric and holes where controls, etc., enter should be properly reinforced, preferably by light plywood gussets.

4. It is common practice to cover a wing entirely around the leading edge with fabric.

05.416 METAL-COVERED WINGS

1. The detail design of such wings should incorporate suitable provision against buckling or wrinkling of metal covering as specified in CAR 05.201.

2. 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.

05.42 DETAIL DESIGN OF CONTROL SURFACES

1. Trailing edges must be as substantial as for wings, to hold

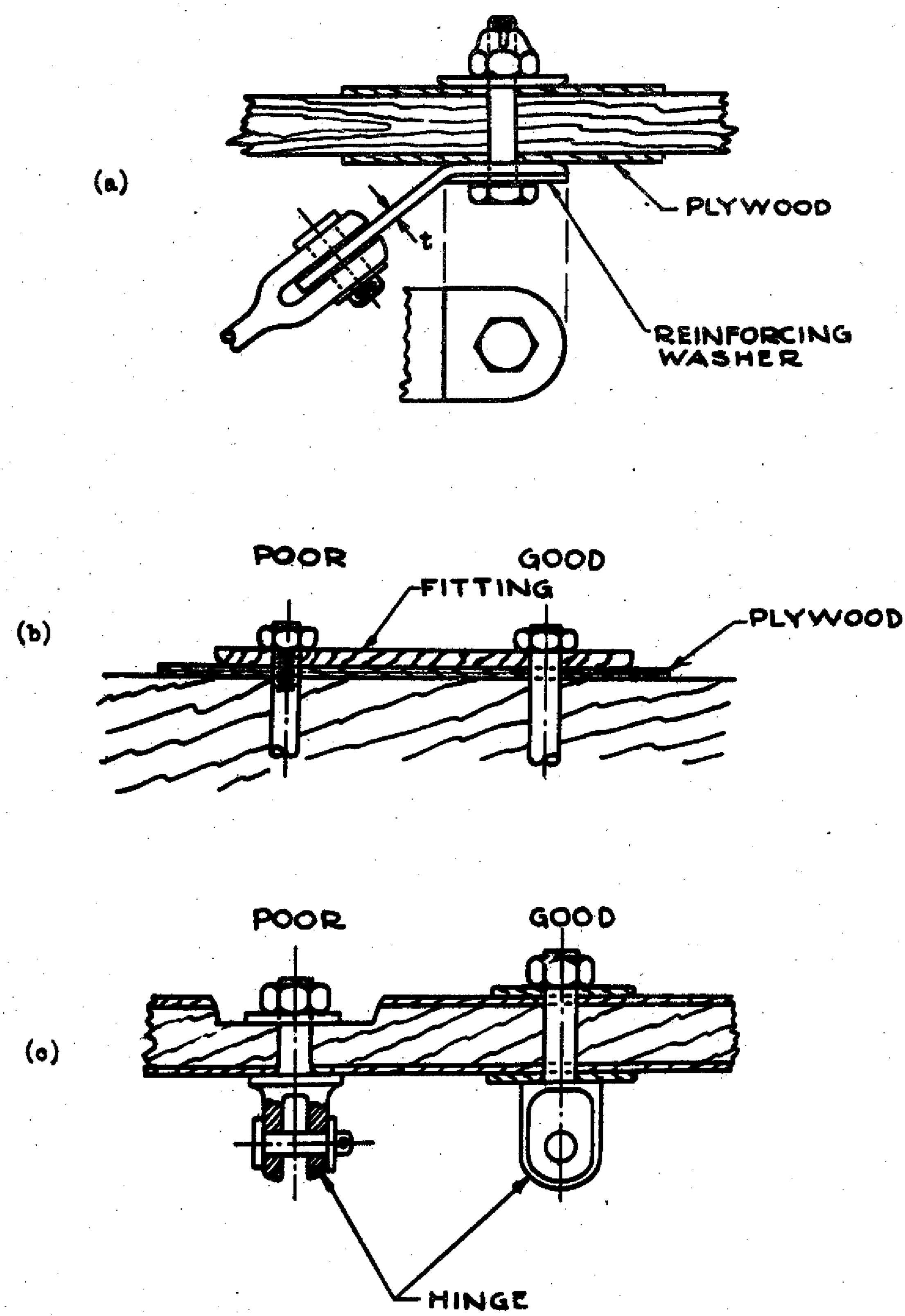

(a)
PLYWOOD
t
REINFORCING WASHER
(b)
POOR
GOOD
FITTING
PLYWOOD
(c)
POOR
GOOD
HINGE

FIG. .4-8

6737

shape under the fabric tension load.

2. The covering of control surfaces should be provided with holes for drainage and "breathing".

3. An improvement in control and reduction in drag can be accomplished by covering the gap between control surfaces. Thin metal or celluloid on the outside of the gap, or fabric inside is frequently used for this purpose. On some designs a circular leading edge on the moving surface fits snugly into the other surface so that the resulting small gap needs no other seal. Care should be taken that the gap covers do not obstruct free drainage, or create undue friction.

4. The stabilizer on experimental designs should be arranged so that the incidence can be revised if necessary after flight tests to get the proper balance.

5. On wings which have no means of adjusting the twist, a small metal tab fastened to the trailing edge of one aileron can be bent as necessary to trim the wing laterally. In extreme cases this may be necessary on the rudder also.

6. If the elevator is unusually low so that it drags in high grass, it will save some repair work to cover it with Grade A fabric instead of with light glider fabric.

7. The leading edge of wood fins and stabilizers is covered with plywood to hold the airfoil contour. Spars may be built-up or solid. Ribs are trussed as in wings unless the surface is thin, when ribs are usually built with solid plywood webs.

