BASIC GLIDER CRITERIA HANDBOOK

1961

Federal Aviation Agency

Flight Standards Service
Washington, D.C.

For sale by the Superintendent of Documents, U.S. Government Printing Office
Washington 25, D.C. - Price 75 cents

Preface

The purpose of the *Basic Glider Criteria Handbook* is to provide individual glider designers, the glider industry, and glider operating organizations with guidance material that augments the glider airworthiness certification standards specified in Civil Air Regulation Part 5. Acceptable methods of showing compliance with the standards are presented as compliance suggestions. Considerable material regarding common practices of construction and fabrication has been included primarily for the information of novice builders and designers, and should not be considered as the only satisfactory practices.

The *Basic Glider Criteria Handbook* was prepared in the Federal Aviation Agency, Flight Standards Service, by personnel of the Engineering and Manufacturing Division. Acknowledgement is extended to the *Soaring Society of America* for their cooperation and advice.

Contents

Chapter 2

Chapter 3

Chapter 4

Chapter 5

Chapter 6

Introduction

The *Basic Glider Criteria Handbook* contains design recommendations based on the present knowledge and development of glider design. They may be used as the minimum standards for establishing the classifications and related airworthiness of both conventional type gliders and those equipped with auxiliary powerplants.

New types of gliders and new materials and types of construction may, however, incorporate features to which the recommendations cannot be logically applied. In such cases, special consideration will be given to the particular new problems involved.

In cases where deviation from the conventional is small, sufficient evidence should be submitted to show that the proposed deviation will not be detrimental to the airworthiness of the design. When the deviation from the conventional is considerable, special recommendations covering the features in question should be obtained from the Federal Aviation Agency.

Insofar as the recommendations are concerned, rotating wing, seaplane and amphibian type gliders are considered unconventional.

Classification of Gliders

For the purpose of applying the recommendations, gliders are classified on the basis of certain imposed operating limitations.

In general, the various glider types can be classified as *high performance* or *utility* types.

A high performance or advanced type glider is especially designed for the maximum performance within a selected range of conditions.

The utility type is a medium performance, nonacrobatic type of glider designed for normal operations.

When doubt exists as to a particular glider's classification, the basic flight envelope, together with a three-view drawing, should be submitted to FAA so that a definite recommendation can be obtained. (If acrobatics are contemplated, provisions should be made for installation of parachutes, which are required for each occupant.)

TABLE 1–i

Glider Operating Limitations

Type of glider	"Never Exceed" placard speed, m.p.h.	Auto-winch tow placard speed, m.p.h.	Airplane tow	Instrument flying
High performance	$.90V_D$ (without dive brakes).[1]	$.90V_{tow}$	Permitted	Permitted.[2]
Utility	$.90_D$ (when dive brakes are installed).	$.90V_{tow}$	Permitted	Not permitted.

[1] See chapter 5, Airspeed limitations.

[2] See chapter 4, Instrument flight.

*Glider Kits**

Gliders built from kits are eligible for FAA certification if supported by a statement certifying that the glider was constructed in accordance with FAA-approved drawings and the manufacturer's manual of directions for building the glider; and that the parts and materials used, if other than those furnished with the kit, meet the manufacturer's recommendations. Also, the following inspections, by FAA representative, and tests should be made.

- Inspection for workmanship, materials and conformity prior to installation of covering (plywood, metal, or fabric) on any wing or control surface. (In general, any type of construction may be "closed over" provided it is possible to conduct an adequate inspection of critical parts subsequently. For example, the interiors of monocoque construction should be inspected prior to closing over unless adequate inspection openings are provided.)
- A final inspection of the completed glider.
- A check of flight characteristics.

Arrangement of Handbook

The material in *Basic Glider Criteria Handbook* has been arranged so that the particular glider airworthiness recommendation is followed by the suggested methods for showing compliance. In cases where methods for showing compliance are not needed, the airworthiness recommendation will have no accompanying guidance material.

Procedures and practices that assure safety equal to those listed in this handbook will also be acceptable. Any provisions which are shown to be inapplicable in a particular case will be modified upon request and evaluation by the Federal Aviation Agency.

For further FAA guidance material on glider certificates, compliance procedures and related subjects, the following references are listed:

Subject	*Document*	*Section*
Airworthiness Certificates	CAM 1	1.60
Experimental Certificates	CAM 1	1.73
Restricted Certificates	CAM 1	1.68
Type Certificates	CAM 1	1.10
Supplemental Type Certificates	CAM 1	1.25
Production Certificates	CAM 1	1.30
Technical Data Submittals	CAR 5	5.14
Inspection and Tests	CAM 1 CAR 5	1.15 5.15
Design Changes	CAM 1	1.20
Repairs and Alterations	CAM 18	------
Foreign Imports	CAR 10	------
Identification and Marking	CAM 1	1.100
Flight Tests	CAM 1	1.15

*Prior to beginning construction, it should be determined that the manufacturer of the kit has obtained a type certificate for the prototype glider built from such a kit.

Definitions

TECHNICAL TERMS AND RELATED SYMBOLS

(For definitions of nontechnical terms, refer to Par. 1.1 of CAR Part 1)

Aerodynamic coefficients, C_L, C_M, C_P, *et cetera.*—The coefficients hereinafter specified are those of the "absolute" (nondimensional) system adopted as standard in the United States. The subscripts N and C used herein 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 (wing load distribution) as required in Chapter 1, p. 13. (See figs. i–II, i–III and i–IV.)

Air density, P.—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 denotes the mass density of air at sea level under standard atmospheric conditions and has a value of 0.002377 slugs per cubic foot. (See definition of standard atmosphere.)

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.

Calibrated airspeed, CAS.—Speed equal to the indicated airspeed reading corrected for position and instrument error.

Design aircraft tow speed, V_{ta}.—The maximum indicated airspeed at which the glider is to be towed by aircraft.

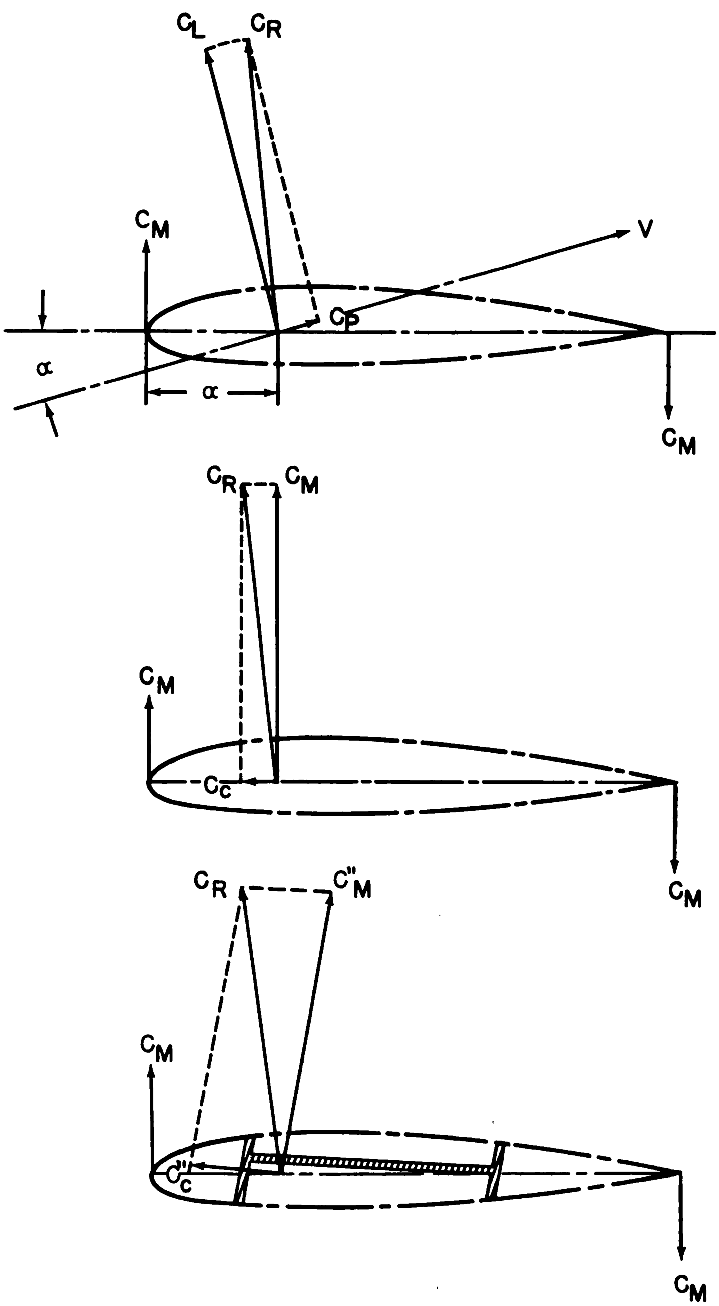
Design auto-winch tow speed, V_{taw}.—The maximum indicated airspeed at which the glider is assumed to be towed by automobile or winch.

Design flap speed, V_f.—The indicated airspeed at which maximum operation of high-lift devices is chosen.

Design gliding speed, V_g.—The maximum indicated airspeed to be used in the structural loading conditions.

Design gust velocity, U.—A specific gust velocity assumed to act normal to the flight path.

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}.



Figures i–II, i–III and i–IV. ***Illustration of airfoil force coefficients.***

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. (See fig. i–I.)

Design wing loading, s = W/S.—The gross weight divided by the design wing area. (Area computed with wing flaps in retracted position, if so equipped.)

Dive speed, V_D.—The maximum indicated airspeed for which the glider is demonstrated to be free from flutter or any other undesirable flight characteristics, but not to exceed V_g for utility category gliders. For high performance gliders V_D shall not exceed 1.2 V_g.

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

$= \frac{1}{2} PV_t^2$ (in terms of true airspeed in feet per second).

$= \frac{1}{2} P_0V^2$ (in terms of equivalent airspeed in feet per second).

$= V^2/391$ pounds per square foot, when V is miles per hour IAS.

Gross weight, W.—The design maximum weight of the glider and its contents, used for purposes of showing compliance with the specified recommendations.

Indicated airspeed, IAS.—Speed equal to the pitot static airspeed indicator reading as installed without correction for system errors but including sea level standard adiabatic compressible flow correction, and instrument error.

Load factor or acceleration factor, n.—The ratio of a force acting on a mass to the weight of the mass. When the force 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 as a multiple of the gravitational constant g.

Limit load.—The maximum load anticipated in service.

Limit load test.—A static test in which the limit loads are properly applied.

Never exceed speed, V_{NE}.—The maximum indicated airspeed for which the glider is certificated for operation.

Primary structure.—Those portions of the glider structure, the failure of which would seriously endanger the airworthiness of the glider.

Proof load.—The prescribed externally-applied load, multiplied by the proof factor (normally 1.0).

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 NACA Technical Report No. 218.)

Standard mean chord.—The length of the standard (geometric) mean chord is given by the gross wing area divided by the span. The S.M.C. is located in terms of the co-ordinates of its quarter chord point and its inclination, both measured with respect to the standard aero-dynamic axes.

Terminal velocity.—The maximum speed obtainable in diving flight.

True airspeed, TAS.—True airspeed of the glider relative to undisturbed air.

Ultimate factor of safety, j.—A design factor used to provide for the possibility of loads greater than those anticipated in normal conditions of operation and for design uncertainties.

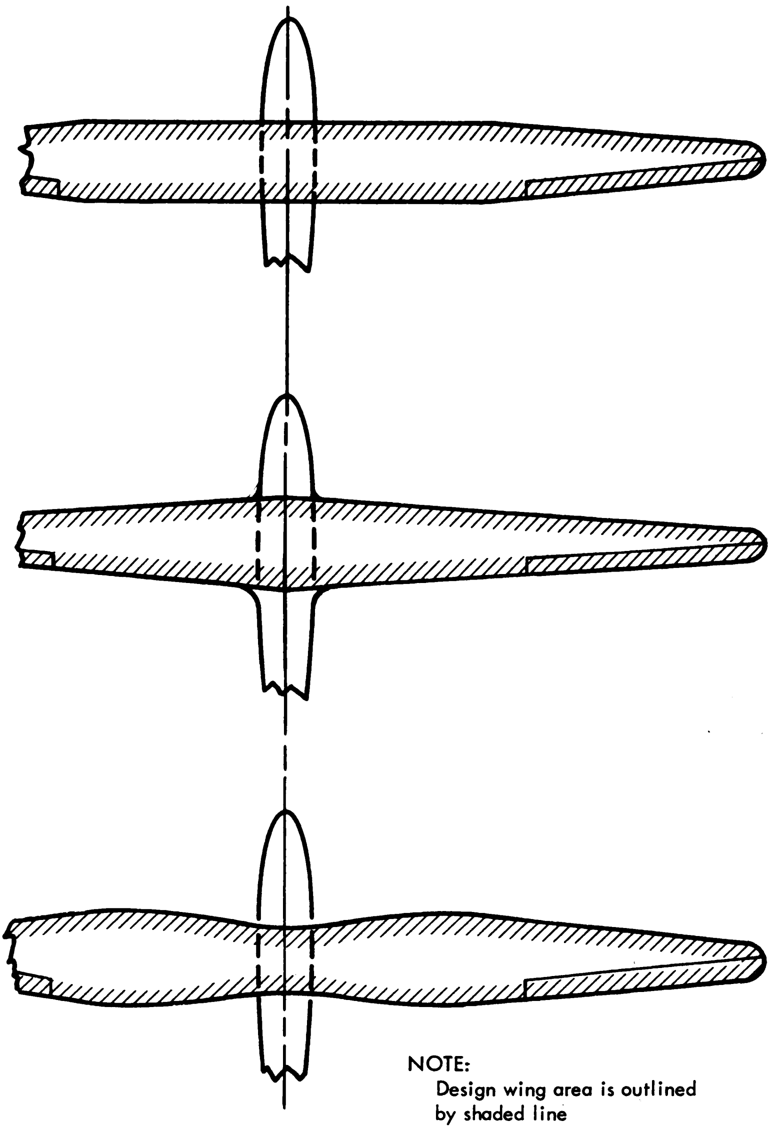


Figure i–I. ***Typical design wing areas.***

Ultimate load.—The maximum load which the structure is required to support. It is obtained by multiplying the limit load by the ultimate factor of safety.

Ultimate load test.—A static test in which the ultimate loads are properly applied. Loads should be supported for at least 3 seconds.

Chapter 1—LOADS

STRENGTH AND DEFORMATIONS

The primary structure should be capable of supporting the ultimate loads determined by the loading conditions and ultimate factors of safety specified herein, if the loads are properly applied and distributed. The structure must support these loads for a minimum period of 3 seconds.

The primary structure should also be capable of supporting, without detrimental permanent deformations, the limit loads of the loading conditions specified, if the loads are properly distributed and applied. In addition, temporary deformations that occur before the limit load is reached should be such that repeated occurrence would not weaken or damage the primary structure.

Compliance Suggestion

DETERMINATION OF DEFORMATION

- Detrimental permanent deformations are usually considered as those that correspond to stresses in excess of the yield stress. The yield is the stress at which the permanent strain is equal to 0.002 inches per inch from standard test specimens.
- In determining permanent deformations from static test results, the effects of slippage or permanent deformation of the supporting jig should be considered.
- If any deflections under load would change significantly the distribution of external or internal loads, such distribution should be taken into account.

Stiffness.—The structure should be capable of supporting limit loads without suffering detrimental permanent deformations. At all loads up to limit loads, the deformation should not interfere with safe operation of the glider.

Wing drag truss.—Fabric covered wing structures, having a cantilever length of overhang such that the ratio of span overhang to the chord at the root of the 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, the counter wires should be of the same size as the drag wires.

Compliance Suggestion

CABLE LIMITATIONS

Multistrand cables should not be used in drag trusses since they stretch excessively.

Loads imparted by safety belts.—Structures to which safety belts are attached should be capable of withstanding the following ultimate acceleration that occupants are assumed to be subjected to during minor crash conditions:

Upward	4.5 g.
Forward	9.0 g.
Sideward	3.0 g.

If the belt is attached to the seat, the structural investigation should be carried through to the primary structure. Also, the above noted accelerations should be multiplied by a factor of 1.33 when applied to the seat attachment to the structure.

Pilot and passenger loads.—Pilot and passenger loads in the flight conditions should be computed for a standard passenger weight of 170 pounds. A minimum ultimate factor of safety of 1.5 should be used in conjunction with the applicable acceleration or maneuvering factor.

Local loads.—The primary structure should be designed to withstand local loads caused by dead weights and control loads. Baggage and ballast compartments should be designed to withstand loads corresponding to the maximum authorized capacity. Concentrated (dead) weights include items such as batteries, radios, seats, et cetera.

Loading equilibrium.—Unless provided for otherwise, the air and ground loads should be placed in equilibrium with inertia forces, considering all items of mass in the glider. All such loads may be distributed in a manner conservatively approximating or closely representing actual conditions.

FLIGHT LOADS

The airworthiness of a glider with respect to its strength under flight loads usually is based on the airspeeds and accelerations (from maneuvering or gusts) that can safely be developed in combination. For certain types of gliders, the acceleration factors (specified in terms of load factors) and gust velocities are arbitrarily specified and should be used for these classes. The airspeeds which can safely be developed in combination with the specified load factors and gusts should be determined in accordance with the procedure specified, and they should serve as a basis for restricting the operation of the glider in flight.

Design airspeed.—The design airspeeds should be selected so that 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 1–I.

Compliance Suggestion

DETERMINATION OF AIRSPEED VALUES

The values of the design airspeeds in table 1–I are minimum values. In certain cases it may be desirable to use larger values for high performance type gliders. In order to provide for a high auto-winch tow placard speed, it may be advantageous to use a higher design gliding speed.

Compliance Suggestion

USE OF K VALUES

The K values specified in table 1–I have been determined on the basis of studies of the "cleanness" of current gliders. The values of K for high performance gliders have been set approximately 11 percent higher than the values of K for utility gliders since instrument flying in high performance gliders is permitted and higher speeds are apt to be encountered in recovery from inadvertent upsets. These attitudes are more likely to occur in instrument flying than in normal operations under good weather conditions. Since these constants have been established on a simplified basis, it is possible that they may lead to irrational values of V_g when applied to particular cases. In any event, it will be necessary to design high performance gliders to a V_g greater than .40 times the terminal velocity or to design utility gliders to a V_g greater than .36 times the terminal velocity. In cases where the value of V_g is based on terminal velocity in accordance with the above, calculations substantiating the terminal velocity value should be submitted.

Load factors.—The flight load factors specified should represent wing load factors. The net load factor or acceleration factor should be obtained by proper consideration of balancing loads acting on the glider in the flight conditions specified in this chapter under "Maneuvering load factors." The net or dead weight load factors should be obtained from balancing computations such as are outlined under "Balancing loads."

Maneuvering load factors.—The limit maneuvering load factors specified in table 1–I should be considered as minimum values unless it can be proved that the glider embodies features of design which make it impossible to develop such values in flight, in which case the proven lower values may be used.

In some cases it may be advisable to use higher values, as when a higher auto-winch tow placard speed is desired.

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

$N = 1 + \Delta n$ where Δn = limit load factor increment

k = gust reduction factor (fig. 1–I)

U = gust velocity, f.p.s.

V = indicated airspeed, m.p.h.

s = W/S, wing loading, p.s.f.

$= 1 + \frac{KUVm}{575s}$ m = slope of lift curve, C_L per radian corrected for aspect ratio, R.