8. Tail ribs are trussed in the usual way with diagonals in compression under normal flight. A minimum practical size for caps and diagonals is about 3/16 by 1/4 inch, when using gussets. When the trussing is glued directly to the caps without gussets the material must be wider and thinner to provide sufficient gluing area. All joints should be made as concentric as is practical.

9. It will be found very convenient for construction purposes to design the wing ribs and aileron or flap ribs together so that the two surfaces can be built up as a unit with their respective spars in place and parted after completion. This avoids tedious lining up and independent jigging.

10. Special care should be taken to provide sufficient strength to carry concentrated loads from aileron and flap hinges into the main structure. A separate structure is usually not necessary

to carry these loads, but the ribs at these points should be somewhat oversize with respect to the hinge reactions because of the redundancy between aileron and wing structure. The control horn reactions must also be provided for. It is desirable to have the control horn at a hinge so as to take out the load through the hinge without bending the beam in the control surface. This also eliminates undesirable flexibility in the control system.

11. Control horns are usually constructed of plywood or metal, preferably the latter. They should be attached to the control surface spans near a hinge to avoid bending the spar, and in such a manner to distribute the loads into the spar without tending to split it.

12. Aileron and flap surfaces, as well as tail surfaces are designed to carry the loads from the control horn either by adopting a torsion-tube construction, or using diagonal ribs. The first method is carried out by boxing in the leading edge of the surface with plywood, providing cap strips at the corners of the box for gluing. In the latter method, the surface ribs are laid out in a continuous zig-zag truss from the horn.

13. Long ailerons often have two control horns in order to provide extra torsional stiffness and keep the surface from feeling "rubbery" under load.

14. Although many high-performance ships have continued to use external control surface horns, it is believed that internal horns are a worthwhile attempt to reduce drag.

05.420 INSTALLATION

1. Movable tail surfaces should be so installed that there is no interference between the surfaces or their bracing when any one is held in its extreme position and any other is operated through its full angular movement.

2. 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 CAAM 05.430.

3. Clearances, both linear and angular, should be sufficient to prevent jamming due to deflections or to wedging by foreign objects, particularly safety pins. 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, or when one wing tip is resting on the ground.

4. 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. Leading edges and struts should have adequate strength to withstand handling loads if handles or grips are not provided.

5. 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.

05.421 STOPS

1. Stops are not specifically required <u>at control surfaces</u> except in the case of adjustable stabilizers or elevator trailing edge tab systems. In these cases the stops should be positioned so as to limit the travel to the approved range. However, it is recommended that some form of stop be employed at all surfaces in order to avoid interferences and possible damage to the parts concerned, particularly in the case of large surfaces where the deflections in the control system may permit the surface to exceed the design range of travel.

05.422 HINGES

1. Hinges of the strap type bearing directly on torque tubes are permissible only in the case of steel torque tubes which have a multiplying factor of safety as specified in CAR 05.276. In other cases sleeves of suitable material should be provided for bearing surfaces.

2. Clevis pins may be used as hinge pins provided that they are made of suitable material and properly locked.

3. 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 hinge centerline. 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.

05.423 ELEVATORS

1. When separate elevators are used they should be rigidly interconnected.

2. When dihedral is incorporated in the horizontal tail the universal connection between the elevator sections should be rugged and free from play.

05.424 AILERONS

1. It is suggested that ailerons on gliders having a design gliding speed V_g in excess of 125 miles per hour and ailerons attached to internally-braced wings, to wings braced by wires only, or to wings which in the opinion of the Authority are susceptible to flutter, should be statically balanced about their hinge lines when in the neutral position.

05.425 WING FLAPS

1. Flaps should be so installed as not to induce flutter or

appreciable buffeting.

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.

05.426 TABS

1. Control surface trailing-edge tabs should be statically balanced about their hinge lines, unless an irreversible non-flexible tab control system be used. The installation should be such as to prevent development of any free motion of the tab.

2. When trailing-edge tabs are used to assist in moving the main surface (balancing tabs), care should be taken in proportioning areas and relative movements so that the main surface is not aerodynamically overbalanced at any time.

3. 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.

05.43 CONTROL SYSTEMS

05.430 GENERAL

1. Rigidity. 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 CAAM 05.420-2)

2. Dual controls. 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 as 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.

3. Control system locks. When a device is provided for locking a control surface while the aircraft is on the ground or water, compliance with the following requirements should be shown:

a. The locking device should be so installed as to positively prevent taxiing or taking off, either intentionally or inadvertently, while the lock is engaged.

b. Means should be provided to preclude the possibility of the lock becoming engaged during flights.

4. Welds. Welds should not be employed in control systems to carry tension without reinforcement from rivets or bolts.

05.431 POSITIONING AND LAYOUT

1. The predominating type of cockpit controls is the stick and pedal system. Wheel control is sometimes used for high performance gliders with restricted cockpit size.

2. Travel. It is suggested that the total travel at the top of the stick should be approximately 12 inches or more in both planes to avoid undue sensitivity of the elevators and provide sufficient leverage on the aileron. When a wheel control is used the angular motion should be not less than 60 degrees either side of neutral. Rudder pedals should have about two inches travel either way.

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 flap control and brakes be operable without changing hands on the wheel or stick. Likewise secondary controls should be so located that the possibility of accidental or mistaken operation is remote.

4. Centering. 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.

05.452 STOPS

1. All control systems should be provided with stops which positively limit the range of motion of the control surfaces. Stops should be capable of withstanding the loads corresponding to the design conditions for the control system.

2. 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. 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.

05.453 HINGES, BEARINGS, AND JOINTS

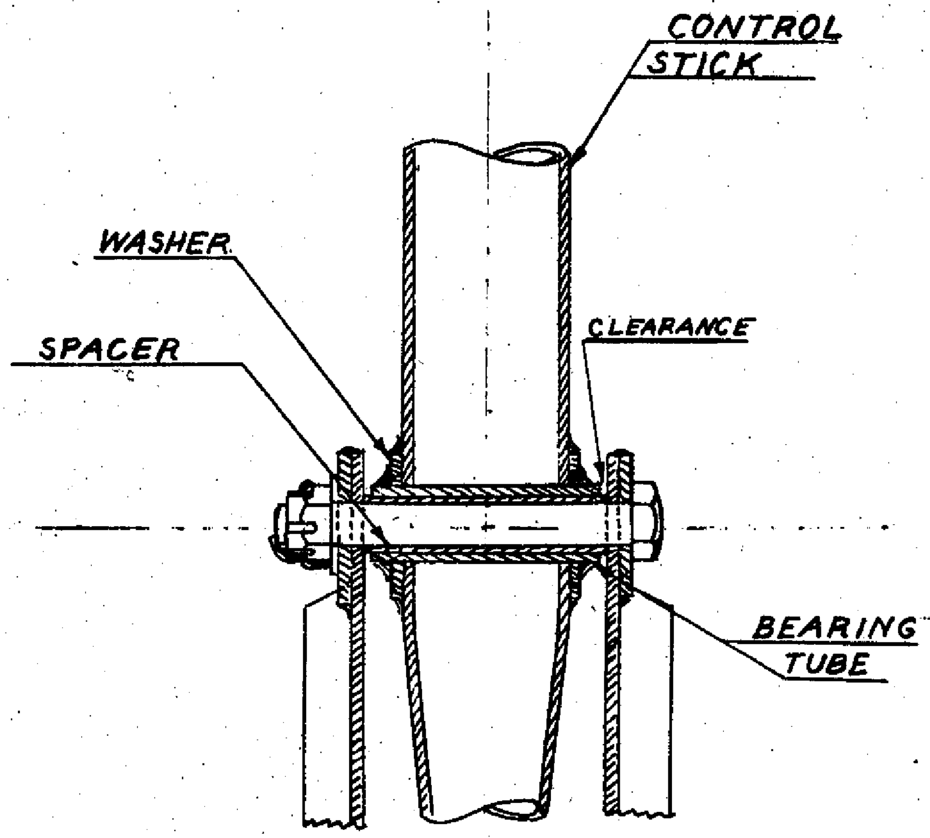
1. Hinges. Standard control surface hinges are available at the supply houses. In general, hinge pins should be 1/8 inch or more in diameter. If clevis pins are used, a washer should be placed under the cotter pin. Standard A-N bolts or clevis bolts are preferable to clevis pins in the rotating joints of hinges or controls.

2. Bearings. Bearings should be arranged so they can be lubricated to prevent undue wear and loosening up of the system.

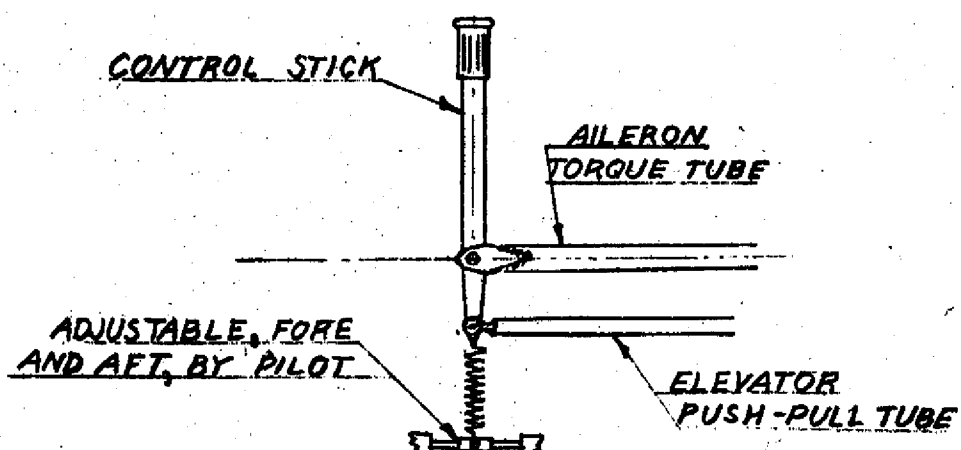
3. Friction. Friction should be especially avoided in the aileron control systems of large high performance gliders as this control tends to be heavy. This may sometimes dictate the use of anti-friction bearings at heavily loaded pivots.

4. Locking devices. Bolts, straight pins, taper pins, studs, and other fastening means should be secured with approved locking devices. The assembly of universal and ball and socket joints should be insured by positive locking means, rather than by springs. Woodruff keys should not be used in tubing unless provision is made against the key dropping through an oversize or worn seat.

5. Cockpit controls. Stick pivots and other similar joints in the control system that tend to wear rapidly should be constructed with a spacer tube on the through bolt to take the wear in the bearing and allow the bolt to be clamped down tight. See Figure .4-9.