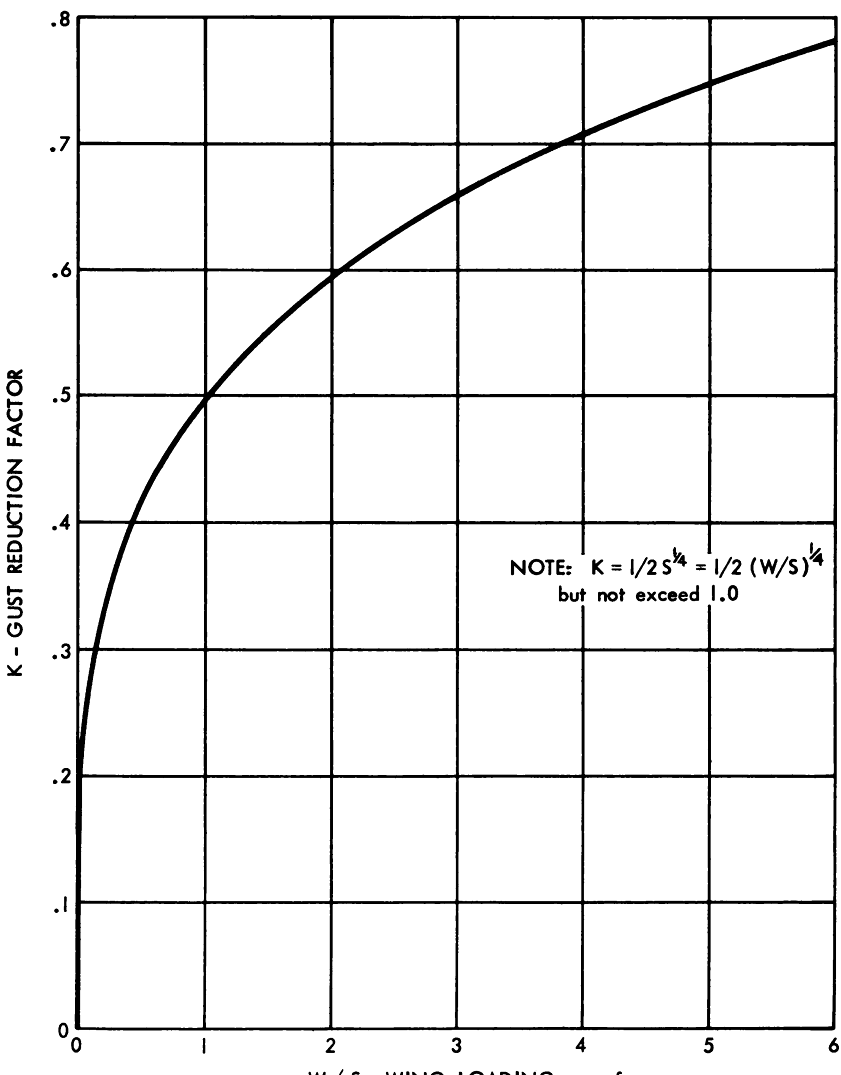


Figure 1–I. ***Gust reduction factor.***

Factors of safety.—A minimum limit factor of safety of 1.0 and a minimum ultimate factor of safety of 1.5 should be used unless otherwise specified. Also, refer to p. 43 for multiplying factors of safety required.

TABLE 1–I

Minimum Design Airspeeds and Minimum Limit Load Factors For the Symmetrical Flight Conditions

Class of glider	High performance	Utility
1. Design Gliding Speed [1] (V_g, m.p.h.).	$K(s)^{1/2}$ [2]	$K(s)^{1/2}$ [2].
2. Design Auto-Winch Tow Speed, V_{taw}, m.p.h.	$35(s)^{1/2}$	$35(s)^{1/2}$.
3. Design Flap Speed Vf, m.p.h	1.67 Vsf	1.67 Vsf.
4. Positive Maneuver Load Factor, n.	5.33	4.67.
5. Positive Gust Load Factor [3]	Corresponding to a 24 f.p.s. "up" gust at V_g.	Corresponding to a 24 f.p.s. "up" gust at V_g.
6. Positive Auto-Winch Tow Load Factor.	([4])	([4]).
7. Negative Maneuver Load Fac-Factor, n.	−2.67	−2.33.
8. Negative Gust Load Factor [5]	Corresponding to a 24 f.p.s. "down" gust at V_g.	Corresponding to a 24 f.p.s. "down" gust at V_g.
9. Design Dive Speed, V_D	Not to exceed 1.2 V_g	Not to exceed V_g.

[1] The design gliding speed, V_g, shall not be less than the design aircraft tow speed V_{ta}.

[2] The following value of K should be used; however a conservative interpolation of the value of K for a particular design will be acceptable:

Glider configuration	High performance	Utility
For gliders of very clean design with cantilever wings.	61	55.
For gliders of clean design having single strut braced wings.	56	50.
For gliders of the utility type having double strut braced wings and open cockpit.	51	46.

[3] Positive Gust Load Factor: $n=1+\frac{KUV_gm}{575s}$

$$n=1+\frac{K24V_gm}{575s}$$

[4] Positive Auto-Winch Tow Load Factor: $n=\frac{1}{(s-e)}\left[\frac{(V^2{}_{taw})}{391}-e\right]$,

where s=des. wing loading, p.s.f.

e=unit wing weight, p.s.f.

V_{taw}= Item 2 of table 1-I

The factor 391 above is based on a C_L value of 1.0 which experience indicates to be the maximum value likely to be reached when auto or winch towing the glider.

[5] Negative Gust Load Factor: $n=1-\frac{K24\ V_gm}{575s}$

SYMMETRICAL FLIGHT CONDITIONS

(Flaps Retracted)

Basic flight envelopes.—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 particular design criteria.

Compliance Suggestion

DETERMINING FLIGHT ENVELOPE

A sample basic flight envelope is shown in fig. 1–II. This envelope has been constructed for a high performance sailplane of aerodynamically "clean" design, having a full cantilever wing, with basic design values as follows:

W/S = s = 3.5 lbs. per sq. ft.
R (Aspect Ratio) = 12
m (Corrected to R of 12) = 4.8 C_L per radian
e (weight of wing) = 1.5 lbs. per sq. ft.

a. In accordance with table 1–I, a value of K = 61 was used so that the minimum design gliding speed which was used was 61 $\sqrt{s}$ = 114 m.p.h. (V_{gmn}). The corresponding placard "never exceed" speed would be .9x114 = 103 m.p.h. (From fig. 1–I.) In this particular case, however, it is assumed by the designer that a somewhat higher placard "never exceed" speed is desirable so a design V_g of 125 m.p.h. is selected making the corresponding placard speed 112 m.p.h. (.9x125).

b. Plot the following equation to obtain line 1 of the positive portion of the *V*-n diagram. (See fig. 1–II.)

$$n = \frac{V^2 C_{L\,max}}{391\ s}$$

where n = maximum possible positive limit wing load factor at the speed *V* (m.p.h.).

c. Draw a vertical line through the velocity corresponding to V_g (line 2 of fig. 1–II).

d. Plot the following equation to obtain line 3 of the negative portion of the *V*-n diagram (see fig. 1–II).

$$n = \frac{V^2}{391\ s}$$

where n = maximum possible negative unit wing load factor at the speed *V* (m-p.h.). This is based on a C_L max. (dynamic) of 1.0 (negative).

e. When applying the recommendations specified above, the following procedure should be followed:

1. Draw a straight line (line 4 of fig. 1–III) from the point where $V = 0$, and $n = 1$ to the point where $V = V_g$ and n = the load factor specified in item 6 of table 1–I. This will intersect line 2 at point E.

2. Draw a horizontal line (line 5 of fig. 1–II) through the greatest value of *n* specified in items 5 and 7 of table 1–I. This will intersect line 1 at point *C*; and it will intersect line 4 at point *D*, providing the positive gust load factor is greater than the maneuvering load factor. If the maneuvering load factor is greater than the positive gust load factor, line 5 will intersect witn line 2.

Example

ITEM 5. The specified maneuver load factor = 5.33

ITEM 6. The gust reduction factor from fig. 1–I is .685 (for $s = 3.5$) so for this example:

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

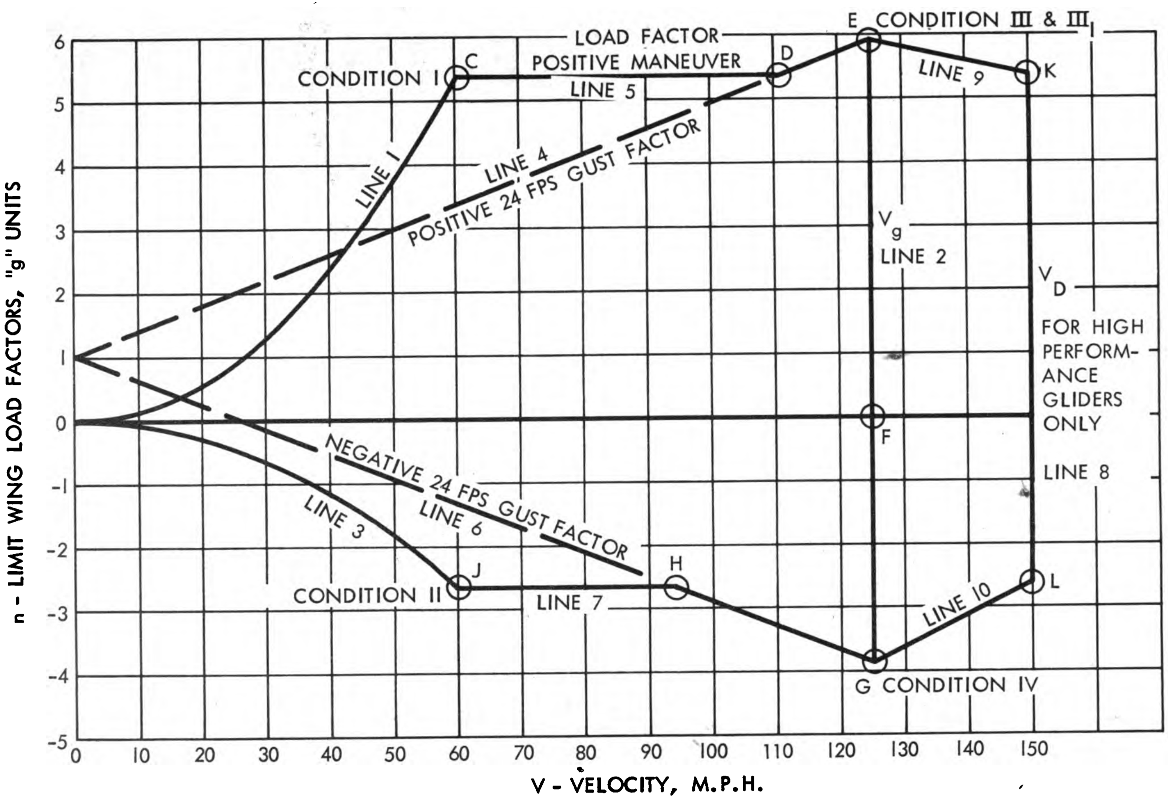


Figure 1—II. ***Sample basic flight envelope showing critical basic flight conditions.***

ITEM 7. $V_{taw} = 35\ (3.5)^{1/2} = 65.5$ m.p.h.

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

The greatest positive load factor of items 5 and 7 above is 5.33 and it therefore determines line 5 of the basic flight envelope of fig. 1–II. It should be noted that the positive portion of the basic flight envelope is represented by the points OCDEF of fig. 1–II.

f. Draw a straight line (line 6 of fig. 1–II) from the point where $V=0$ and $n=+1$ to the point where $V=V_g$ and $n=$the load factor specified in item 9 of table 1–I. This will intersect line 2 at point G.

g. Draw a horizontal straight line (line 7 of fig. 1–II) through the negative value of n specified in item 8 of table 1–I. This will intersect line 3 at point J and will intersect line 6 at point H providing the negative gust load factor is greater than the maneuvering load factor. If the negative maneuvering load factor is greater than the negative gust factor, line 7 will intersect with line 2.

Example

ITEM 8. The specified maneuver load factor = $\underline{-2.67}$

ITEM 9. The negative load factor due to a down gust is as follows:

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

It should be noted that the negative portion of the basic flight envelope is represented by the points FGHJO of fig. 1–II.

h. The glider need not be investigated for gust loads at speeds higher than V_g. For V_D higher than V_g, in high performance gliders only, but not to exceed 1.2 V_g the right hand corner of the V-n diagram need only be investigated for the maneuvering load factors. This portion of the V-n diagram is established as follows:

1. Draw a vertical line at V_D max. (not to exceed 1.2 V_g) (line 8 of fig. 1–II).
2. Draw a straight line (line 9 of fig. 1–II) from point E to intersect line 8 at $n=5.33$ (point K) (item 5 of table 1–I).
3. Draw a straight line (line 10 of fig. 1–II) from point G to intersect line 8 at $n=2.67$ (point L) (item 8 of table 1–I). This additional portion of the V-n envelope is represented by the points EKLG.

i. In general, an investigation of the following specific basic flight conditions, which correspond to points on the basic flight envelope, will insure satisfactory coverage of the critical loading conditions.

1. *Condition I.*—(Positive High Angle of Attack.)

This condition corresponds to point C on the basic flight envelope. The aerodynamic characteristics C, $C.P.$ (or C_m), and C_c to be used in the investigation should be determined as follows:

$$C_{n_I} = \frac{n_I s}{q_I}$$

where n_I = Wing load factor corresp. to pt. C.

q_I = Dynamic pressure corresp. to the velocity V_I, which in turn corresp. to point C.

$C.P.$ or C_m = value corresp. to C_{n_I} as determined from the airfoil characteristics curves.

C_c = value corresp. to C_n, as obtained from the airfoil characteristics curves.

2. *Condition II.*—(Negative High Angle of Attack.)

This condition corresponds to point J on the basic flight envelope.

(*a*) $C_{n_{II}} = \dfrac{n_{II} s}{q_{II}}$

(*b*) C_c = value corresp. to $C_{n_{II}}$ (may be assumed equal to zero if positive).

(*c*) $C.P.$ or C_m = value corresp. to $C_{N_{II}}$.

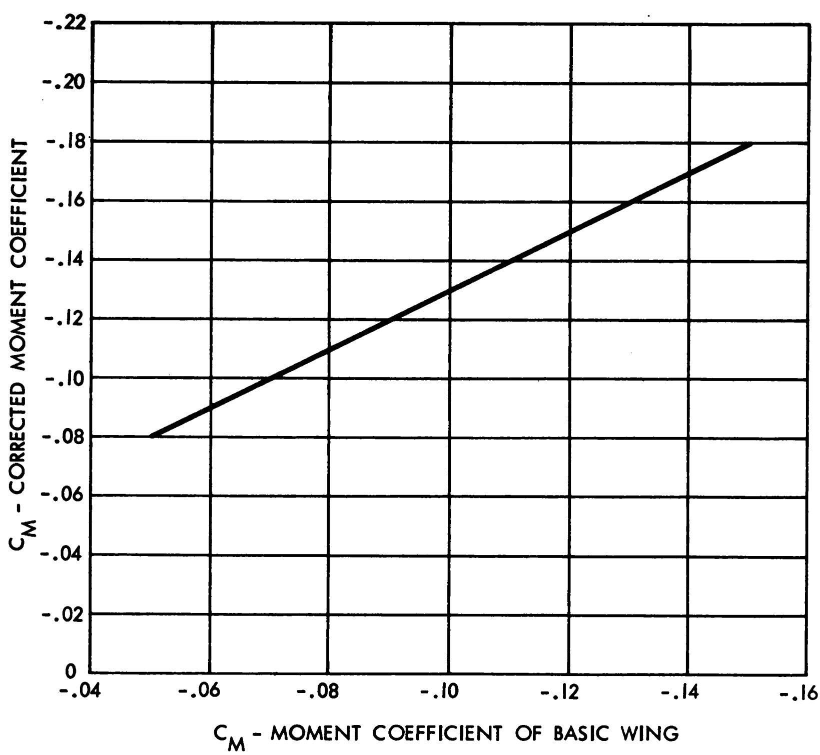


Figure 1–III. *Corrected moment coefficient for condition III_l.*

3. *Condition III.*—(Positive Low Angle of Attack.)

This condition corresponds to point E on the basic flight envelope.

(a) $C_{III} = \frac{n_{III}s}{q_{III}}$

(b) $q_{III} = q_g$

C_c = value corresp. to $C_{n_{III}}$ (may be assumed equal to zero if positive).

(c) $C.P.$ or C_m = value corresp. to $C_{n_{III}}$

4. *Condition III_I.*—(Modified Positive Low Angle of Attack.)

In order to cover the effects of limited use of the ailerons at V_g on the wings and wing bracing, such structure should be investigated for the following:

(a) $C_{n_{III_I}} = C_{n_{III}}$

(b) C_c = value corresp. to $C_{N_{III_I}}$

(c) C_m' = value obtained from fig. 1–III where C_m is the value corresp. to $C_{N_{III}}$. C_m' need only be applied to that portion of the span incorporating ailerons, using the basic value of C_m determined in Condition III over the remainder of the span.

5. *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:

(a) $C_{nIV} = \frac{n_{IV}s}{q_{IV}}$

(b) C_c = value corresp. to $C_{n_{IV}}$ (may be equal to zero if positive).

(c) $C.P.$ or C_m = value corresp. to $C_{n_{IV}}$.

SYMMETRICAL FLIGHT CONDITIONS

(Flaps or Auxiliary Devices in Operation)

High-lift devices.—When flaps or other auxilliary high-lift devices are installed on the wings, suitable provisions should be made to account for their use in flight at the design flap speed V_f. Minimum values of the design flap speed are specified in table 1–I. These provisions should be based on the intended use of such devices.

Compliance Suggestion

INSTALLATIONS ON INTERNALLY BRACED WINGS

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:

The average value of C_m' used in design Conditions III and IV 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. V_g is the design gliding speed used in Conditions III and IV.

Compliance Suggestion

COMPUTATIONS NECESSARY

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.

- 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.
- 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. 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 for acceptable flap data.

UNSYMMETRICAL FLIGHT CONDITIONS

In the required investigation for the effects of unsymmetrical flight loads, the following assumptions should be made:

- Modify Conditions I and III (Chap. 1, pp. 9, 10) and the most critical negative condition by assuming 100 percent of the air load to be acting on one side of the glider and 40 percent on the other.
- 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 percent of the normal (symmetrical) loading to those determined by considering 30 percent of the normal total load to be acting upward on one wing panel and 30 percent to be acting downward on the other. The unbalanced moment or torque is caused by the differential of 60 percent 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. (See fig. 1–IV.)

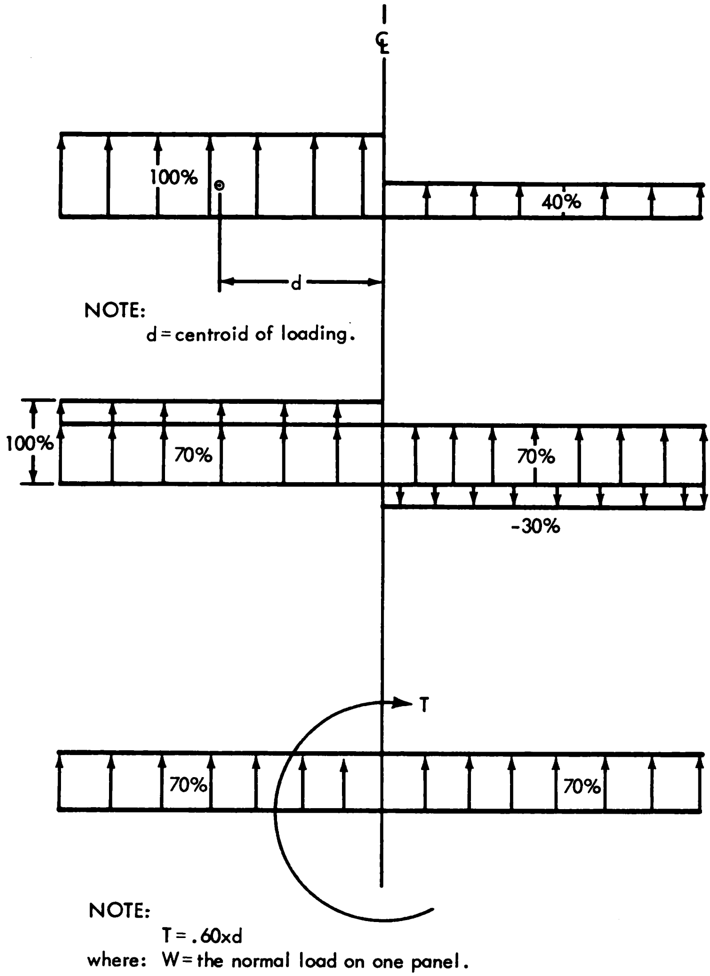


Figure 1—IV. *Span load distribution for unsymmetrical flight conditions.*

The angular acceleration "*a*" resulting from the torque "T" may be obtained as follows:

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

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

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

$$T_n = I_n \, a \text{ (ft. lbs.)}$$

when $I_x = M_N \, d_N^2$

where M_N = the mass of part N
d_N = the distance from the longitudinal axis to the CG of part N in feet.

SPECIAL FLIGHT CONDITIONS

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

Compliance Suggestion

SPAN DISTRIBUTION DETERMINATION

(*a*) For wings having mean taper ratios equal to or greater than .33, the span distribution should be determined as follows:

1. If the wing does not have aerodynamic twist (that is, if the zero lift lines of all sections are parallel), the span distribution for normal force coefficient (C_N) should be assumed to vary in accordance with figs. 1–V and 1–VI, which are assumed to represent two extreme cases of tip loading. Each case should be investigated, unless it is demonstrated that only one is critical. As an alternative method, it will be acceptable to investigate each design condition for only one span distribution using a rational distribution, except in the case of the high-angle-of-attack condition which gives the maximum forward chord loads (Condition I). For this condition the analysis should be made for both the rational distribution and that given in fig. 1–V.

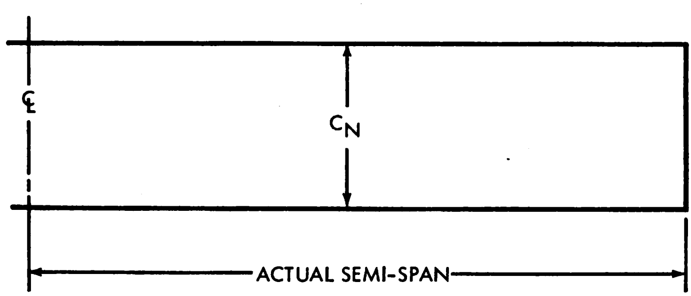


Figure 1–V. ***Span distribution—No tip loss.***

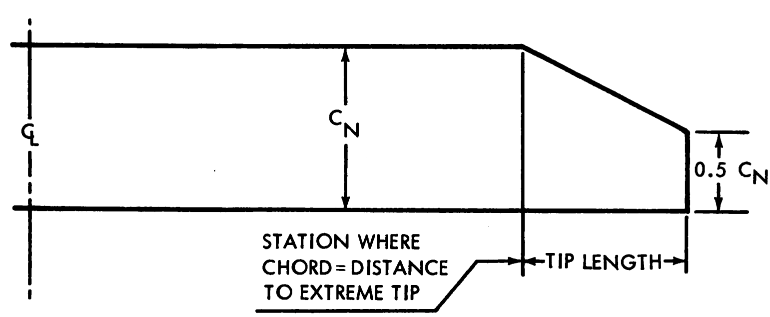


Figure 1–VI. ***Span distribution—With tip loss.***

Figures 1–V and 1–VI. ***Span distribution of C_N for wings.***

2. If the wing has aerodynamic twist, the span distribution should be determined by the alternative method given in 1, above.
3. For these purposes, the mean taper ratio is defined as the ratio of the tip chord (obtained by extending the leading and trailing edges to the extreme wing tip) to the root chord (obtained by extending the leading and trailing edges to the plane of symmetry).
4. Acceptable methods of determining a rational span distribution are given in publication ANC-1 "Spanwise Air Load Distribution," and in NACA Technical Report Nos. 572, 585 and 606.

b. For all wings having mean taper ratios less than .33 the span distribution should be determined by rational methods, unless it is shown that a more severe distribution has been used. Acceptable methods of determining rational span distribution are given in the aforementioned documents of a, 4, above.

c. When the normal force coefficient is assumed to vary over the span, the value used should be adjusted to give the same total normal force as the design value of C_n acting uniformly over the span.

d. When figs. 1-V and 1-VI are used, the chord coefficient should be assumed to be constant along the span.

e. 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.

Compliance Suggestion

DETERMINING CHORD DISTRIBUTION

a. The approximate method of chord loadings outlined under "Rib Tests" in Chapter 2 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, 456, 631, and 634 are suitable for this purpose. These methods consist of 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 air foils can be obtained directly from the NACA.

b. On high speed gliders the leading edge loads developed may be exceptionally severe, particularly the "down" loads which are produced by negative gusts when the glider is 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.

c. 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.

Compliance Suggestion

DETERMINATION OF RESULTANT AIR FORCES

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. 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.

a. In general, the summation of all forces acting on a wing can be expressed as a single resultant force acting at a certain 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 coefficient, 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.

b. For convenience and clarification, table 1–Ia has been developed and the various curves obtained as a part of this method are illustrated in figs. 1–VII, 1–VIII, and 1–IX. 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 (coordinates) to which the curve is drawn. The procedure is as follows:

1. Fig. 1–VII illustrates the actual wing planform, plotted to a suitable scale. This should agree with the definition of design wing area.

TABLE 1–Ia

Determination of Resultant Air Forces

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=

$\bar{b}$=

$\bar{c}$=

MAC=

2. Fig. 1–VII (*b*), shows the variation of wing chord, C, with the span. The values of *C* are entered in table 1–Ia 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.
3. Fig. 1–VII (*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 1–Ia under item (3).
4. Fig. 1–VII (*d*), is obtained by plotting R_bC (item (4)) in table 1–Ia against span. The ordinates of this curve are proportional to the actual force distribution over the span. The area under curve 1–VII (*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 1–VII (*d*) by the area under curve 1–VII (*b*), using the same units of measurement for each area. This value of K_b is indicated by the dotted line on curve 1–VII (*c*).
5. To determine the location of the mean aerodynamic center along the span, fig. 1–VIII is drawn. The ordinates are obtained by multiplying the ordinates of curve 1–VII (*d*) by their distance along the span, as shown in item (5), of table 1–Ia. The area under curve 1–VIII (*a*), divided by the area under curve 1–VII (*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. 1–VII (*a*) by the dimension *b*.

6. The locus of the aerodynamic centers of each individual wing chord is plotted on fig. 1–VII (*a*) as the dotted line A–B. In table 1–Ia the distance "x" from the base line O–E to the line A–B is entered under item (6).

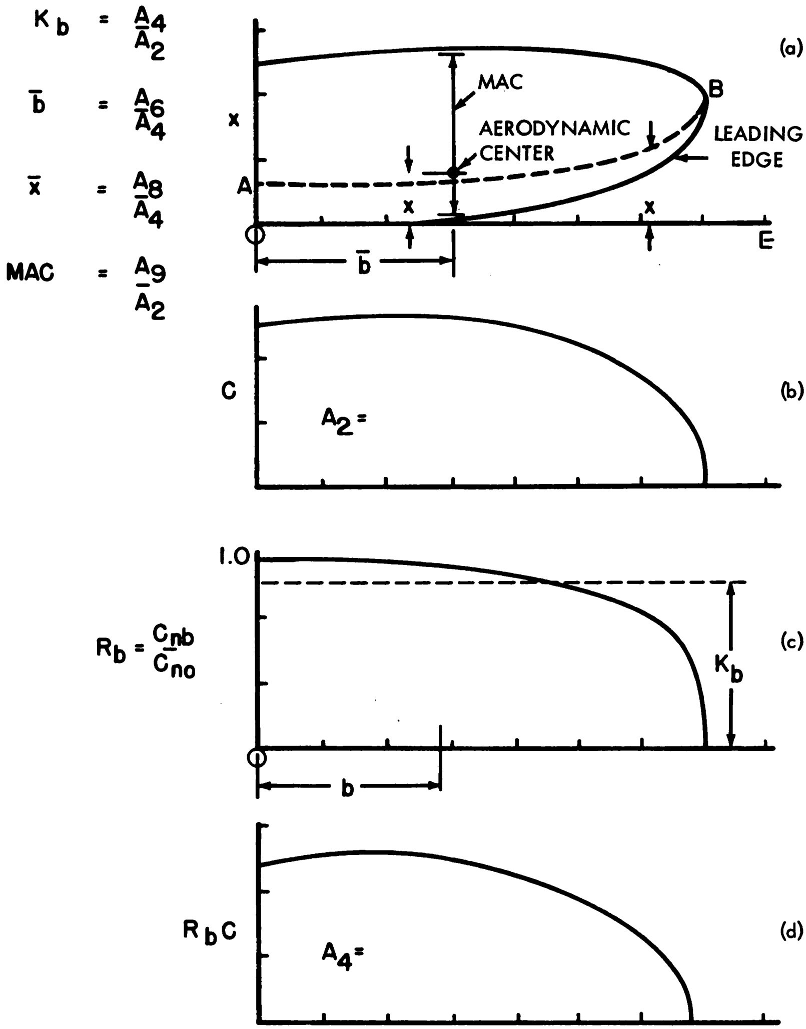


Figure 1–VII. *Determination of effective normal force coefficient.*

7. Fig. 1–VIII is now plotted, using as ordinates the values of $R_b\ C_x$ obtained from item (7) of table 1–Ia. The area under curve 1–VIII (*b*), divided by the area under curve 1–VII (*d*), gives the distance of the mean aerodynamic center from the base line O–E in fig. 1–VII (*a*). This distance is indicated as $\bar{x}$ on that figure.

8. If it is assumed that the moment coefficient about the aerodynamic center of each individual chord is constant over the span, the magnitude of the mean aerodynamic chord is determined by means of fig. 1–VIII (c). The ordinates for this curve are

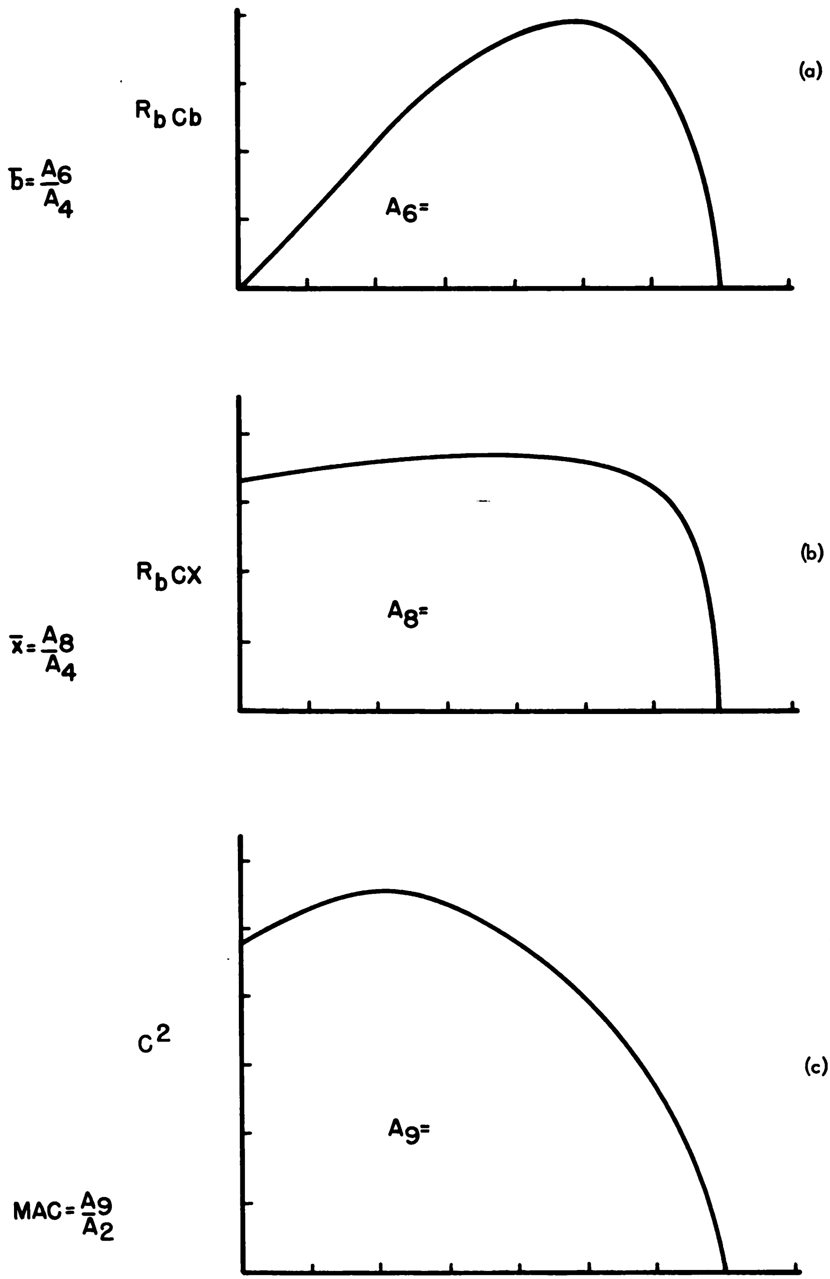


Figure 1–VIII. ***Determination of mean aerodynamic center.***

determined from item (8) of table 1–Ia. The area under curve 1–VIII (*c*), divided by the area under curve 1–VII (*c*), gives the value of the mean aerodynamic chord. By way of illustration, it is drawn on fig. 1–VII (*a*), so that its aerodynamic center coincides with the location of the mean aerodynamic center of the wing panel.

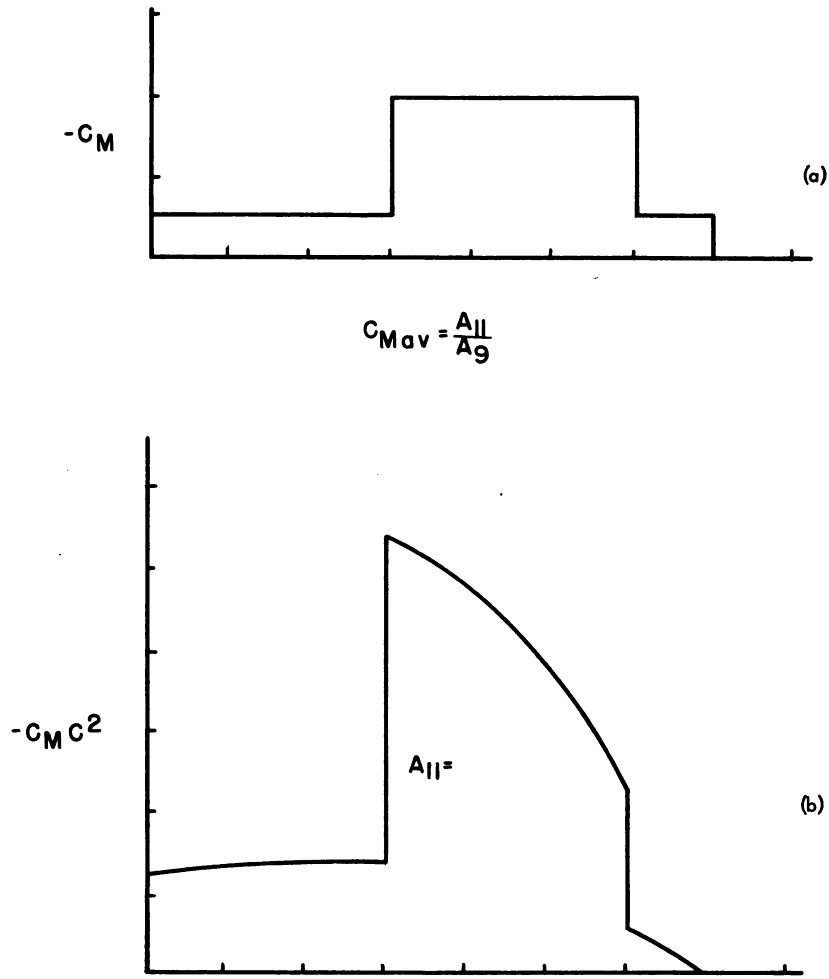


Figure 1–IX. ***Determination of mean effective moment coefficient.***

9. 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 1–Ia under item (9) the local values of the moment coefficient

about the aerodynamic center are entered. These are also plotted as fig. 1–IX, *a*, to illustrate a type of distribution which might exist.

10. Fig. 1–IX (b) is plotted from the values indicated under item (10) of table 1–Ia. The area under this curve divided by the area under curve 1–VIII (c) gives the mean effective value of the moment coefficient for the entire wing panel.
11. In the case of twisted wings a different span distribution exists for each angle of attack. The location of the resultant forces can, however, be determined in the above manner for any known span distribution.

BALANCING LOADS

The balancing loads referred to in this chapter should be computed by a rational method or by a suitable arbitrary method.

Compliance Suggestion

METHOD OF ANALYSIS

a. The basic design conditions 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 speed) flight; in accelerated conditions both linear and angular accelerations 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 the opposite direction and with a force equal to "n" times its weight.

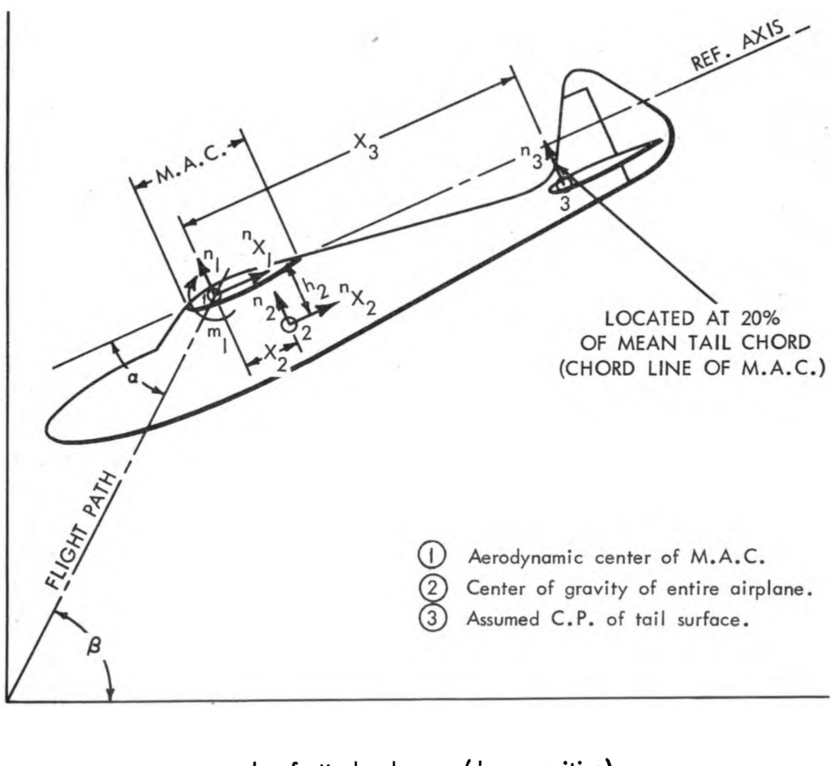
b. If then the resultant moment of the air forces acting on the glider is not zero with respect to the center of gravity, an angular acceleration results. An exact analysis would require the computation of this angular acceleration and its application to each item of mass in the glider. In general, such 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.

Compliance Suggestion

BALANCING THE GLIDER

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

1. It is assumed that the limit load factors specified for the basic flight conditions Chap. 1, p. 9 are wing load factors. A solution is therefore made for the net 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.



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

Figure 1—X. ***Basic forces in flight conditions.***

2. 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 percent 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.
3. In fig. 1–X 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. Where large independent items having considerable drag are present it is advisable to extend the arrangement of forces shown in fig. 1–X to include the additional external forces.

b. As shown in fig. 1–X, 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 mean aerodynamic chord is outlined in Chap. 1, p. 15.

c. 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 1–II. In using fig. 1–X and table 1–II the following assumptions and conventions should be employed:

1. Careful attention should be paid to those footnotes of fig. 1–X which pertain to the sign convention that has been adopted. If known forces are opposite in direction from those shown in fig. 1–X, a negative sign should be prefixed before inserting in the computations. In particular, it should be noted that the vertical distance, h_2, is negative when the wing aerodynamic center is above the CG, and positive when the wing aerodynamic center is below the CG. 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.
2. All distances should be divided by the mean aerodynamic chord before being used in the computations.
3. The chord load acting at the tail surfaces may be neglected.

d. In table 1–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. 1–X. (Condition IV will usually result in the largest balancing tail load.)

- 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.

e. The following explanatory notes refer by number to items appearing in table 1–II:

(4) The wing loadings should be based on the design wing area.

(5) n_1 = limit load factor required for the condition being investigated. (See Chap. 1, p. 3, "Load factors" and Chap. 1, p.8i.)

(8) Determine C_c as specified in Chap. 1, p. 8i.

(10) The value of C_M is specified in Chap. 1, p. 8i.
See also Chap. 1, p.15, in cases involving wing flaps.

(12) The net tail load factor n is found by a summation of moments about point (2) of fig. 1–X from which the following equation is obtained:

$$n_3 = \frac{1}{(x_3 - x_2)} \left[m_1 - n_{x1}\, h_2 + n_1\, x_2 \right]$$

NOTE.—The above explanatory notes apply only when the force arrangement shown in figure 1–X is used. If a different distribution of external loads or a different system of measuring distances is employed, the computations should be correspondingly modified.

CONTROL SURFACE LOADS

In addition to the flight loads specified in the beginning of this chapter, the primary structure should meet the requirements described herein to account for the loads acting on the control surfaces. The following loading conditions include the application of balancing loads 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 should be used unless otherwise specified. Also, see table for multiplying factors of safety required in certain cases.

Compliance Suggestion

UNIT AND CHORD LOADING OF SURFACES

a. The recommendations for the design of control surfaces, as outlined above, are based on the two separate functions of control surfaces: balancing and maneuvering. The requirements also are specified to account for the effects of gust loads.

b. The average unit loading normal to any surface is determined from the design gliding speed V_g as shown by figs. 1–XIII, 1–XIV, and 1–XVI. When dealing with tail surfaces, it is customary to specify the value of the average unit loading for the entire surface, including both the fixed and movable surfaces. The total load, which is equal to the average unit loading multiplied by the area of the entire surface, is then distributed so as to simulate the conditions which exist in flight. In the case of ailerons, flaps or tabs, the value of the average unit loading is usually determined only for the particular surface, without reference to the surface to which it is attached.

c. The average unit loading is assumed to be constant over the span. Also, as shown in figs. 1–XI, and 1–XII, the unit loading at the hinge line is constant over the span.