STICK BEARING WITH SPACER



ADJUSTABLE ELEVATOR BUNGEE

FIG. 4-9

05.434 CABLES, PULLEYS AND FAIRLEADS

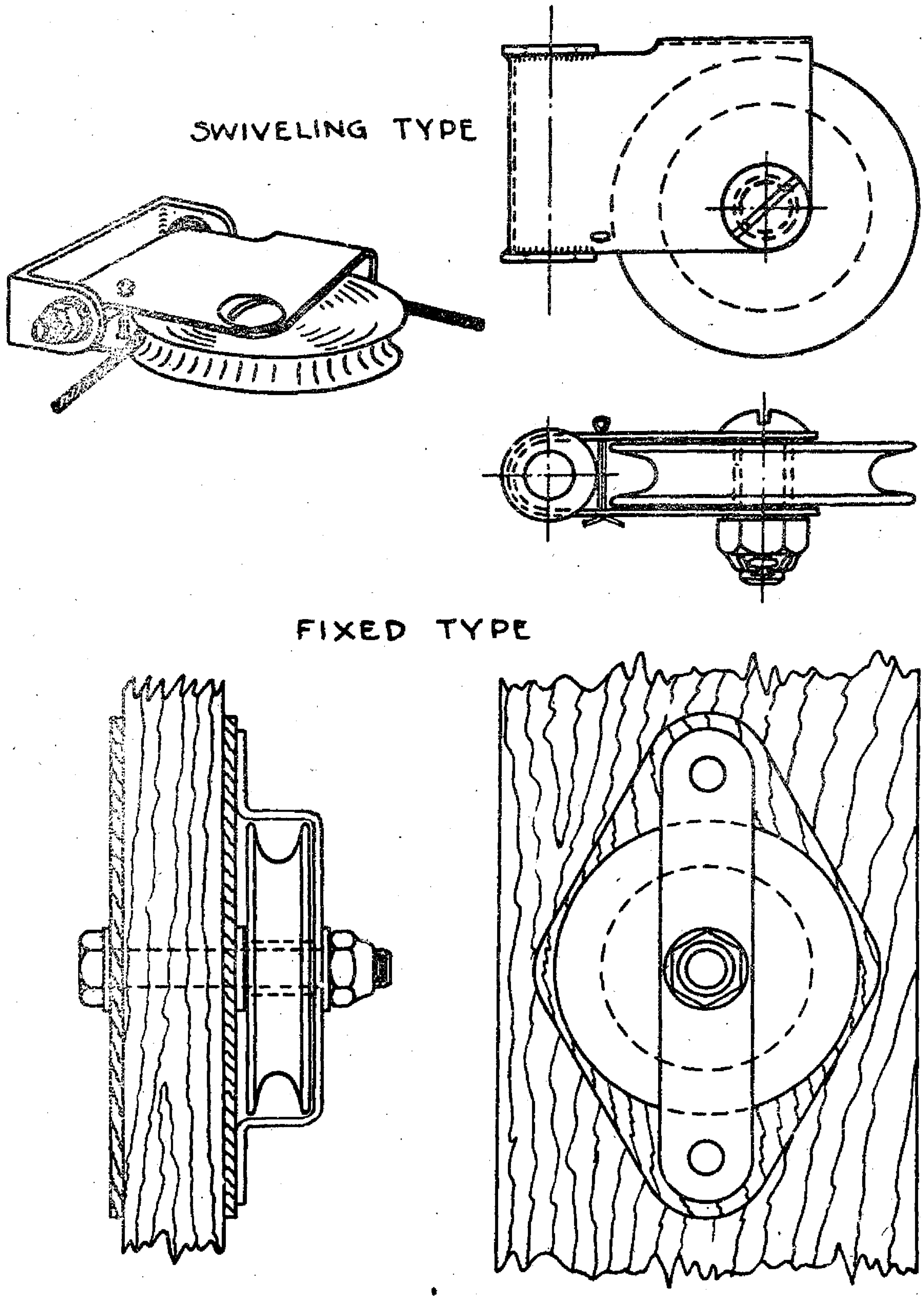
1. Cables. Control cables should be of the 6 x 19 or 7 x 19 extra-flexible type, except that 6 x 7 or 7 x 7 flexible cable is acceptable in the 3/32 inch diameter size and smaller. For properties see Table 4-11 of ANC-5. End splices should be made by an approved tuck method such as that of the Army and Navy, except that standard wrapped and soldered splices are acceptable for cable not over 3/32 inches in diameter. Approved swaged-type terminals are also acceptable. Dimensions for approved splices are given in Figure .4-1. It should be remembered that cable sizes are governed by deflection conditions as well as by strength requirements, particularly when a long cable is used. Some acceptable types of cable joints are given in Figure .4-3.

2. Turnbuckles. Turnbuckles should be located so as to be accessible for adjustment and preferably not in the center of long unsupported spans where they can slap around too freely. Examples of turnbuckle installations are given in Figure .4-10.

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

4. Fairleads. Fairleads of non-metalic material, such as the phenolic plastic compounds should be used to prevent cables, chains and links from chafing or slapping against parts of the glider, 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. A somewhat greater value may be used in secondary control systems.

5. Pulleys. Standard pulleys of the A-N type are lighter than metal pulleys and readily available. Pulleys that carry a 180 degree wire bend should be carefully mounted so that there is no danger of their leaning over and binding under load. They should also line up accurately with the plane of the cables or the flanges will wear out quickly. See Figure .4-11. All pulleys should be provided with annular guards so located that a slack wire cannot get off the pulley in any manner and jam it.



CONTROL-PULLEY BRACKETS FIG. .4-11

05.455 MOTIONS AND CLEARANCES

1. Tension changes. 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.

2. Aerodynamic balancing. 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 over-balance effects.

3. Creeping. 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.

4. Interference. Proper precautions should be taken with respect to control systems to eliminate the possibility of jamming, interference from cargo, passengers or loose objects, and chafing or slapping of cables against parts of the glider. All pulleys should be provided with satisfactory guards. A control column or stick located between a pilot and a passenger should not be used unless a throw-over type of wheel control is incorporated. The cockpit controls should be protected by a flexible boot or similar means, if necessary, to preclude any possibility of any objects becoming fouled in the air controls.