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

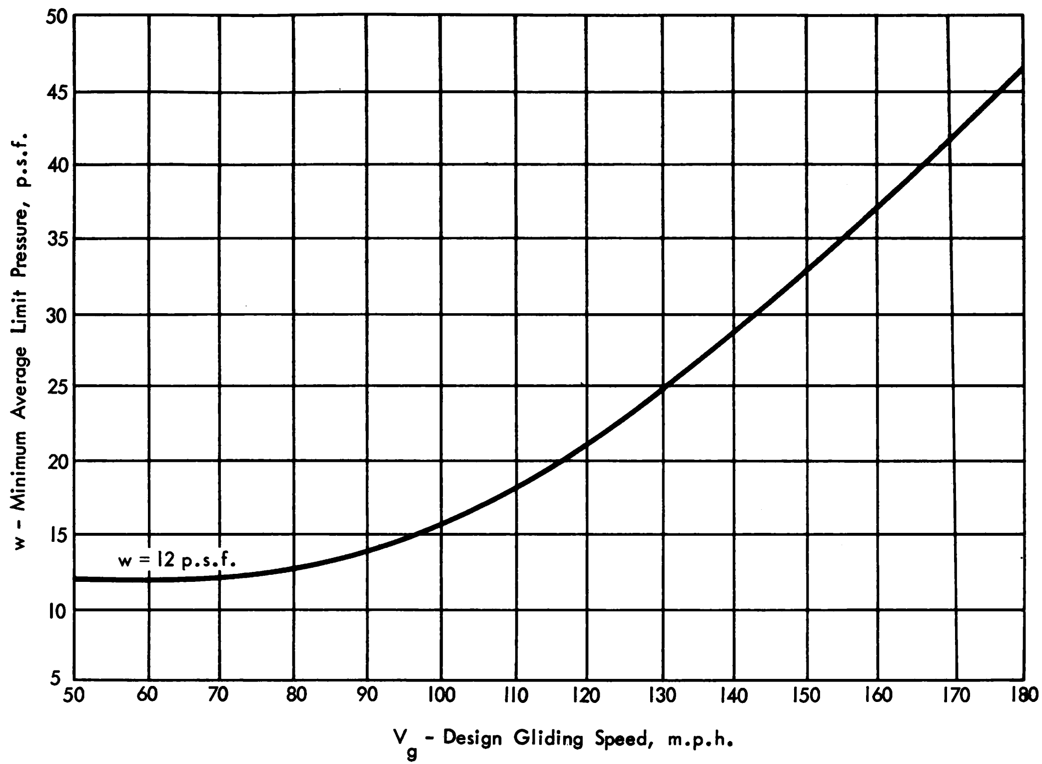


Figure 1–XIII. ***"Maneuvering" horizontal tail loading.***

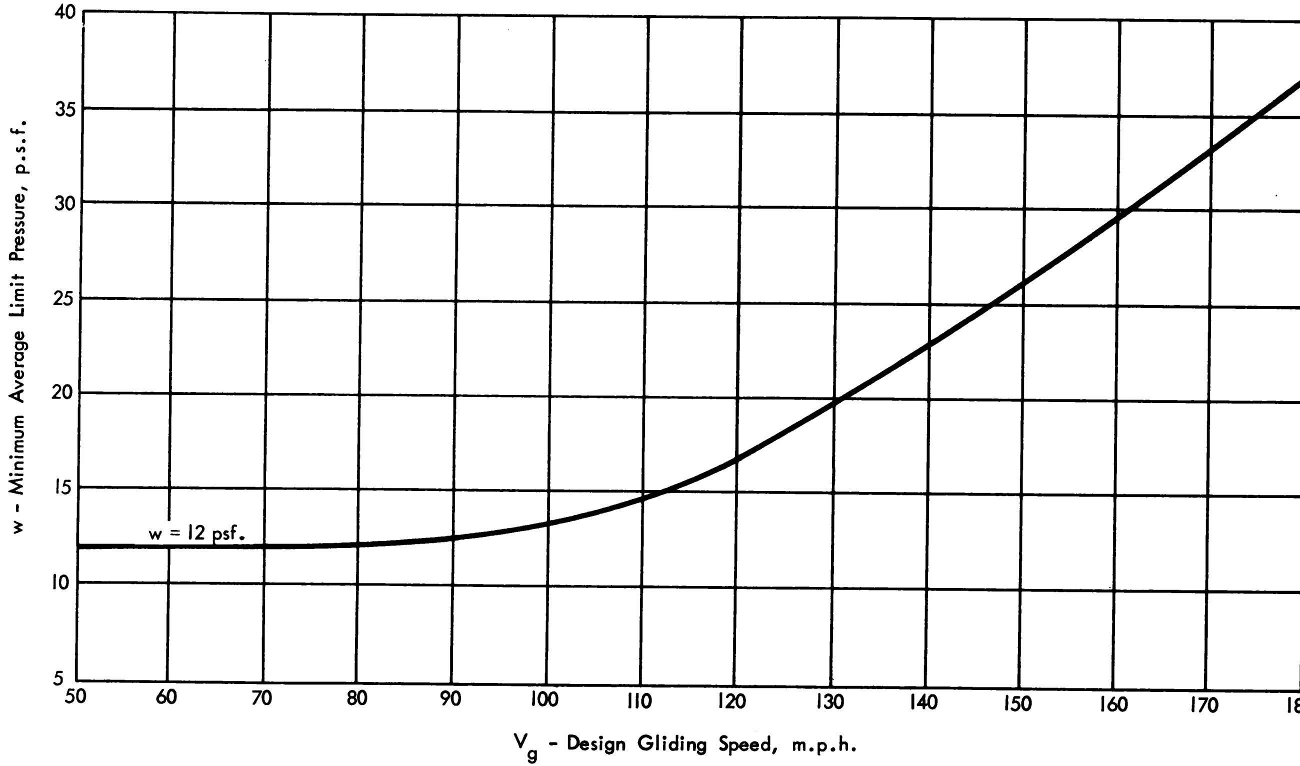


Figure 1—XIV. *"Maneuvering"* vertical tail loading.

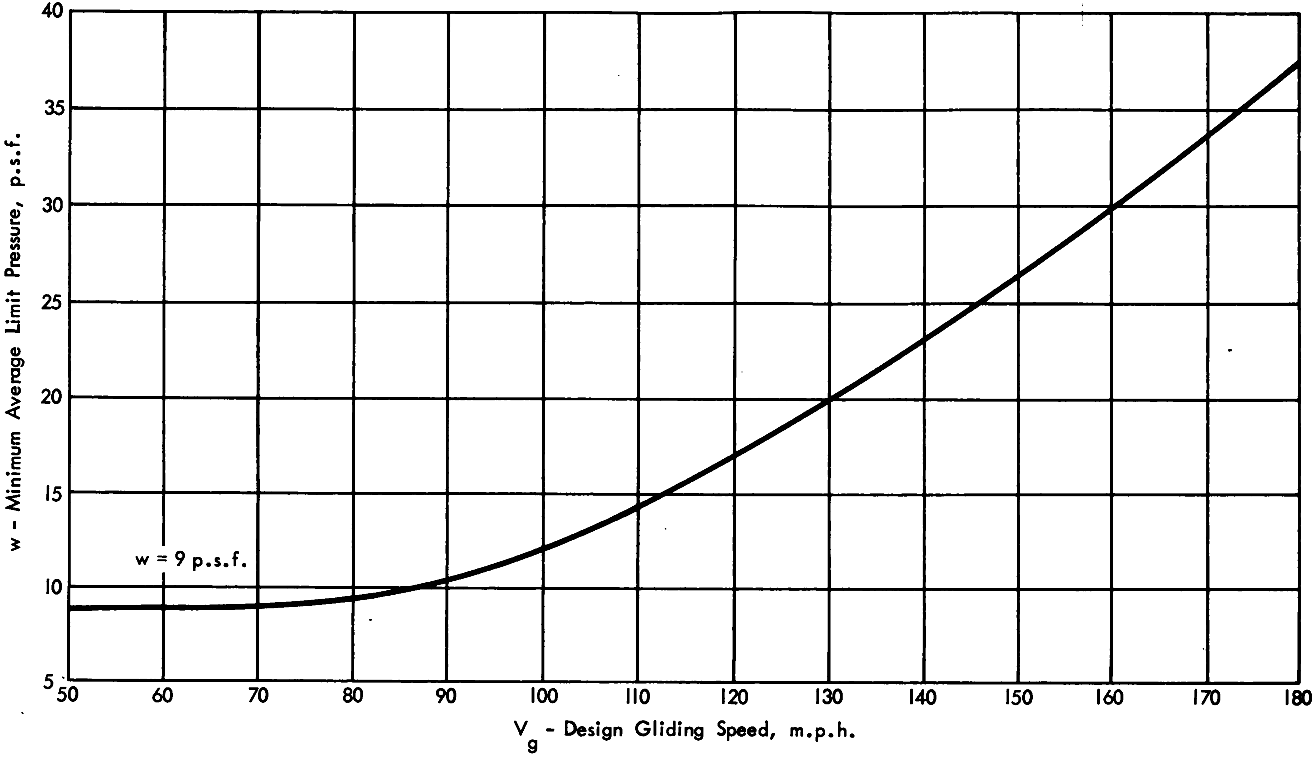


Figure 1–XVI. "*Maneuvering*" *aileron loading.*

HORIZONTAL TAIL SURFACES

Balancing.—The limit load acting on the horizontal tail surfaces should not be less than the maximum balancing load obtained from the loading conditions specified. The load should be distributed in accordance with fig. 1–XI. The balancing loads apply only to the horizontal tail surfaces as the ailerons and the vertical tail surfaces are used only to a small extent for balancing purposes.

The chord distribution shown in fig. 1–XI is intended to simulate a relatively high angle of attack condition for the stabilizer in which very high unit loadings can be obtained near the leading edge.

Loads should also be determined for the most critical center of gravity position of the glider.

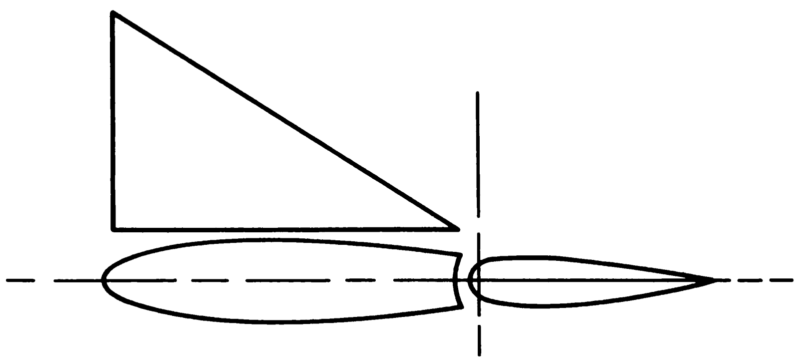


Figure 1–XI. ***"Balancing" and "damping" tail load distribution.***

Maneuvering. (*Horizontal surfaces*).—The minimum average limit pressure specified in table 1–I should be applied in either direction and distributed in accordance with fig. 1–XII.

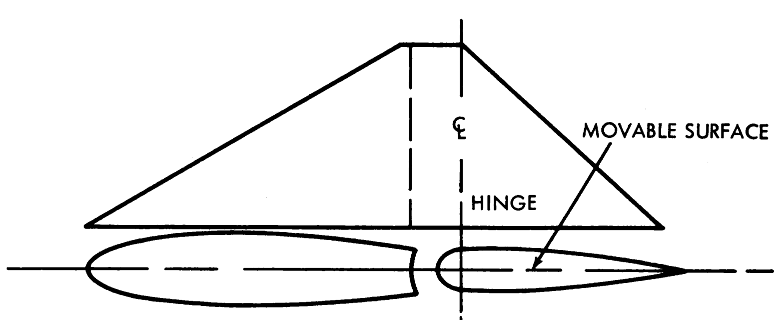


Figure 1–XII. ***"Maneuvering" tail load distribution.***

Compliance Suggestion

DETERMINATION OF MANEUVER LOAD

- The recommendations above are intended to place the determination of such loads on a speed versus force-coefficient basis, to specify values which agree substantially with current practice, and to provide for the effects of increasing glider speeds. The method is designed for application to conventional gliders and, in determining the maneuver loads, the type of service for which the glider is to be used should be considered.
- The values of the average unit loading specified represents loadings which can be attained by deflecting the control surfaces. Higher values may be desirable in certain cases, depending on the purpose of the glider.
- The chord distribution shown in fig. 1–XII represents approximately the type of loading obtained with the movable surface deflected. For tail surfaces, this type of loading is critical for the movable surface and for the rear portion of the fixed surface.

Damping (*Horizontal stabilizer*).—The total limit load acting on the fixed surface (stabilizer) in the maneuvering condition should be applied in accordance with load distribution of fig. 1–XI acting in either direction. The load acting on the movable surface in the maneuvering condition may be neglected in determining the damping loads.

Compliance Suggestion

DISTRIBUTION OF DAMPING LOAD

When a control surface is deflected suddenly the full maneuvering load tends to build up immediately, after which the glider begins to acquire an angular velocity. This angular motion causes the direction of the relative airstream over the fixed surface to change, which causes the air load on this surface to build up in a direction such as to oppose the angular rotation of the glider. This load is concentrated near the leading edge of the fixed surface and is commonly referred to as the damping load. This is considered as a supplementary condition based on the initial maneuvering condition. The damping load is closely related in magnitude to the initial maneuvering load which produces it so that it is convenient to use the latter loading condition to determine the damping load on the fixed surface. To avoid the necessity for a separate analysis for damping loads, the distribution is made the same as for the balancing loads. In the case of the horizontal surfaces, the damping load therefore acts as a minimum limit for the design of the fixed surface and need not be investigated when the balancing load is critical.

Loading for slab tail designs.—The maneuvering horizontal tail loadings shown in fig. 1–XII are applicable to all movable tail surface configurations including the slab tail design. For the balancing tail load distribution the load should be distributed in accordance with fig. 1–XII(1).

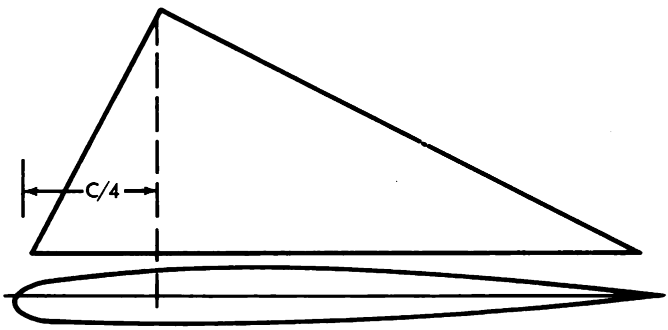


Figure 1–XII(1). ***Balancing load distribution, Slab Tail.***

For the slab tail maneuvering and gust loading condition, the chordwise tail load distribution should be applied as shown in fig. 1–XII(2).

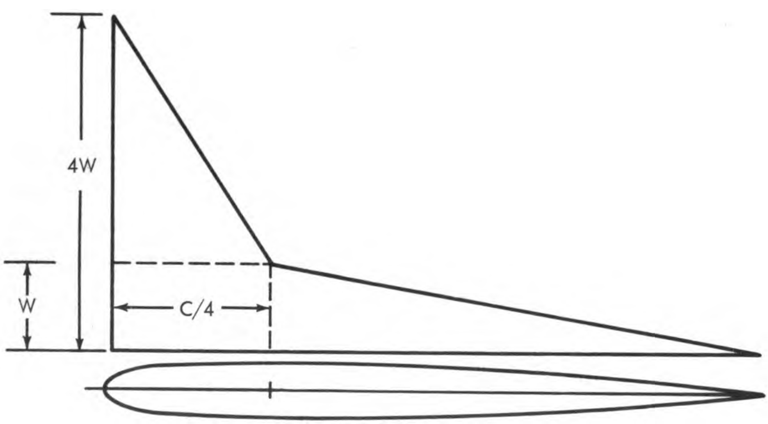


Figure 1–XII(2). ***Maneuvering and gust load distribution, Slab Tail.***

• In calculating the loads for slab tail designs, the effect of tabs, if used, should be accounted for. In addition to the above flight load distributions, an investigation should be made of the ground gust condition imposed on the slab tail. This results in a high trailing edge load similar to that induced by reverse airflow on a wing. (See fig. 1–XII(3.) The design airspeed should be $V = 10\ \sqrt{W/S} + 10$ m.p.h.; $C_L = -0.8$.

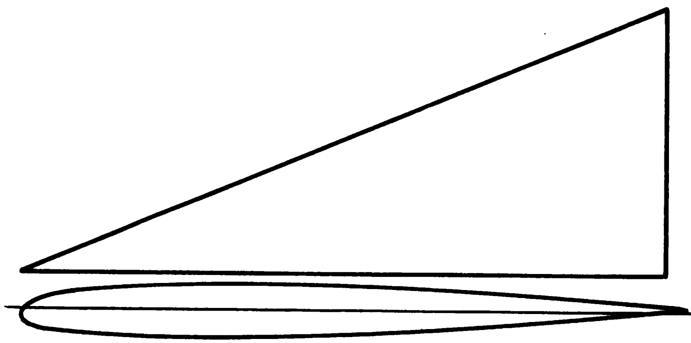


Figure 1–XII(3). ***Ground gust load distribution, Slab Tail.***

VERTICAL TAIL SURFACES

Maneuvering.—The minimum average limit pressure specified in fig. 1–XIV should be applied in either direction and distributed in accordance with fig. 1–XII. (Above comments for horizontal surfaces also apply, in general, here.)

Damping (*Vertical surfaces*).—The total limit load acting on the fixed surface (fin) in the maneuvering condition should be applied in accordance with the load distribution of fig. 1–XI acting in either direction. The load acting on the movable surface in the maneuvering condition may be neglected in determining the damping loads. (Above comments for horizontal surfaces also apply, in general, here.)

Gusts (*Vertical surfaces*).—The minimum average limit pressure should not be less than that corresponding to a 15 f.p.s. sharp-edged gust at the design gliding speed, V_g. The gust should 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 should not be taken as less than 2.0. The chord distribution should simulate that for a symmetrical airfoil, except that the distribution of fig. 1–XI may be used where applicable.

AILERONS

Maneuvering.—The minimum average limit pressure specified in fig. 1–XVI should be applied in either direction and distributed in accordance with fig. 1–XV.

Wing flaps.—Wing flaps should be loaded in accordance with the provisions under "High–lift devices" p. 10. In any case, the average limit prssure should not be less than 9 p.s.f. and should be considered uniformly distributed unless a more rational distribution is used.

Compliance Suggestion
FLAP DESIGN

The critical loading is usually obtained when the flap is completely extended. The above recommendations apply only when the flaps are not used at speeds above a certain predetermined design speed. 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. (See NACA Report No. 824 and Technical Note No. 690.)

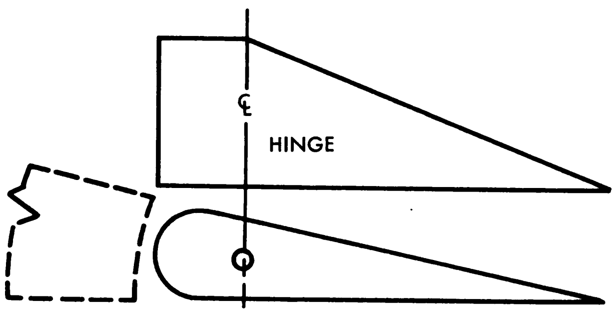


Figure 1–XV. ***Aileron load distribution.***

Special devices.—Special design recommendations should be obtained in connection with the design and analysis of surfaces equipped with unconventional ailerons, auxiliary airfoils and similar devices. Requests for special recommendations should 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.

Compliance Suggestion
DESIGN OF SPOILERS

In lieu of wind tunnel data, it is recommended that the design of spoilers and their attachment structures be premised on the limit loading obtained from the following formula:

$W_{sp} = .0052\ V_{sp}^2$ Where W_{sp} = the limit loading, p.s.f.

V_{sp} = the IAS at which max. operation of the spoilers is assumed, m.p.h.

It should be assumed that the load is uniformly distributed over the surface.