5. Clearance. At the control surfaces themselves ample clearances (5 degrees) should be left beyond the normal deflections, to prevent interferences and damage when the surface is slammed over by wind on the ground.

6. Nose wheel. 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.

05.456 SINGLE CABLE CONTROLS

Single-cable controls are discouraged except in special cases in which their use can be proved to be satisfactory. 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.

05.457 SPRING DEVICES

The use of springs in the control system either as a return mechanism or as an auxiliary mechanism for assisting the pilot (bungee device) is discouraged except under the following conditions:

a. The glider should be satisfactorily maneuverable and controllable and free from flutter under all conditions with and without the use of the spring device.

b. In all cases the spring mechanism should be of a type and design satisfactory to the Authority.

c. Rubber cord should not be used for this purpose.

05.458 FLAP CONTROLS

1. The flap operating mechanism should be such as to prevent sudden inadvertent or automatic opening of the flap at speeds above the design speed for the extended flap conditions. Means should be provided to retain flaps in their fully retracted position and to indicate such position to the pilot.

2. 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.

05.439 TAB CONTROLS

1. Position indicator. When adjustable elevator tabs are used for the purpose of trimming the glider, a tab position indicator should be installed and means should be provided for indicating to the pilot a range of adjustment suitable for safe take-off and the directions of motion of the control for nose-up and nose-down motions of the glider.

2. Reversibility. Tab controls should be irreversible and non-flexible, unless the tab is statically balanced about its hinge line.

3. Wear and vibration. 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.

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

5. Direction of operation. Proper precautions should be taken against the possibility of inadvertent or abrupt tab operation and operation in the wrong direction.

05.45 LANDING GEARS

05.450 GENERAL

1. It is undesirable and unnecessary to mount wheels on an external landing gear structure as is done on power planes for ground clearance. One landing wheel located near the center of gravity (preferably aft) supplemented by a nose skid and either a tail skid or a skid directly aft of the wheel is customary. Some designs dispense entirely with the wheel, and use skids only.

2. A tail-skid may be desirable on some designs to protect the bottom of the fuselage and rudder. If the skid takes much shock on take-off or landing, the spring leaf type is desirable.

05.451 LANDING SKIDS

1. Main skids. Landing skids of ash or similar material are used on most gliders to take and distribute nose-down landing loads. Skids must be sprung by rubber blocking or other means when pneumatic wheels are not provided to absorb the major part of the shock. Even with wheels, the skid is often sprung.

2. Shock absorption for skids. Whether "rubber doughnuts" blocks, tennis balls, or other springing is supplied, there should be two or more points of support besides the front anchorage, capable of taking side as well as direct vertical load. When cleanness is important the skid is faired full length into the fuselage by means of a flexible boot.

3. Skid design. Skids should be reasonably easy to replace especially on gliders not provided with a wheel. A minimum size of 1/2 x 2 inches is recommended. On very course-surfaced airports a metal-faced skid will have longer life.

05.452 LANDING WHEELS

05.4520 GENERAL

Landing wheels are most satisfactory for all varieties of operations if located close underneath the average C.G. of the glider. This enables the pilot to hold the tail either high or low as may be desired. When braking, the nose bears on the ground and helps to slow the glider down as well as to kill the lift on the wings. The attitude of the glider when held over on the nose should not allow the tail to rise higher than necessary to kill most of the wing lift, as it would turn over more easily in a tail wind.

05.4521 SHOCK ABSORPTION

1. The wheel should be located so as to project enough below the forward skid so that practically full tire deflection is available at a speed slightly above the stall. More than this amount is undesirable if the skid is to protect the tire from rough obstructions.

2. When using a small wheel, with limited tire deflection for shock absorption, it is desirable to spring the skid. Large wheels can take all the loads. Wheels and tires need not be of special glider type, unless a wheel or tire failure will prove dangerous to the particular glider in question.

3. Regardless of the type or extent of the shock absorbing qualities of the glider, it is advisable, from service considerations, to design a weak and easily replaceable part into the system, the partial failure of which will not damage the remaining landing gear structure. Such procedure will prevent unduly high stresses from being transmitted to the main fuselage or wing structure, and will greatly simplify repairs.

05.4522 WHEEL SUPPORT STRUCTURE

The wheel support structure should be kept as independent as possible of the lift truss system. Otherwise an incipient or unnoticed failure in the wheel or attachment from a bad landing might cause a failure of the lift truss system in flight.

05.4523 BRAKES

Brakes on landing wheels are necessary to give control over the landing run, such as when landing down wind or down hill. Glider type brakes built into the wheel are available on some size wheels which might be suitable for the heavy two-seaters. For smaller and lighter wheels, a simple brake consisting of a shoe of metal or composition material pressed down on the circumference of the tire is satisfactory. A control wire is carried forward from the brake to a handle in the pilot's cockpit.

05.4524 DOUBLE WHEELS

Although not as common, landing wheels are sometimes used in pairs. If spaced so that the C. G. of the glider cannot fall outside of the wheel base, the glider will not lie on the wing tip as with a single wheel.

05.46 DETAIL DESIGN OF FUSELAGES

05.4600 GENERAL

1. Purpose. The fuselage is designed to carry the pilot, support the tail surfaces and landing gear with as little weight and drag as is consistent with the general purpose of the glider.

2. Nose. The nose of the fuselage should be made sufficiently strong to give the pilot reasonable protection in case of a crash, as well as carry the catapulting and towing loads from the hook.

3. Landing gear. The landing wheel and skid, if any, should be well supported structurally. The former should be boxed around so that snow, mud or sand cannot pack into the interior of the fuselage.

4. Ground angle and clearance. The bottom of the fuselage aft of the wheel should provide sufficient ground angle and clearance so that the wing can be held at a high enough angle of attack for take-off and landing. An angle on the wing of at least 10 or 12 degrees is advisable, part of which can be provided by setting the wing at an angle of incidence on the fuselage.

5. Tail skid. The tail end should be provided with a skid or so arranged that the bottom of the rudder is protected from obstructions when landing.

6. Provision for turn-over. The fuselage and cabins should be designed to protect the passengers and crew in the event of a complete turn-over and adequate provision should be made to permit egress of passengers and crew in such event.

05.4610 PILOT AND PASSENGER COMPARTMENTS

1. Ventilation and visibility. The pilot's compartment should be so constructed as to afford suitable ventilation and adequate vision to the pilot under normal flying conditions. In cabin gliders the windows should be so arranged that they may be readily cleaned or easily opened in flight to provide forward vision for the pilot.

2. Seats. Seats or chairs for passengers should be securely fastened in place in both open and closed gliders, whether or not the safety belt load is transmitted through the seat. (See Part 15 and CAR 05.261 for safety belt requirements.) Pilot seats on gliders licensed for airplane towing should be designed to carry a parachute. A recess at the back for a backpack is the most common means, a deep seat with cushion to replace the pack can also be used, but most gliders are too shallow for this arrangement. It is not desirable to cramp the pilot so tightly in a high performance glider that he is handicapped or fatigued on long flights.

3. Pilot and passenger enclosures. Removable "scoops" around the pilot should be securely attached to carry the air loads encountered at a maximum towed speed, but must be easy to

release and push off in case the pilot has to bail out. They should be so designed that their removal in flight at high speeds will not injure or inconvenience the pilot or passengers, or block the exits. The nose may be built up around the pilot and only a local portion be removable. Sufficient room should be provided for exit wearing a back chute. A clear fore and aft opening of not less than 24 inches is desirable.

05.4620 STEEL TUBE FUSELAGES

1. General. Steel fuselages are built up by gas welding or bronze welding of round or square steel tubing into a rigid truss structure. Sections through the fuselage are usually combinations of rectangular, triangular, and diamond shape. The basic shape is extended up to carry the wing and down for the wheel and landing skid. The portion forward of the wing is frequently cantilevered out under the pilot, the top and side fairing being removable as a whole for exit.

2. Load distribution. Loads from the pilot's seat, belt, wing and strut attachments, wheel and tail surfaces should be properly distributed into the fuselage trussing. Eccentricities and bending of the truss members should be avoided wherever possible. In arranging the truss members considerable stress analysis labor will be saved by eliminating redundant members.

3. Diagonal braces. Diagonal braces are necessary for stability in rectangular or diamond shaped bulkheads having unsymmetrical external loads applied on them. Where one side of a rectangular truss is broken, as for a cockpit opening, the adjacent bulkheads usually require a diagonal to provide torsional stability. This is not necessary when an extra over or under truss serves to carry the shear across the panel. Many gliders have an extra bottom "V" structure which serves to support the skid and wheel under the main structure. It is not necessary that all the bulkheads in a rectangular fuselage have a diagonal, but an occasional diagonal between the tail and rear wing attachment bulkhead will increase the torsional rigidity and provide means for transferring the load if a member is damaged.

4. Joints. Joints should be designed for simplicity with not more than six intersecting members if possible to reduce the likelihood of strain cracks after welding. Joints must not be butted square unless there is no possibility of tension or bending in the member. On members designed by tension rather

than by column loads, it will be beneficial to make the end joints at a flat angle to the tube axis so as to develop the full tensile capacity of the member.

5. Splices. Splices in tubing should be made by telescoping, the outer tube being cut at an angle or by butting at an angle of 20 to 30 degrees to the tube axis and supporting with an inserted sleeve. External sleeves may also be used. All welds should be at an angle rather than straight around the tube, unless the member is loaded in pure torsion only.

6. Fairing. Truss fuselages can be faired out so as to give better shape and lower drag. Strips of wood, tubing, or dural sections are used for this purpose. These members should be strong enough to take the fabric tension and handling loads. The fairing strips are supported by clips on the structure or plywood formers built out from the bulkheads.

05.4650 PLYWOOD MONOCOQUE FUSELAGES

1. General. Plywood monocoque fuselages are either of simplified type with flat faces or of full curved form for the best aerodynamic efficiency. The simple type is built up on four main longerons with flat sides. The top "deck" is flat, round, or "V" shaped, faired down to the tail from the neck carrying the wing. The bottom surface is carried down in "V" shape to support the keelson above the landing skid. Pneumatic wheels are carried between main fuselage bulkheads, and recessed up into the fuselage body. Since the skids are usually sprung on rubber or other means for shock absorption, heavier bulkheads are provided locally to carry the loads into the main fuselage shell.

2. Plywood sizes. In general, the lightest practical sizes of plywood will be thick enough to carry the design shear loads in the side panels so that diagonals will not be required. Diagonals may be needed at panels where the external loads are high as between the main wing bulkheads, in the pilot's bay, etc. A minimum thickness of 1 mm. for birch and 3/64" for spruce or mahogany is recommended for fuselage covering.