Dive brakes.—The operating mechanism and supporting structure for dive brakes should be designed for critical loads occurring in the brakes extended flight condition, with the brake configuration in any position, from fully retracted to fully extended. When an automatic brake-limiting device is employed, these parts may be designed for critical combinations of airspeed and brake position permitted by the device. Brake installations should be fitted with a "lock open" device to maintain the unit in the extreme open position. NACA Technical Memo No. 926 contains information on dive control brakes. (Wing flaps may be employed as dive brake units.)

CONTROL SYSTEM LOADS

All control systems should be designed using as a basis at least the limit forces hereinafter specified, unless a more rational method is used. Unless otherwise specified, a minimum limit factor of safety of 1.0 and a minimum ultimate factor of safety of 1.5 should be used. See also table 1–III for multiplying factors of safety required in certain cases. See chap. 2, p. 51, "System and Components Tests" for the operation requirements for control systems. The forces in the control system members, cables or push rods operating the movable surfaces should be computed and their effect on the rest of the structure should be investigated and allowed for in the design of such structure.

Compliance Suggestion

DETERMINATION OF LOADINGS

The control forces recommended 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 percent greater than those corresponding to the limit loads specified 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 lb. fore-and-aft
(*b*) Rudder: 100 lb. on one pedal only and
200 lb. on each pedal simultaneously
(*c*) Aileron: 50 lb. laterally or as 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 1.15 need not be applied to the fittings in the control system.

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

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

Aileron systems.—In applying the recommendations for control system loads, it should be assumed that the ailerons are loaded in opposite directions. A control force of 60 pounds should be assumed to act as part of a couple equal to the specified force multiplied by the diameter of the control wheel. Suitable assumptions should be made as to the distribution of the control force between the ailerons.

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

- For nondifferential ailerons, 75 percent 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 percent should be assumed to be resisted by an up aileron, the remainder by the other aileron.
- For differential ailerons, 75 percent 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.

Flap and auxiliary control systems.—In applying the recommendations for control system loads, suitable minimum manual forces should be assumed to act on flap control systems and other similar controls.

Compliance Suggestion

MINIMUM LIMIT FORCES

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

(1) Flaps.................... 50 lb.*
(2) Spoilers.................. 50 lb.*
(3) Hand operated brakes.... 75 lb.*
(4) Foot operated brakes..... 100 lb.*

It should be noted that the flap position that is most critical for the flap proper may not be critical for the flap control mechanism and supporting structure. In doubtful cases the flap hinge moment can be plotted as a function of flap angle for various angles of attack within the design range. The necessary characteristic curves should be obtained from reliable wind tunnel tests.

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

Towing and launching (release mechanism) control systems.—In applying the recommendations for control system loads, a control force of 75 pounds should be assumed to act in a direction corresponding to that normally used by the pilot, and should be applied at the grip of the control handle or lever.

To aid in assuring emergency release it is recommended that a suitable placard or painted instruction be placed adjacent to the glider release hook to indicate that tow rope size (strength) in excess of the limit load of the release mechanism should not be used.

Also, for auto or winch towing, provisions should be made for a back load trip release mechanism to preclude any possibility of failure to release the towline.

GROUND LOADS

The following conditions represent the minimum amount of investigation recommended for conventional landing gear. For unconventional types, it may be advisable 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 proven 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 should be used unless otherwise specified. Also, see table 1-III, for multiplying factors of safety required in certain cases.

Compliance Suggestion

CONVENTIONAL LANDING GEARS

Insofar as the requirements of ground loads are concerned, landing gears will be considered conventional if they consist of:

- A single wheel or double coaxial 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 fig. 1–XVII), or
- 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.

NOTE.—Wing tip skids may be employed if desired.

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

- For wheel type landing gears, the minimum vertical limit load factor should be 4.0. The resultant ground reaction should pass through the center of the axle and should be obtained by combining the vertical component with a rearward acting horizontal component equal to one-quarter of the vertical component.
- For skid type landing gears, the minimum vertical limit load factor should be 5.0. The resultant ground reaction should pass through the center of the skid's contact area, and should be obtained by combining the vertical component with a rearward acting horizontal component equal to one-half of the vertical component.

Compliance Suggestion

DETERMINATION OF LOADING

The loading for this condition is illustrated in fig. 1–XVII. As it is difficult to accurately define "level altitude," any approximately level altitude 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. 1–XVII (*a*).

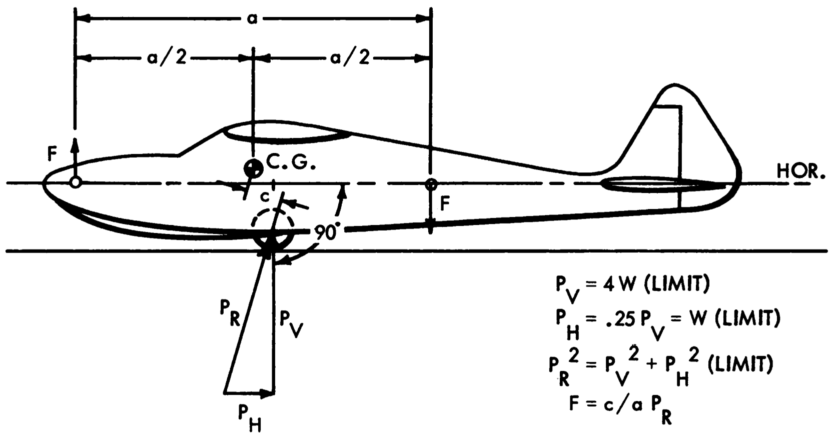
Level landing with side load.—As a separate condition, a limit side load equal to 0.167 times the limit vertical component should be assumed to act at the center of the contact area of the tire or skid, together with the loads specified under "level landing." The side load should be assumed to act in either direction normal to the plane of symmetry.

Compliance Suggestion

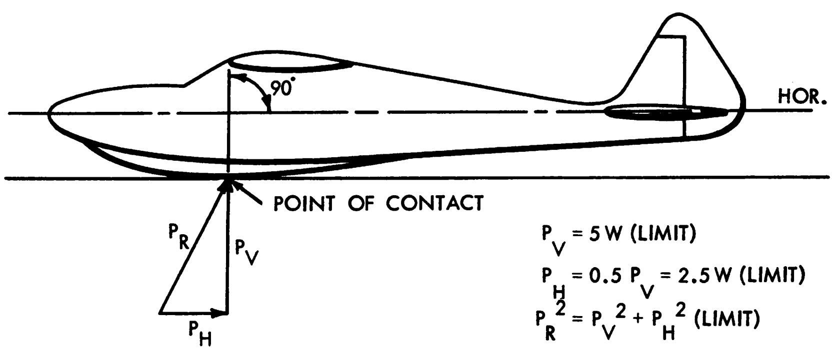
DETERMINATION OF LOADING

The loading for this condition is shown in fig. 1–XVIII. An acceptable method of balancing externally applied rolling moments about the longitudinal axis resulting from the side load is also shown in fig. 1–XVIII (*a*). 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. 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.

Nose-down landing.—For this landing condition, the following assumptions should be made:



(a) For Wheel Type Landing Gears



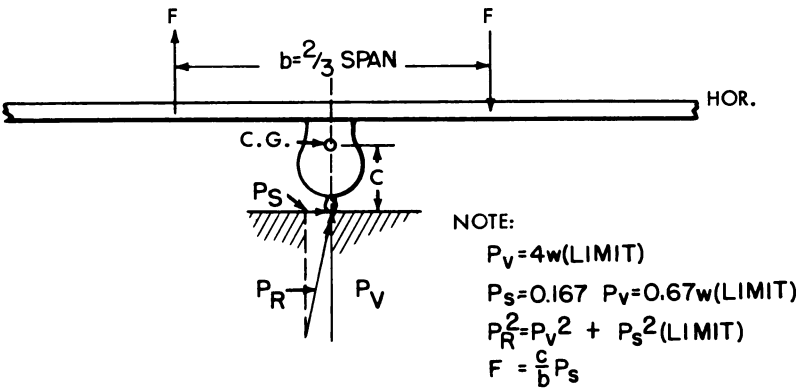
(b) For Skid Type Langing Gears

Figure 1–XVII. ***Level landing.***

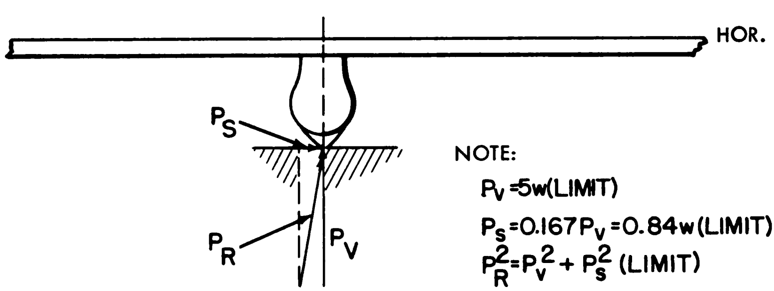
- For wheel type landing gears, the glider should be assumed to make contact on the nose skid or wheel and the main wheel or wheels. The minimum limit resultant inertia force should act at the center of gravity of the glider, should be equal to 4.0 times the weight, and should act forward and downward at an angle of 14 degrees with the vertical. The direction of ground reactions at the contact points should be opposite to the resultant inertia force.

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

NOTE.—These conditions are illustrated in figure 1–XIX.



(a) For Wheel Type Landing Gears

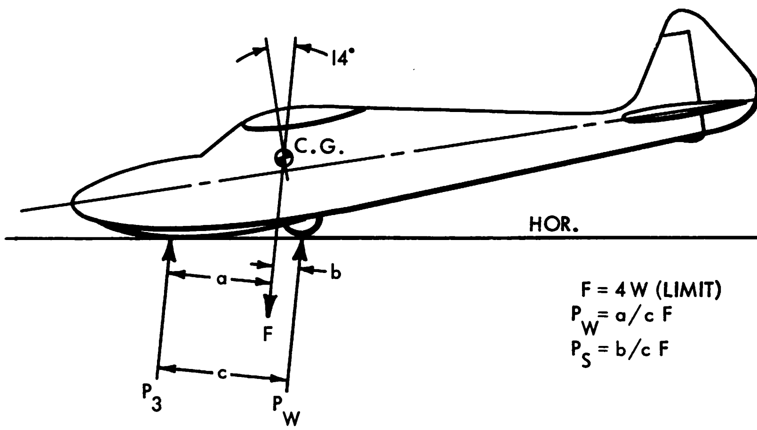


(b) For Skid Type Landing Gears

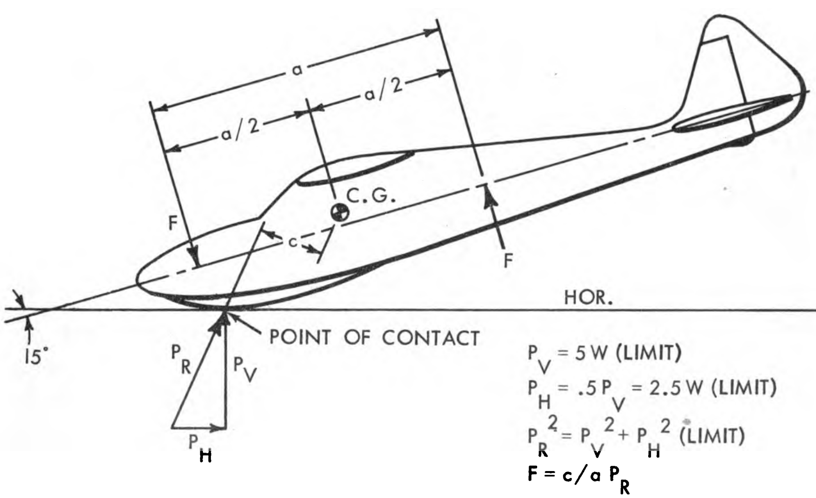
Figure 1–XVIII. ***Level landing with side load.***

Head-on landing.—The forward portion of the fuselage should be capable of resisting an ultimate load of 6.0 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 tenable provided that the specified ultimate load can be resisted without endangering the occupants, assuming safety belts to be fastened. (Also Chap. 3, p. 115.) (Note the 6.0 factor above is ultimate and not a limit load factor.)

Wing-tip landing.—Suitable provisions should be made to provide adequate structure to resist possible severe ground reactions at the wing tips.



(a) For Wheel Type Landing Gears



(b) For Skid Type Landing Gears

Figure 1–XIX. ***Nose-down landing.***

Compliance Suggestion

LOADING CONDITIONS

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, and the ground in a direction parallel to the longitudinal axis. The unbalanced turning moment may be assumed to be resisted by:

- The methods shown in fig. 1–XX, or
- The angular inertia of the glider.

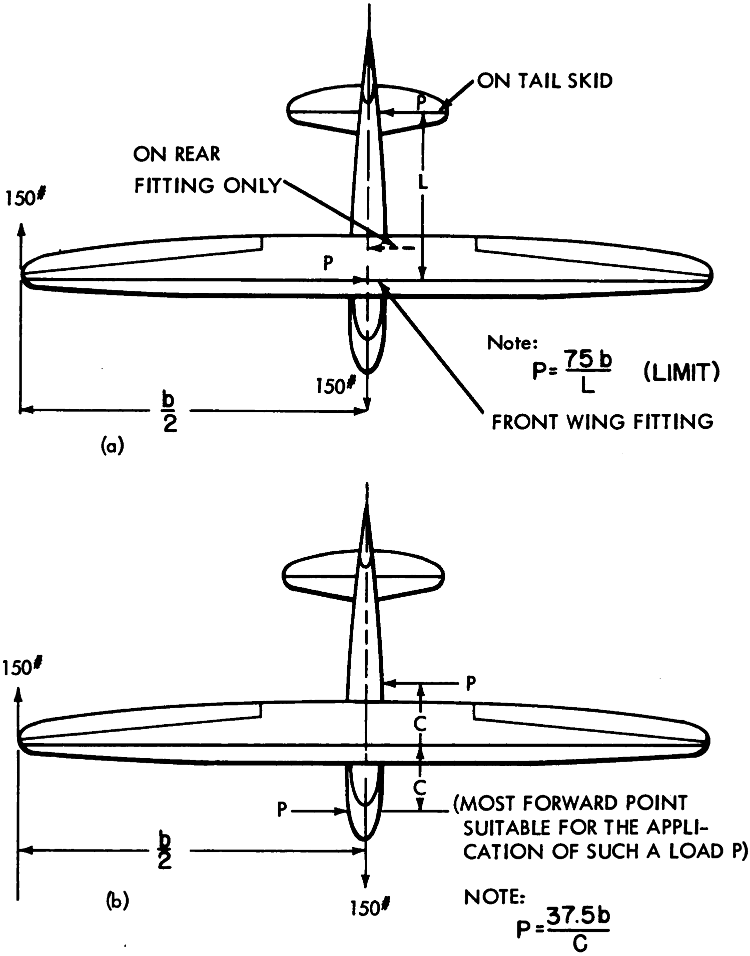


Figure 1–XX. ***Wing tip landing.***

LAUNCHING AND TOWING LOADS

The towing and launching fittings (and/or mechanisms) and that portion of the structure ahead of the main wing fittings should be checked as recommended below. Holding fittings and the structures to which they are attached also should be investigated for pertinent loads. A minimum limit factor of safety of 1.0 and a minimum ultimate factor of safety of 1.5 should be used unless otherwise specified. Also, see table 1–III for factors of safety required in certain cases.

Loads on fittings.—A limit load of 900 pounds or 2.0 times the gross weight, whichever is greater, should be assumed to act in the following separate cases:

- Forward at the towing and launching fitting (or mechanism), and aft at the rear holding fitting.
- At the towing and launching fitting and directed forward and upward at the maximum angle which will afford adequate clearance with the glider; however, it need not exceed an angle of 30 degrees with the longitudinal axis.
- At the towing and launching fitting, and directed forward and downward at an angle of 75 degrees with the longitudinal axis.
- At the towing and launching fitting, and directed forward and sideward at an angle of 30 degrees with the longitudinal axis.

The above loadings are shown in fig. 1–XXI. The effects of these loads need not be investigated aft of the front wing spar.

Wing truss strength.—Unless the strength of the wing in resisting rearward acting chord loads is equal to or greater than the strength in resisting forward acting chord loads, provisions should be made to provide adequate strength of wing drag trusses to resist chord inertia loads developed in glider launching and towing.

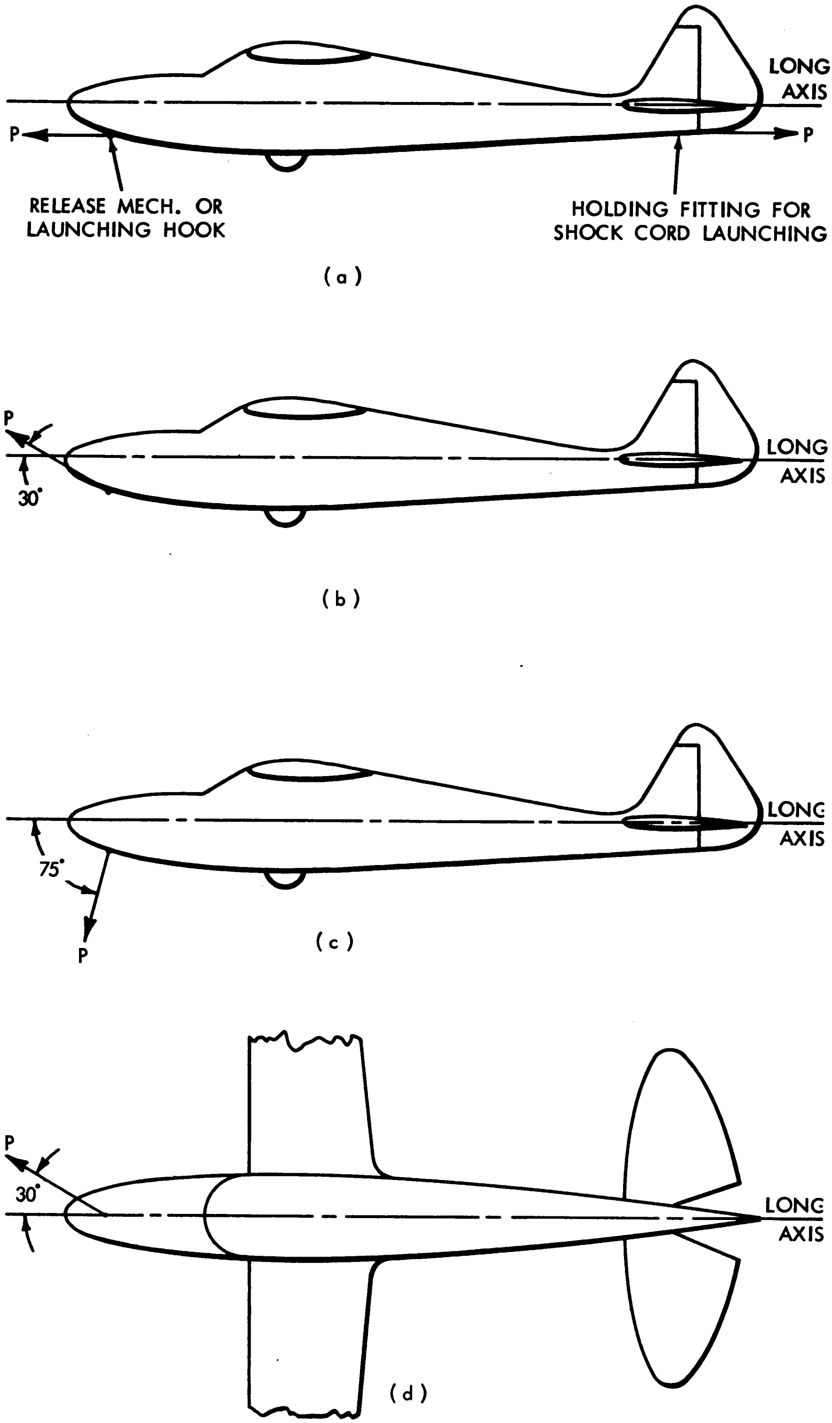
Compliance Suggestion

LOAD FACTOR

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

MULTIPLYING FACTORS OF SAFETY

In addition to the minimum factors of safety specified for each loading condition, the multiplying factors specified in table 1–III and the following paragraphs should be incorporated in the structure. The total factor of safety required for any structural component or part equals the minimum factor of safety 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 1–III. The use of a fitting factor is not considered necessary in cases wherein certain special factors are used, such as those covered by items 1a, 2, 6, 7 and 8 of table 1–III.



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

Figure 1–XXI. ***Launching and towing loads.***

Fittings.—All fittings in the primary structure should incoporate the multiplying factor of safety specified in table 1–III. For this purpose, fittings are defined as parts used to connect one primary member to another and should include the bearing of these parts on the members thus connected. Continuous joints in metal plating and welded joints between primary structural members are not classified as fittings. (See Chap. 3.)