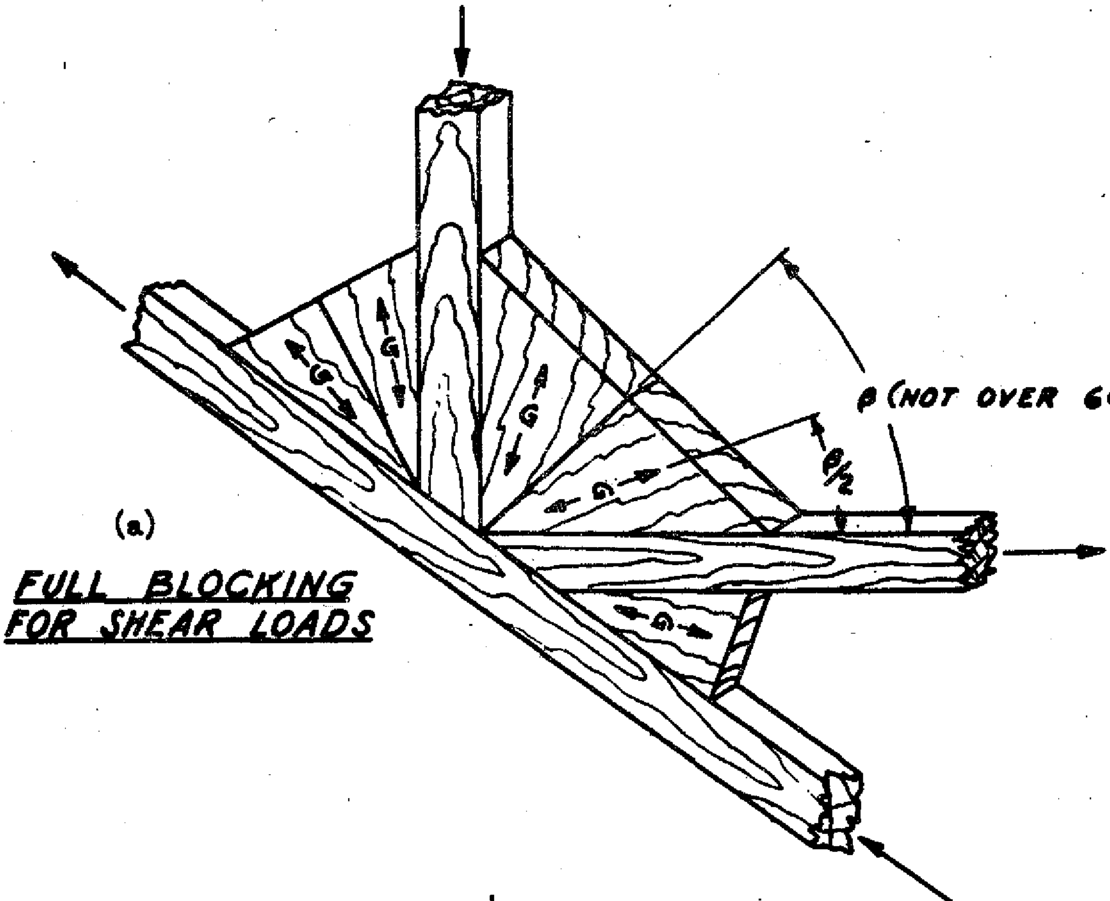
3. Bulkheads. The bulkheads in this type of fuselage can be made up of straight struts fastened at the corners by blocking notched for the longerons and gusseted with plywood.

4. Curved monocoque type. The curved monocoque style of construction necessitates laying the plywood on in smaller panels

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

5. Rigidity. Care should be taken that the minimum section just forward of the fin attachment is not so small as to be too flexible under torsional loads from the vertical tail surfaces, causing flutter in rough air or at high speed. This precaution also applies to all kinds of fuselages in which the rear part is necked down to form a long thin "boom". In such cases the stiffness in bending about both axes is important. No definite limits for rigidity can be set down, but it is well known that the natural vibrations in torsion and both bending directions should all be of different periods to prevent interaction. These vibration rates can be measured on the ground by vibrating artifically and counting the oscillations.

6. Concentrated loads. The loads from main fittings should be well distributed in the bulkheads by means of suitable blocking. Birch plywood or ash pads should be provided under fittings. Wood corner blocks which carry shear through the glue joint should be laminated pie-fashion if necessary to avoid gluing at an angle of more than 30 degrees off the grain direction. Butt glue joints on end grain will not carry shear or tension loads. For various methods of installing corner blocks, see Figure .4-12.



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G
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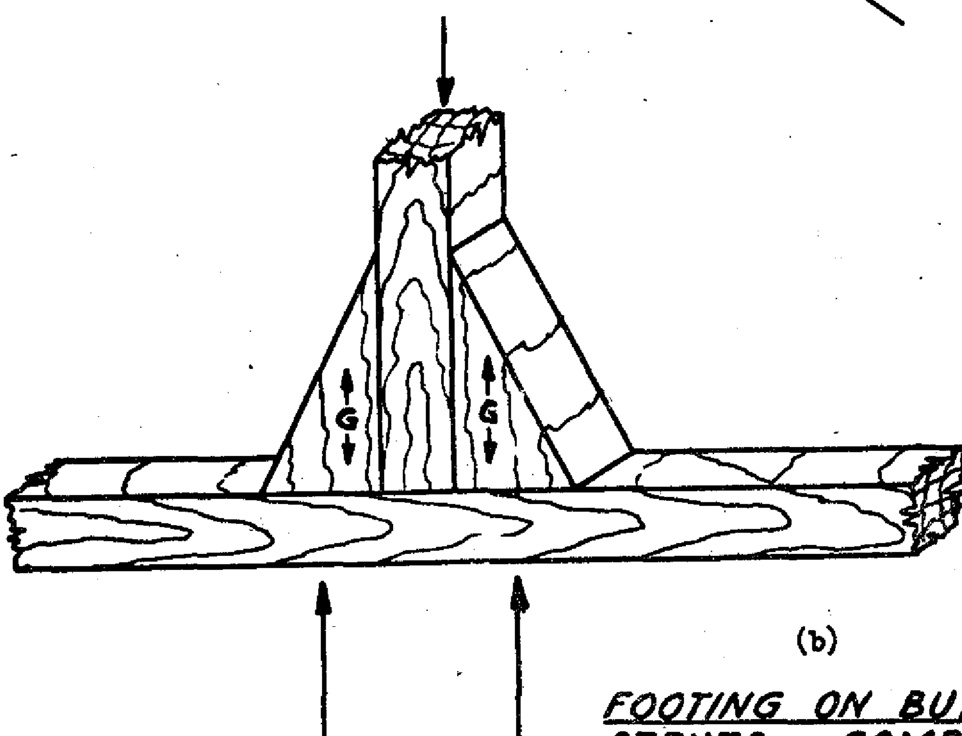
θ (NOT OVER 60°)
θ/2
(a)
FULL BLOCKING FOR SHEAR LOADS