Castings.—All castings in the primary structure should incorporate a multiplying factor of safety not less than that specified in table 1–III.

Parallel double wires.—When parallel double wires are used in wing lift trusses, each wire should incorporate a multiplying factor of safety as specified in table 1–III.

Wires at small angles.—Wire or tie-rod members of wing or tail surface external bracing should incorporate a multiplying factor of safety 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 reaction R.

TABLE 1–III

Additional (Multiplying) Factors of Safety

Item	Additional ultimate factor of safety, j_s	May be covered by Item No.
1. Fitting (except control system fittings)	1.15	1a, 2, 4, 5, 6, 7, 8, 9
1a. Fittings at joints where nontapered pins or bolts are removed during routine dismantling [1,2]	2.00 2.00	2 2
2. Castings	2.00	1a, 7, 8
3. Parallel double wires in wing lift truss	1.05	4
4. Wires at small angles	See p. 43	
5. Double drag truss wires	See p. 43	
6. Torque tubes used as hinges	1.5	
7. Control surface hinges [1]	6.67	
8. Control system joints [1]	3.33	
9. Wire sizes	See p. 44	
10. Wing lift truss (when affected by landing loads)	1.20	

[1] For bearing stresses only.
[2] Includes non-tapered pins and bolts which are removed during routine dismantling.

Double drag trusses.—Whenever double drag trussing is employed, all drag wires should 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.

STRUCTURAL ANALYSIS AS CRITERIA OF STRENGTH

Structural analyses are considered criteria of complete proof of strength only in the case of structural arrangements for which experience has shown such analyses to be reliable. References should be shown for all methods of analyses, formulas, theories and material properties which are not generally accepted as standard. In some cases the acceptability of a structural analysis may depend on the indicated excess strength incorporated in the structure.

Mechanical properties of materials.—The structural analysis should 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, 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.

Compliance Suggestion

COMPUTING OF LOADS AND STRESSES

Acceptable methods for computing the allowable loads and stresses corresponding to the minimum mechanical properties of various materials are given in Bulletin ANC–5 "Strength of Metal Aircraft Elements," obtainable from the Superintendent of Documents, Washington, D. C.

PROOF OF WINGS

For proof of wings by structural analysis only, see "Structural Analysis as Criteria of Strength," p. 46.

Compliance Suggestion

The following points should be considered in the wing analysis:

- *Joint slippage.*—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 percent efficient and also under the assumption that the bending moment transmitted by the joint is 75 percent 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.
- *Bolt holes.*—In computing the area moment of inertia, et cetera, of wood beams pierced by bolts, the diameter of the bolt hole should be assumed to be 1/16–inch greater than the diameter of the bolt.

- *Box beams.*—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 maybe employed.
- *Drag trusses.*—Drag struts should be assumed to have an end fixity, with a coefficient of 1.0, except in cases of unusually rigid restraint, in which a coefficient of 1.5 may be used.

PROOF OF CONTROL SURFACES

Suitable structural analyses of control surfaces will be accepted as complete proof of compliance with the ultimate load requirements provided that the surfaces are of conventional construction. However, proof tests are required to prove 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 table 1–III.

Compliance Suggestion

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" equations. 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.

Compliance Suggestion

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. Methods based on least work or deflection theory offer the only "exact" solution. Approximate methods, however, are satisfactory if based on rational or conservative overlapping assumptions.

PROOF OF CONTROL SYSTEMS

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, however, are required to prove compliance with yield load requirements.

Analysis or individual load tests should be conducted to demonstrate compliance with the multiplying factor of safety requirements specified in Chap. 1 pertaining to control system joints subjected to angular motion.

In addition to the proof tests and analyses, operation tests are also required.

Compliance Suggestion

CRITICAL LOADINGS

In some cases involving special leverage or gear 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.

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 ailerons, wing flap, and tab control systems.

PROOF OF LANDING GEARS

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. For gliders, equipped with an auxiliary powerplant and conventional airplane type arrangement of main and tail or main and nose wheels, compliance should be shown with the landing gear requirements of Civil Air Regulations Part 3.

Compliance Suggestion

WHEELS AND TIRES

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

SHOCK ABSORPTION

There are no definite recommendations regarding the energy absorption characteristics of glider landing gears. On heavy gliders either air wheels or shock absorbing skids should be used.

PROOF OF FUSELAGES

Proof of fuselages by structural analysis only should be in accordance with procedures described above.

PROOF OF FITTINGS AND PARTS

Compliance Suggestion

In the analysis of a fitting it is advisable to tabulate all the forces which act on it for the various design conditions. This procedure will reduce the chances of overlooking some combination of loads which may be critical.

The additional ultimate factor of safety of 1.15 for fittings (table 1–III) is to account for various factors such as stress concentrations, eccentricity and uneven load distribution, which tend to decrease the strength of the fitting.

COMBINED STRUCTURAL ANALYSIS AND TESTS

In certain cases it may be advisable to combine structural analysis procedure with the results of load tests of portions of the structure not subjected to accurate analysis.

Compliance Suggestion

Generally, structural analyses are considered satisfactory for showing compliance of conventional type structures with the ultimate load requirements. However, certain limit load tests and operation tests should be conducted in accordance with the procedures outlined below. For unconventional type structures for which reliable stress analysis methods have not yet been developed, it is usually necessary to resort to combined analysis and limit load tests, or ultimate load tests alone, to show compliance with the ultimate load requirements.

D-nose wing spars, fuselages, and wings of sheet-stringer construction and structures which may be adversely affected by such factors as cutouts and discontinuities, covering behavior including wrinkling, shifting of neutral plane, changes in stress distribution, et cetera, are considered as unconventional structures because they cannot be substantiated by analysis methods alone.

Prior to deciding whether combined analysis and limit load tests or ultimate tests alone are most practicable, it is recommended that an investigation of the effort and cost involved be conducted. The following points should be considered:

1. Components which have been tested to their ultimate strengths should not be in certificated gliders unless it can be shown that no damage including detrimental permanent set has occurred in any part of the structure or that such damage has been properly repaired.

2. Test results should be reduced to correspond to the minimum mechanical properties of the materials used (which requires additional tests).
3. In certain cases, the cost of preparing structural analyses may be appreciably less than the cost of strength tests. This is especially true when extra aircraft components, test jigs, et cetera, must be constructed for test purposes and there are only a relatively small number of gliders involved.
4. In certain cases, components of the gliders may be built sufficiently overstrength without excessive weight penalty, so that tests of the particular component may be conducted without causing failure or permanent set. When ultimate load tests are conducted, the stress analysis need include only the determination of the external loads onto the structure or component. Detailed stress analyses need not be prepared except for fittings. However, if the strengths of the fittings are demonstrated by static tests, the test loads should be equal in magnitude to the fitting design load, including the 1.15 factor for fitting design. Material variation factors need not be used in fitting tests.

LOAD TESTS

Demonstration of compliance with structural loading recommendations by means of load tests only is permissible provided that ultimate and limit tests are conducted to demonstrate compliance.

Compliance Suggestion

STATIC TESTING

In static testing of structural components, no material correction factor is required. However, care should be taken to see that the strength of the component tested conservatively represents the strength of subsequent similar components to be used on gliders to be presented for certification. Included in the test report should be a statement certifying to this fact.

SPECIAL TESTS

If load tests do not show compliance with the particular multiplying factor of safety recommendations, the tests should be supplemented by special tests or analyses to prove compliance with such recommendations.

SYSTEM AND COMPONENT TESTS

Wing ribs.—The strength of ribs should be demonstrated by static tests to at least 125 percent of the ultimate loads for the critical loading conditions.

Control surfaces.—Static tests of the control surfaces to limit loads should be conducted.

Control systems.—Static tests of the control systems to their limit loads should be conducted. Also, operating tests of the control systems loaded to 80 percent of their limit load values specified in Chap. 1 should be conducted to show that the control systems will operate properly when subjected to loads which simulate flight conditions.

Launching and towing release mechanisms.—Static tests of the launching and towing release mechanisms to their limit loads should be conducted. Also, operation tests of release mechanisms should be conducted to show they will function properly when loaded throughout the range from zero to 100 percent of the limit force values specified in Chap. 1 for release mechanisms. These operational tests should include the back load trip release mechanism, if installed. Hardware used in towlines should be of sufficient rigidity to avoid distortion and possible malfunctioning or breakage after repeated use. The above recommendations concerning operation tests do not apply to "open-hook" launching fittings used solely for shockcord type launching apparatus. The strength of the control system actuating the release mechanism is covered in Chap. 1.

Compliance Suggestion

TEST PROCEDURE

These operation tests should be conducted with the fuselage supported and restrained in a manner which will avoid any excessive "springing" of the fuselage nose sections when the test load is suddenly released. Inasmuch as the purpose of the test is to demonstrate the releasing characteristics of the mechanism under load, the fuselage resisting loads may be applied at any convenient points on the fuselage near the release mechanism location.

TEST LOADS, APPARATUS AND METHODS

The determination of test loads, the apparatus used and the methods of conducting the tests should be brought to the attention of the regional office concerned prior to conduct of the tests.

Compliance Suggestion

ULTIMATE LOAD AND DESTRUCTION TESTS

An ultimate load test is used to determine the ability of the structure to withstand its ultimate design load. For conventional type structures, the stress analysis determines this satisfactorily but if the structure is of an unconventional type, an analysis cannot be relied on. In other cases, the manufacturer may not wish to submit a complete stress analysis. Ultimate load tests are, therefore, recommended in such cases as an available method for determining whether the structure will fail before it reaches the required ultimate load.

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. An ultimate load test is not necessarily a destruction test.

In cases wherein the strengths of structures are demonstrated by ultimate load tests in lieu of stress analyses, the proposed test program should be submitted for verification before the tests are conducted. Particular items which should be given special consideration in an ultimate load test are (1) the testing of all members, or portions of the structure requiring excess factors of safety and (2) taking into account the question of material variations from standard specifications and possible discrepancies between the test specimen and drawings.

Compliance Suggestion

MATERIAL TESTS

Standard properties.—Drawings which are to be used as a basis for a type certificate will specify certain minimum guaranteed material properties, usually by reference to existing standard specifications. It is not necessary that the glider applicant substantiate the strength characteristics of standard materials if the purchasing invoices show they are in accordance with standard specifications. In cases where new or unconventional materials are used, special strength tests should be conducted to establish guaranteed minimum properties suitable for design.

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:

1. Ultimate tensile stress
2. Yield point in tension
3. Modulus of elasticity (E)

Further information on strength properties of materials, stress-strain diagrams, et cetera, can be obtained from ANC–5, materials textbooks, et cetera.

Special materials tests.—In many cases, special tests should be conducted 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 repetitive loadings even in cases wherein the stresses are relatively low.

METHODS OF CORRECTING TO STANDARD

It is necessary to know the type of failure which is critical in order that proper correction factors may be used to reduce the results to standard. For a "built-up" structure, failure can occur in many different ways and at different places. In general, it is only necessary to derive correction factors for the particular portion and for the particular mode of failure which occurred. When the total load sustained is 15 percent greater than the "ultimate" load required, no additional material corrections are necessary. When the strengths of fittings are demonstrated by ultimate load tests, material correction factors are not considered necessary since the fitting design factor of 1.15 is believed sufficient to cover it.

The material correction factor 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 mode of failure. In general, it is desirable to obtain a stress-strain diagram for the material specimen. A chemical analysis might be advisable if there is doubt as to the actual material used in the test structure.

TEST PROCEDURES

Jigs.—Tests of tail surfaces, wings and similar components are usually conducted by mounting the surface to be tested onto a specially built jig or framework, using the regular attachment fittings of the unit being tested. The jig 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; otherwise the strength and rigidity of the jig will be imparted to the test thereby leading to erroneous conclusions of excess strength. Special care should be taken to obtain net deflections of the surface tested; that is, the deflections of the jig should be deducted from the total deflection.

Loading schedule.—A loading schedule should be prepared. The schedule shows the load distribution to be used and 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 usually expedient 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 percent) of the ultimate load. In the usual case, such increments will be one-fourth (25 percent) of the required "limit" load, so that the limit value of test load will have been reached at the fourth increment. The ultimate value of test load will then be reached at the sixth increment. After reaching the ultimate load, the size of the increments should be reduced so that the second additional increment will produce 115 percent of the "ultimate" load. However, if the structure should show signs of failing at any time, the loading increments should be accordingly reduced so that the test loads will exceed the failing load by as small a margin as possible.

Supports.—It is advisable to support the unit being tested by means of jacks during the load application. A safety framework or blocking should be provided in all cases so that the structure will not deflect too much 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 such is desirable. 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. See figs. 2–I and 2–II.

Special procedure in limit load tests.—At the start of the test, it is advisable to apply at least a part of one increment and remove it before measuring the initial positions of the deflection stations. After each load increment is applied, the jacks should be lowered for a period of at least 1 minute before deflection readings are taken. When the total "limit" load has been loaded onto 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.

Special procedure in ultimate load tests.—The procedure outlined for the limit load test described in the preceding paragraph should be followed until the limit load test is completed. It will be noted that the ultimate load test, if conducted first, would not permit the determination of the permanent set caused by the "limit" load. After the limit load test, the loading should be continued beyond the limit load value 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 ultimate, it is permissible to reinforce the failed portion. When that is practicable, the test should then be resumed. The details of reinforcement should be carefully noted. If the material correction factors will be small, it is not necessary to proceed to the 115 percent overload. If the test specimen cannot be used after test, it is desirable to continue the test to destruction, that is; through to complete failure. 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.

Special procedures when ultimate load tests are conducted in lieu of stress analysis.—In such cases the procedure is the same as for ultimate load 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 should be obtained and tested.

Check of test structure.—In most cases, a conformity check is made of the test specimen against the drawings and other design data used as a basis for certification.

In the case of ultimate load tests in lieu of stress analysis, a particularly thorough conformity check is made. After the test, the portions that failed usually are further checked for dimensions and strength properties.

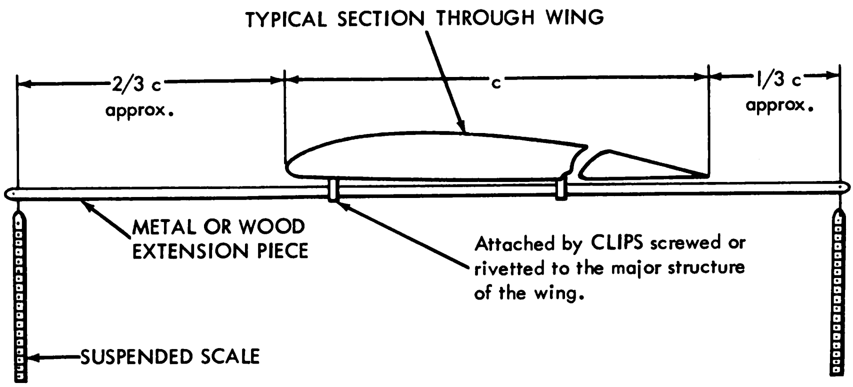


Figure 2–1. Chordwise extension piece to magnify wing torsional deflection readings.

TEST REPORT

In all cases the manufacturer making a test is required to submit a complete report covering details of tests. The report should include photographs or drawings of the test setup and the test specimen; photographs of failed parts or sections; records of deflections and readings taken; date of test; identification number of report; serial and model number of glider and signature of responsible witnesses and/or test personnel. In addition, the following points should be covered when applicable:

- Substantiation by references or computations of the selection of critical test conditions and loadings.
- Loading schedule used in the test.
- Description of test setup, with reference to drawing numbers.
- Chronological account of the test procedure and events.
- Deflection readings, preferably in tabular form. Also jig deflections.
- Disposition of tested structure.

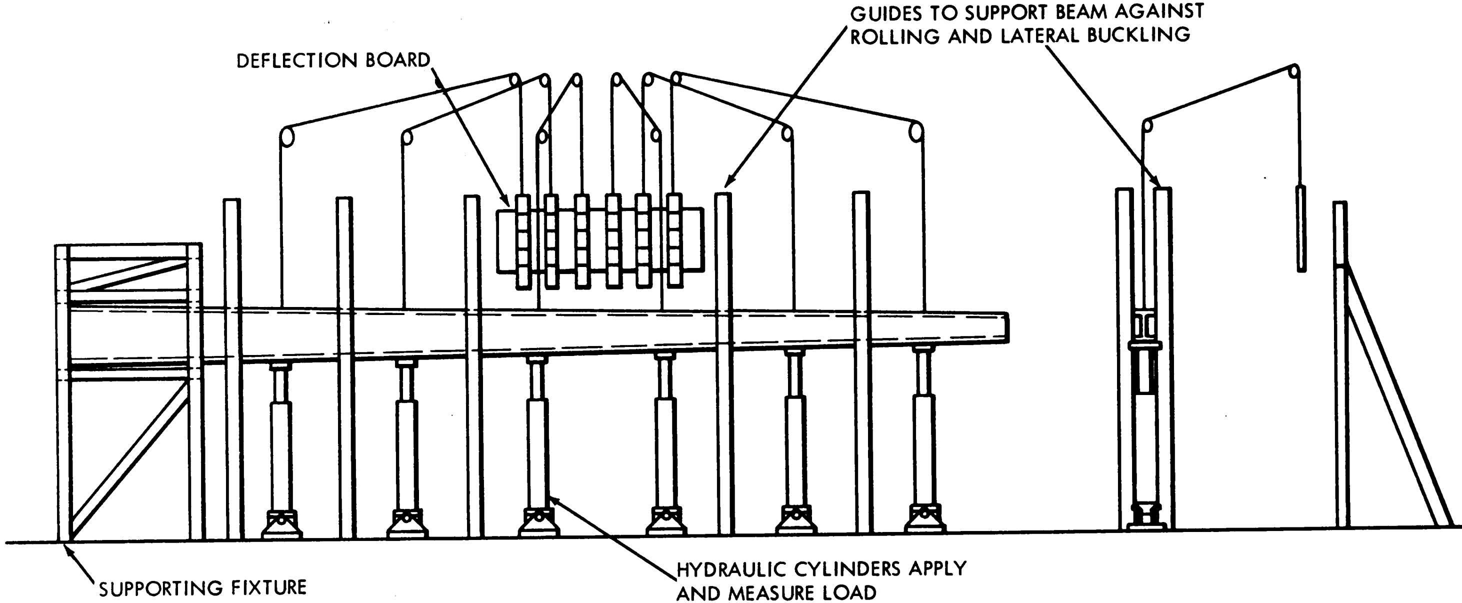


Figure 2–11. ***Example of use of single deflection board.***

WING TESTS

Rib tests.

A. Selection of ribs.—

1. *Wing of uniform chord.*—At least two ribs should be tested for each of the two loading conditions.
2. *Tapered wing.*—Considerable judgment should be used. Where a series of ribs of varying sizes is involved, the largest rib of each series should be selected for testing.
3. Where tapered wings incorporate either two or three similar series of ribs, if two series are used, the tests should be made on two each of the largest rib in each series, or a total of four ribs. If there are three series, it would probably be sufficient to test two identical ribs of the largest size used and one each of the largest, in each of the other series.
4. Usually, no more than a total of four ribs need be tested. However, in all cases, it is essential that representative ribs be tested.

B. Test loadings.—

1. The rib tests should at least cover the positive high angle of attack condition (Condition I) and a medium angle of attack condition. The total load to be carried by each rib should equal 125 percent 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 factor for Conditions I and III.
2. 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 m.p.h. If this is not applicable, 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 usually is advisable in the case of a rib which appears to be weak.
3. The following loadings are recommended for two-spar wing construction when the rib forms a complete truss between the leading and trailing edges.
   a. For the high angle of attack flight condition, ribs of chord length greater than 60 inches should be subjected to 16 equal loads at the load points given in tables 2–IV or 2–V. In order to determine which set of load points is applicable to the particular airfoil used, it is first necessary to determine the following airfoil characteristics:
   (1) PD (Pressure Distribution) classification—this is expressed by a capital letter followed by a two digit

number such as C 10, B 11, D 12, et cetera. For the present purpose, only the number portion of the classification need be considered.

(2) C_{mac}—moment coefficient about the aerodynamic center.

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

These characteristics are readily obtainable for most airfoils from N.A.C.A. Technical Reports Nos. 610 and 628. For airfoils in the 10 or 11 classification, the load points should be taken from table 2–IV, using the line corresponding to the C_{mac} value of the airfoil. (Table 2–IV should also be used for ribloading points in cases where the P.D. classification is not available, or in cases where the designer does not wish to determine it.) For airfoils in the 12 classification, the load points should be taken from table 2–V, using the line corresponding to the C_{mac}, and the camber of the airfoil. In cases where the actual position of load number 1 is less than ½ inch from the leading edge, loads 1 and 2 may be combined into a single load (of twice the unit value) and applied at their centroid. For ribs having a chord of less than 60 inches, 8 equal loads may be used, their arrangement being such as to produce shears and moments of the same magnitude as would be produced by the application of 16 equal loads at the locations specified above.

b. For the medium angle of attack condition 16 equal loads should be used on ribs of chord 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. When 16 loads are used, they should be applied at 8.34, 15.22, 19.74, 23.36, 26.60, 29.86, 33.28, 36.90, 40.72, 44.76, 49.22, 54.08, 59.50, 65.80, 73.54, and 85.70 percent of the chord. When 8 loads are used they shall be so arranged as to give comparable results.

4. When the lacing cord for attaching the fabric passes entirely around the rib, all of the load should be applied on the bottom chord.
5. When the covering is to be attached separately to the two chords of the rib, the loading specified in paragraph c below should be modified so that approximately 75 percent of the ultimate load is on the top chord and 50 percent on the bottom, the total load being 125 percent of the ultimate load.
6. The aforementioned load distribution also is applicable for ribs attached directly to the spar of single spar wings.

TABLE 2-IV—*Rib Load Points for High Angle of Attack*

PD Classification	(2) Camber	$C_{m_{a.c.}}$	Load points in percent chord															
			1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
10 and 11 [1]	Any value	0 to -.019 [3]	.5	1.9	3.4	5.2	7.2	9.6	12.4	15.5	19.0	23.4	28.2	33.2	40.3	48.2	72.0	90.0
		-0.020 to -.039	.5	2.0	3.5	5.6	8.0	10.5	13.4	16.8	20.8	25.2	29.8	35.3	42.1	50.2	72.0	90.0
		-.040 to -.059	.7	2.0	4.0	6.3	8.8	11.4	14.8	18.5	22.8	27.2	32.7	38.0	44.8	52.8	72.0	90.0
		-.060 to -.079	.8	2.6	4.5	6.7	9.5	12.7	16.2	20.0	24.2	28.8	34.0	40.0	46.7	54.4	72.0	90.0
		-.080 to -.099	.8	2.8	5.0	7.5	10.5	13.7	17.4	21.2	25.7	30.3	35.5	41.5	47.8	55.4	72.0	90.0
		-.10 or greater	.8	3.0	5.5	8.2	11.4	14.8	18.6	22.7	27.3	32.2	37.5	42.9	49.6	57.5	72.0	90.0

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

TABLE 2-V—*Rib Load Points for High Angle of Attack*

PD Classification	Camber	$C_{m_{a.c.}}$	Load points in percent chord															
			1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
12	0.0 to 2.9	0.00 to -.0199	.7	2.3	3.9	5.8	7.9	10.4	13.1	16.3	20.1	24.4	28.9	34.3	41.0	49.0	72.0	90.0
		-.02 to -.0399	.8	2.5	4.5	6.4	8.7	11.3	14.1	17.5	21.3	25.5	30.5	36.2	43.2	51.1	72.0	90.0
	3.0 or greater	0.00 to -.0199	.8	2.5	4.5	6.5	8.7	11.0	13.5	16.4	19.7	23.6	28.0	33.5	39.7	47.7	72.0	90.0
		-.02 to -.0399	.9	2.6	5.0	7.2	9.6	12.0	14.7	17.9	21.5	25.6	30.1	35.5	41.8	49.7	72.0	90.0

C. Test methods.—Standard procedure.—

- The ribs should be attached to short spar sections to simulate conditions 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. See fig. 2–III.
- 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 ½-inch, and, for metal covered wings, should be spaced at least 8 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 8 inches, whichever is the greater. In any case, the lateral supports should simulate, as slosely as practicable, the actual conditions represented in the glider.
- 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 1-inch long to distribute the load at the loading points.

WING PROOF AND STRENGTH TESTS

A. Test loads.—The loads to be used and their distribution over the wing will depend upon the particular condition for which the test is conducted. In general, when tests are made to prove the strength of the entire wing, four tests should be conducted 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, involving cantilever wings, the wings are sufficiently symmetrical structure wise that only those tests involving the maximum loadings in the positive direction need be conducted.

B. Test methods.—

1. *General*.—The procedure outlined elsewhere for the preparation of the manufacturer's test report is also applicable to reports of wing tests.
2. *Mounting*.—For the tests, the wing will usually be mounted on a jig. The method of attachment should simulate the method of attachment in the actual glider. In particular, the method of attachment should be such that the strength and rigidity of the jig is not imparted to the wing thereby leading to erroneous conclusions regarding the strength of the wing adjacent to the attachment fittings. See figs. 2–IV through 2–XI.

3. *Chord component.*—Usually, the chord of the wing will not be horizontal 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.
4. *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.

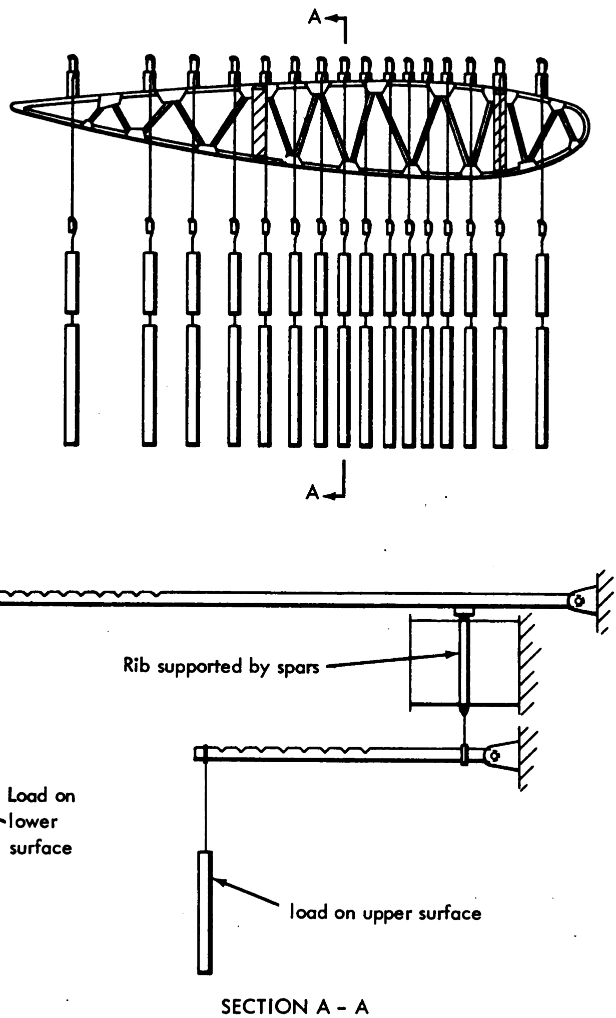


Figure 2–III. ***Example of setup for test of wing ribs.***

C. Rigidity.—At present, the Federal Aviation Agency has no established criteria for permissible bending deflection in wings.

D. Test report.—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. The bending deflections should be separated from the torsional deflections and separate curves should be plotted for each.

CONTROL SURFACE TESTS

Tail surface and aileron tests.—

A. Test loads.—

1. *Kinds of tests.*—Tests on tail surfaces and ailerons may be either limit load tests or ultimate tests. In either case, the loading distribution itself (not the total load) is the same.
2. *Load distribution-tail surfaces.*—The horizontal and vertical tail surfaces should be tested for both of the conditions illustrated in figs. 1–XI, and 1–XII. The magnitude of the load in each case depends upon which of the specified conditions is critical.
3. *Ailerons.*—Ailerons should be tested for the load distribution shown in fig. 1–XV.
4. *Balance area.*—The ordinate of the loading curve at the hinge line is constant over the span. It is not affected by the chord of the movable surface. The unit loading on the balance area, if any, is the same as at the hinge line as shown in figs. 1–XII and 1–XV.

B. Test methods (see figs. 2–XIII and 2–XIV.)

1. *Horns.*—Control surface tests should include the horn of the arm to which the control system is attached. Control surface horns should be rigidly held by tubes or straps instead of flexible cables, unless the test is purely a control system test. Cable usually stretches excessively, and the resultant angular deflection of the control surface may disturb the position of the loading bags or possibly cause them to tumble onto the floor.
2. *Mounting.*—The control surfaces may be mounted on the glider 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 simulate the attachment conditions applying in the glider.

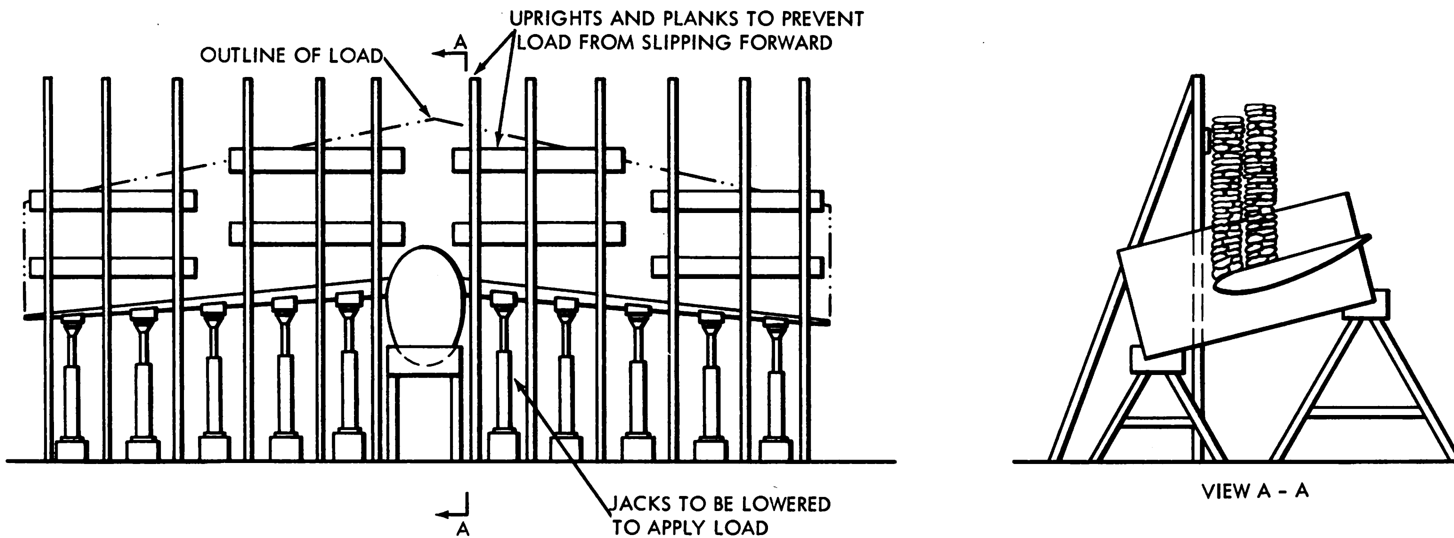


Figure 2–V. ***Setup for test of complete wing and its attachment to the fuselage.***

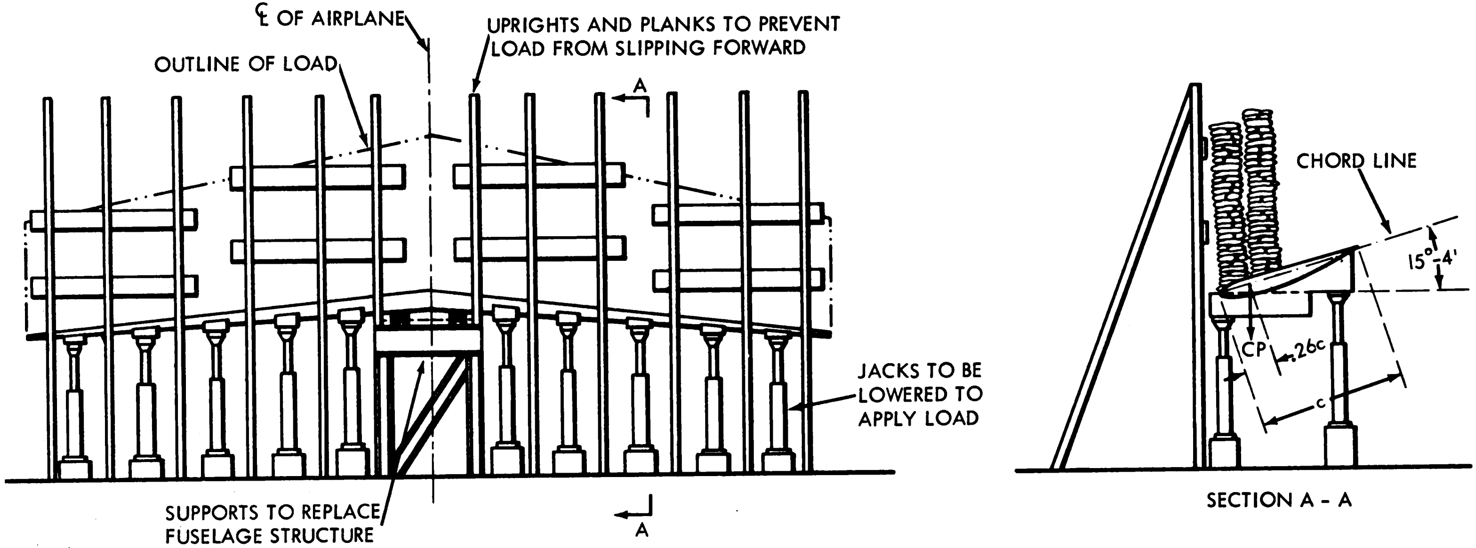


Figure 2—VI. ***Typical wing test setup using a fixture in lieu of fuselage.***

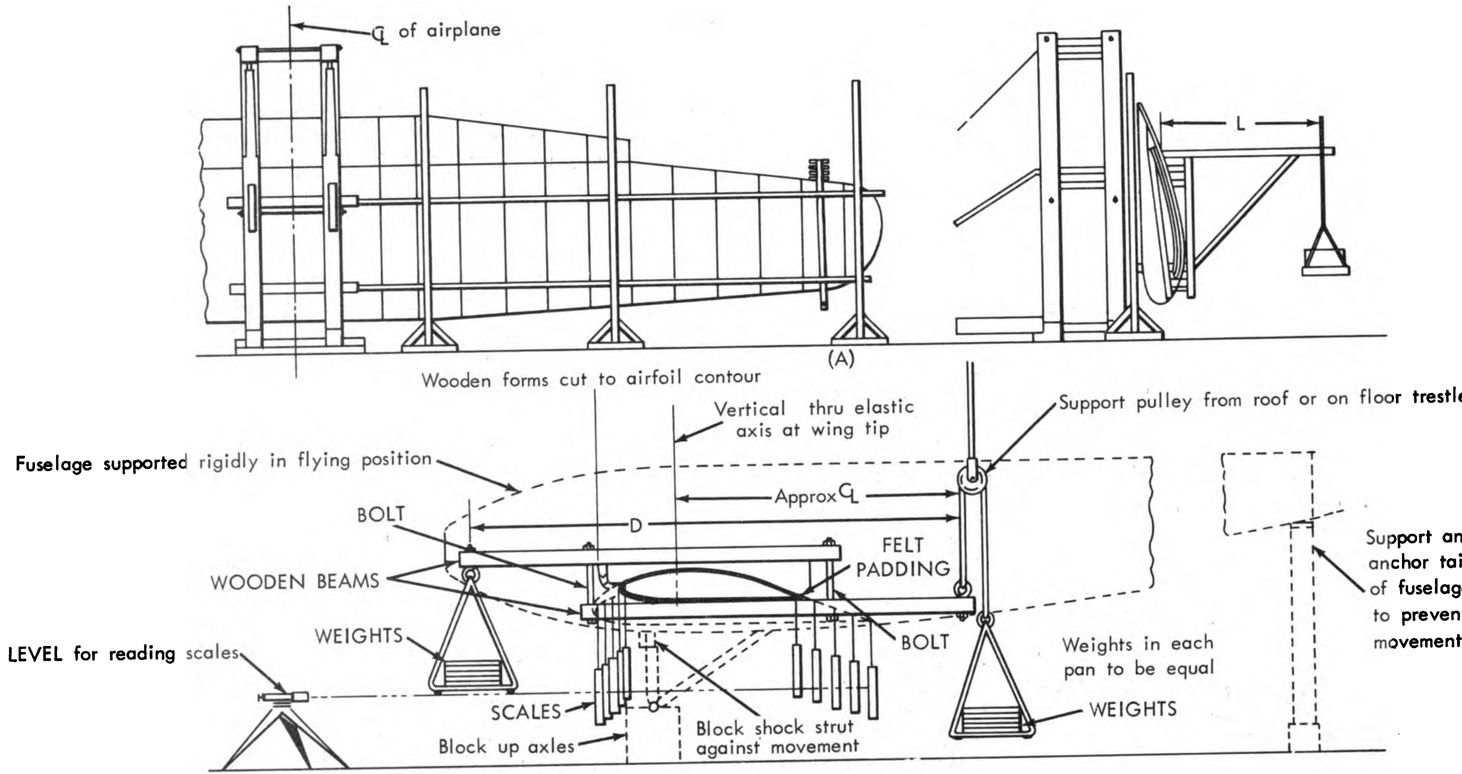
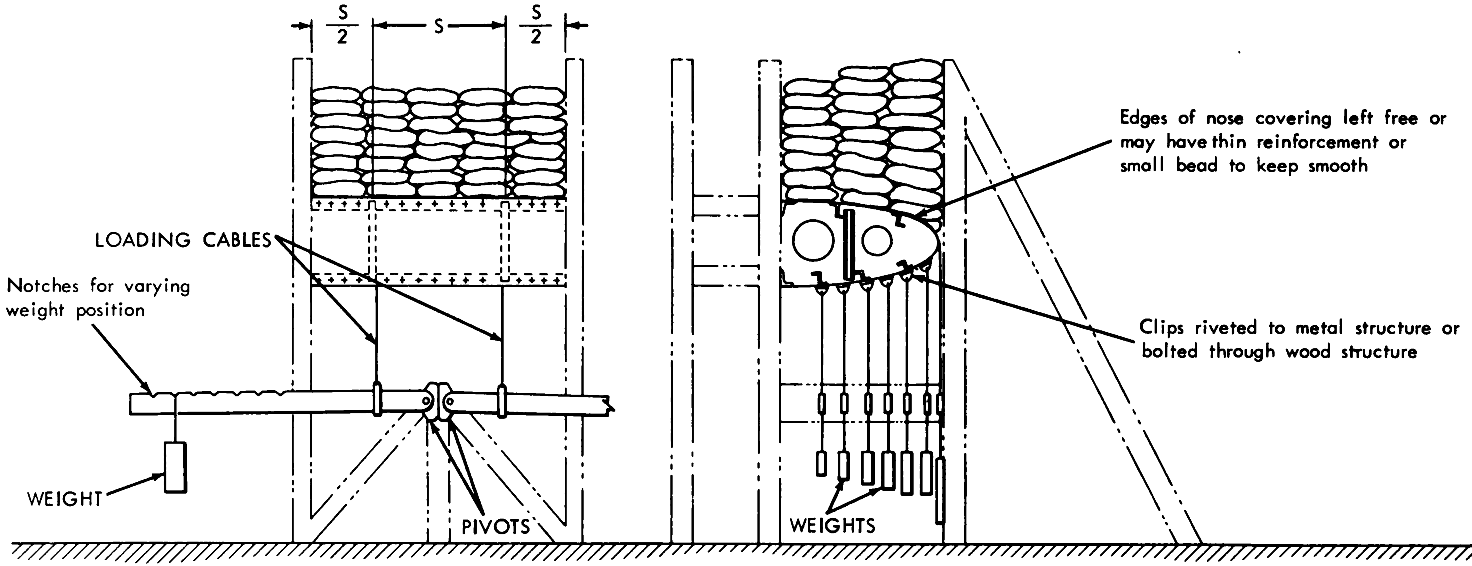


Figure 2–VII. *Setup for torsional test of wing.*



NOTES
1. S=Airplane normal nose rib spacing for the chosen critical station
2. Each nose rib same as the chosen most critical one
3. For some designs shot bag loading of lower surface only will be acceptable

Figure 2–VIII. ***Typical setup for leading edge structure static test.***

3. *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. The final coats of dope can be applied subsequent to the tests.