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(b)
FOOTING ON BUTTED STRUTS - COMPRESSION

FIG. .4-12

05.47 TOW CABLE RELEASE MECHANISMS

Reference is made to CAR 05.342(d) regarding operation tests on tow cable release mechanisms, and to CAR 05.255 on the limit loads to be applied to the operating handle.

Releases should be designed to accommodate tow-line rings of 1-1/2 inches diameter, or greater, unless a special tow line terminal is employed. The following precautions should be observed:

a. It should be impossible for bolts, lugs, or other projections on the mechanism itself, or the structure surrounding the mechanism, to foul the tow line, or tow line parachute, under any conditions. If a special fitting is supplied to carry the parachute, it should be so located that the parachute cannot become entangled on the lift struts or wires. The forward end of the nose skid should be so constructed as to prohibit fouling between it and the tow line and/or parachute.

b. The operating handle in the cockpit should be so located, designed, and placarded that there is a minimum of danger of confusing it with the spoiler, brake, or other control handles.

c. Cables or wires used to operate the release should be protected from wear and abrasion from normal operation and from damage by occupants in entering or leaving the glider. When fixed or flexible tubing is used for cable or wire guides, there should be sufficient protection and drainage to prevent the formation of ice inside the tube.

05.70 FLIGHT REQUIREMENTS

Flight characteristics which render a glider unairworthy include:

a. Undue tendency to spin or spiral dive (see also CAAM 05.703).
b. Difficulty in recovering from spins or spiral dives (see also CAAM 05.703).
c. Lack of control at and below the stalling speed. This includes the inability to prevent the glider from falling off to one side under conditions encountered in an inadvertent stall in straight flight (see also CAAM 05.703).
d. Inability to control the glide path and speed within safe limits. This includes the inability to lose altitude in unexpected vertical up gusts, and inability to prevent "floating".
e. Lack of adequate control on the ground (see also CAAM 05.705).
f. Undue sensitivity or sluggishness in controls or extreme differences in pressure and/or motion between any of the three controls. Over-balancing of any control will be cause for rejection.
g. Lack of adequate visibility for pilot.
h. Undue tendency of the glider to stall easily while being towed.
i. Any features which prevent the pilot from knowing when excessive loads are being transmitted to the glider by the tow line (such as release placed too near the C.G.).
j. Undue possibility of accidentally exceeding the maximum certified airspeed in normal maneuvers or recovery from spins, stalls or spiral dives.

05.701 BALANCE

As used in the regulation (CAR 05.701) the term "balanced" refers to steady free flight in calm air without exertion of control force by the pilot.

The normal operating speed referred to in CAR 05.701 should fall within the following limits.

Class I and II Gliders
Between 1.25 V_s and 1.6 V_s

Class III Gliders

Between 1.25 V_g and .8 V_g (Table 05-2 of CAR 05) without the use of balancing devices that are adjustable in flight. If in tests for stability (see CAR 05.702 and CAAM 05.702), the oscillations are of such magnitude that there appears to be undue danger of exceeding V_g, the value .8 V_g should be reduced.

CAR 05.701 does not require that any special balancing devices, such as tabs or ballast, be employed in cases where the flight tests may be passed without changing the rigging during the tests. When balancing devices which are adjustable in flight are employed, adjustments may be made in flight during the flight tests. However, the glider must be controllable at all times at all possible settings of the balancing mechanism.

Balance may be attained by the use of movable surfaces, such as tabs or adjustable stabilizers, etc., or by ballast. (See also CAAM 05.7110).

05.702 STABILITY

Stability will be measured in the free-control condition. Although stability and balance are closely connected, they should not be confused. For instance, a nose heavy glider is not necessarily unstable. If, in this case, forces are present which tend to make the glider assume some speed at which it is balanced, it will be statically stable. If it actually attains this speed and ceases to oscillate, it is both dynamically and statically stable. If the craft is stable but extremely nose heavy, the speed at which it will cease to oscillate in the free-control condition may be higher than the placard gliding speed, or the oscillations involved may result in a speed higher than the placard speed. (See CAAM 05.701 for balance requirements).

Normal flight conditions as used in CAR 05.702 refer to normal maneuvers and speeds between V_s and V_g (placard).

05.703 SPINNING

Class I gliders are sometimes designed purposely with the ability to spin, since this usually affords a means of making a safe exit from storm clouds that have become too turbulent for normal flying. However, the use of suitable spoilers may eliminate the necessity for such a maneuver. Vicious or uncontrollable tendencies are to be avoided in Class I, as well as in Class II and III gliders.