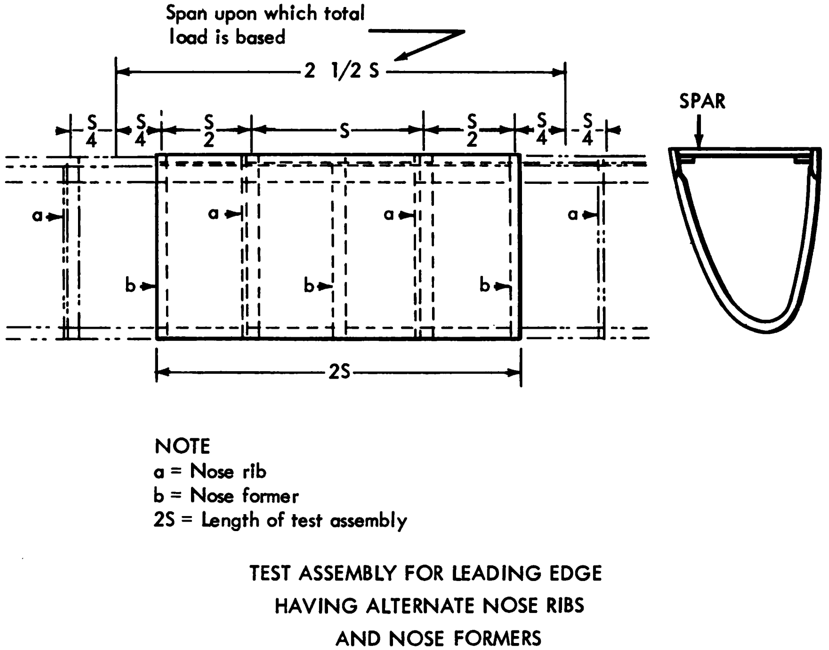


Figure 2–IX. ***Test assembly for leading edge having alternate nose ribs and nose formers.***

4. *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.
5. *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.
6. In the case of a test for the so-called "balancing" condition, it should be noted that all the load acts in one direction on the fixed surface.

C. Test report.—In all cases the manufacturer making a test is required to submit a complete report covering details of tests. The report should include photographs or drawings of the test setup and the test specimen; photographs of failed parts or sections; records of deflections and readings taken; date of test;

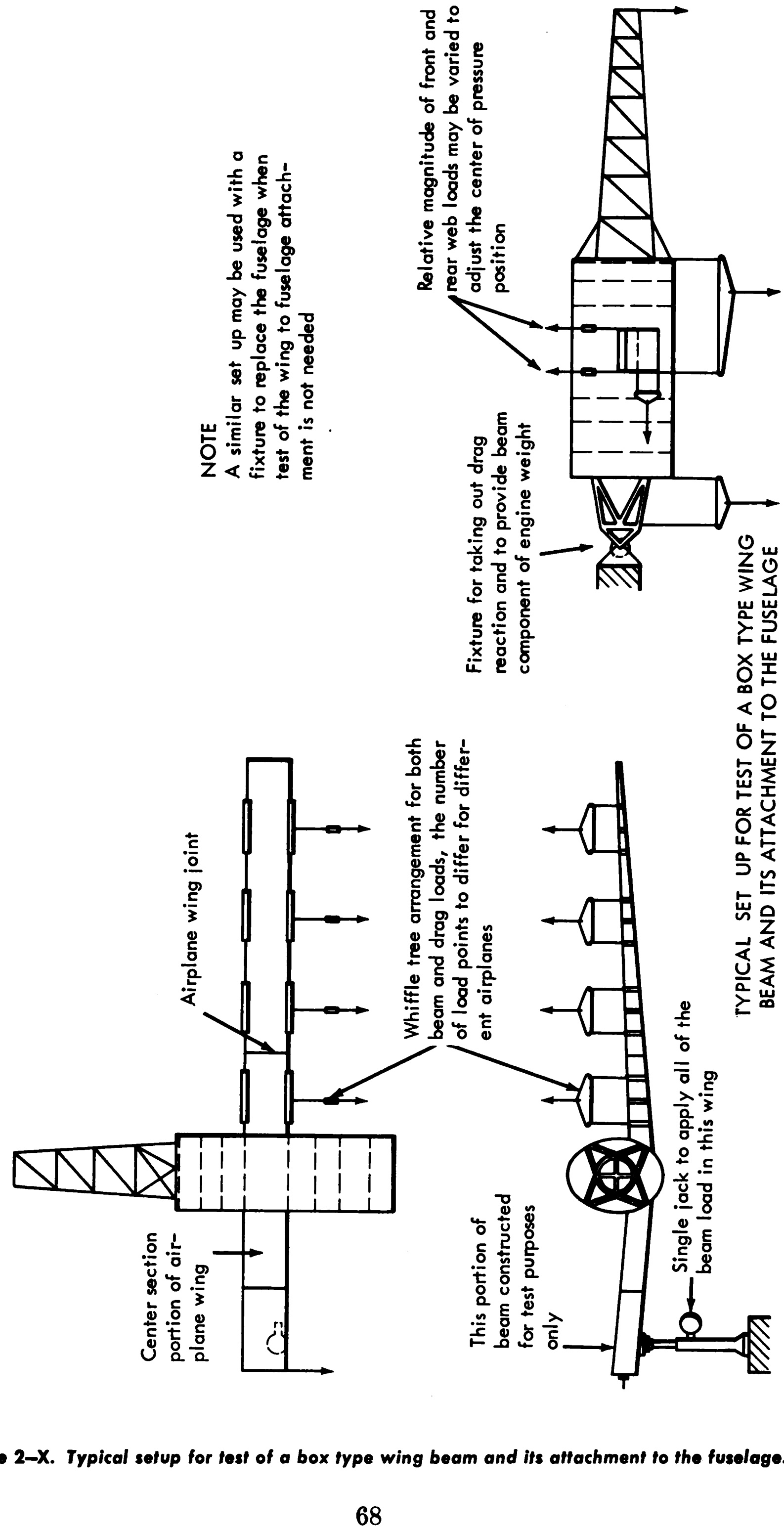


Figure 2–X. ***Typical setup for test of a box type wing beam and its attachment to the fuselage.***

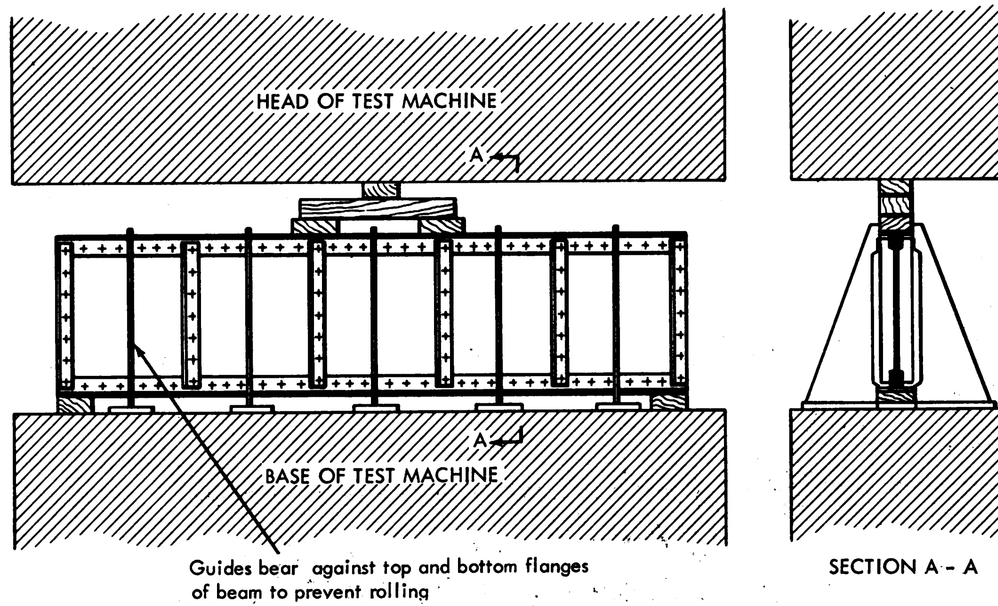


Figure 2—XI. ***Typical setup for test of a short section of wing beam.***

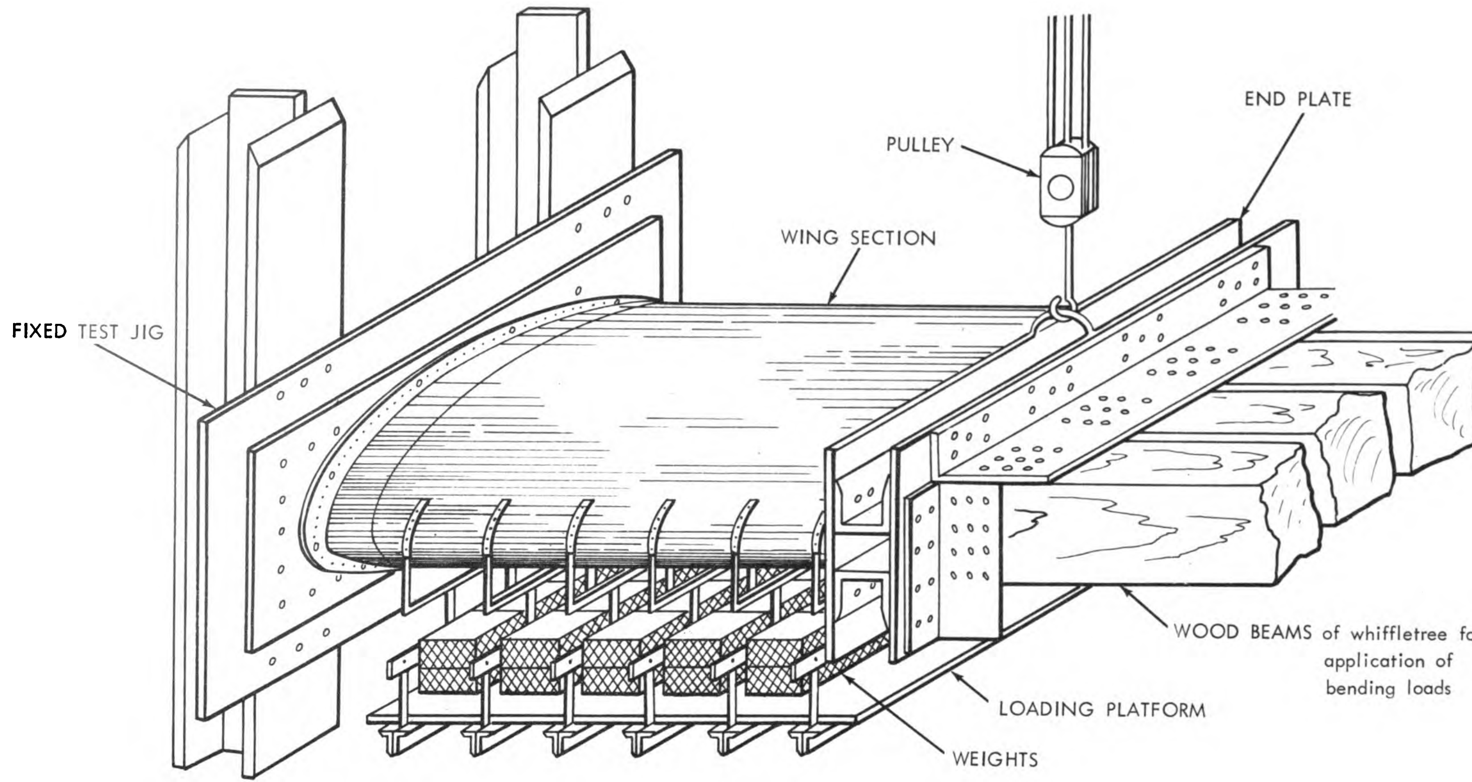


Figure 2–IV. ***Typical test setup of wing structure.***

identification number of report; serial and model number of glider and signature of responsible witnesses and/or test personnel. In addition, the following points should be covered when applicable:

1. Substantiation by references or computations of the selection of critical test conditions and loadings.
2. Loading schedule used in the test.
3. Description of test setup, with reference to drawing numbers.

Flap tests.—

A. Test methods and loads.—

Wing flaps should usually be tested on a jig because of the difficulty of mounting a complete wing and flap assembly in the 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 the ailerons, rudder and the elevator. 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.

B. Test report.—In all cases the manufacturer making a test is required to submit a complete report covering details of tests. The report should include photographs of drawings of the test setup and the test specimen; photographs of failed parts or sections; records of deflections and readings taken; date of test; identification number of report; serial and model number of glider and signature of responsible witnesses and/or test personnel. In addition, the following points should be covered when applicable:

1. Substantiation by references or computations of the selection of critical test conditions and loadings.
2. Loading schedule used in the test.
3. Description of test setup, with reference to drawing numbers.

Control system tests.—

A. Test loads.—

1. *Operating test.*—The controls should be operated from the pilot's seat when the system is sustaining the "limit" loads specified in Chap. 1.
2. *Strength tests.*—The test loads used in a limit load test or an ultimate test are premised either on the control surface design loads or on the control system design loads depending on the particular limiting conditions. This is explained in Chap. 1.

B. Test methods.—

1. *Method of load application.*—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:

   (a) 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 surfaces are then loaded. This method has the disadvantage that stretching of the control system cables may result in movement of the surfaces thereby causing the loading bags to shift in position or possibly tumble onto the floor. The control systems for adjustment devices such as the stabilizer and the trailing edge tabs, and wing flaps are usually of irreversible type such that no additional fixation should be necessary during the tests. The type of blocking used at the control wheel, the control stick, or the rudder pedal should be such that it can readily be removed and replaced with the system under load, and so that when removed it will not interfere with limited movement of the controls. Simple blocking of the control stick and tying the rudder pedal in position by a wire or cable, are usually satisfactory methods of accomplishing this.

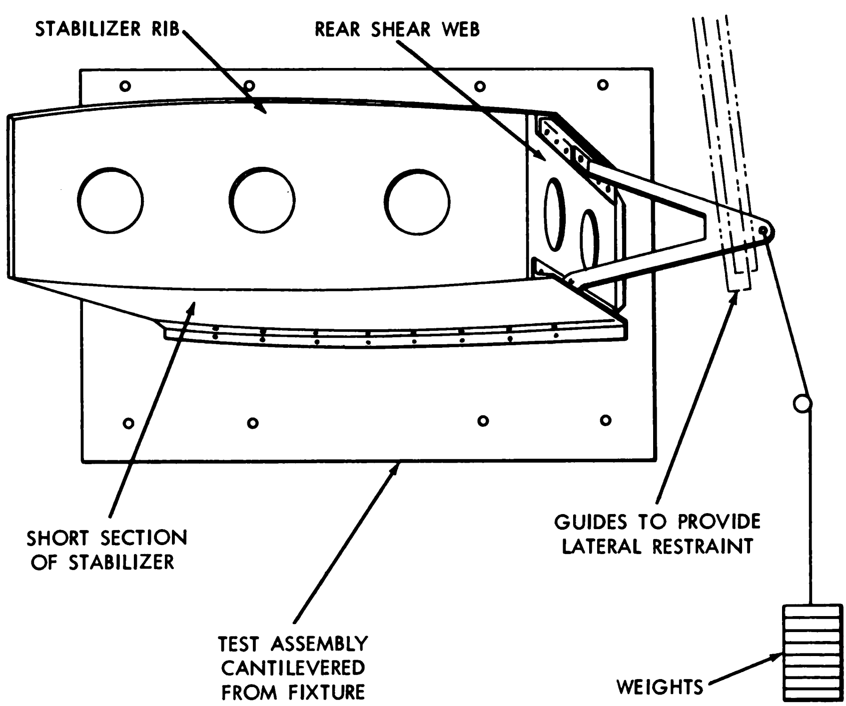


Figure 2–XIII. ***Typical test of a control surface hinge bracket and its attachment to wing.***

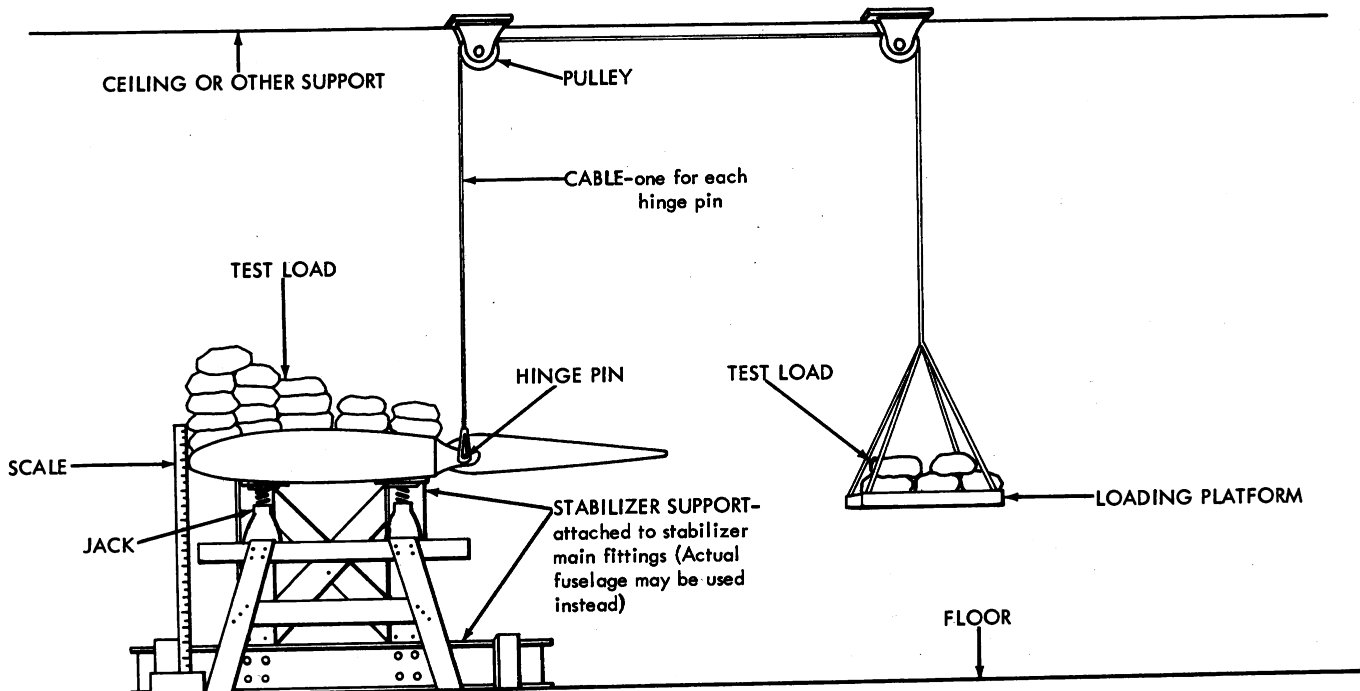


Figure 2–XIV. ***Example of setup for test of horizonal tail surface for "balancing" condition.***

(b) The control system for the main surfaces may be secured as in (a) and the load applied to the control surface horns or other outer extremities of the system by the use of cables leading over pulleys to a loading platform.

2. *Application.*—The test loads should be applied, if practicable, under conditions which produce the most critical loads in the system. For instance, push-pull tubes should be loaded in compression rather than tension. Also, brackets and fittings should be loaded under the most critical conditions if there exists a variation in strength or loading throughout the range of movement of the control system.

3. *Operation test.*—The purpose of the operating test is to deter mine that the controls and their attachment fittings are free from binding, jamming or excessive friction or deflection at the specified operating test loads.

4. *Ultimate load test.*—The purpose of a control system ultimate test is to ascertain that all parts of the system have sufficient strength and rigidity for service. Ultimate load tests should be conducted when the combination of stress analyses and limit load tests are not considered satisfactory criteria of strength. During the ultimate load test the loads should be carried to the limit load values and then removed to make sure that no undue permanent deflections up to the limit load values are within the elastic limit of the materials used in detail parts of the system.

C. Rigidity.—Extra-flexible cable as used in control systems sometimes will stretch a relatively large amount when subjected to its limit or ultimate loads even though of adequate size and prestretched. Although not particularly desirable this does not impair the airworthiness of a system unless the deflection is excessive.

D. Test report.—In all cases the manufacturer making a test is required to submit a complete report covering details of tests. The report should include photographs or drawings of the test setup and the test specimen; photographs of failed parts or sections; records of deflections and readings taken; date of test; identification number of report; serial and model number of glider and signature of responsible witnesses and/or test personnel. In addition, the following points should be covered when applicable:

1. Substantiation by references or computations of the selection of critical test conditions and loadings.
2. Loading schedule used in the test.
3. Description of test setup, with reference to drawing numbers.

Fuselage tests.—

A. Test loads.—

1. The test loads for a fuselage usually comprise 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 in the fuselage structure which during the tests are resisted by the wing reactions and the weights forward of the center of gravity. The loads from the vertical tail cause a bending moment which during the tests is resisted by the wing reactions. For the landing conditions the fuselage will be loaded by inertia loads and the landing gear (wheel or skid) reactions.
2. Two separate tests should be conducted, 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 or conditions for all parts of the fuselage. If the upper part of the fuselage appears to be weak, it should be tested for upward acting tail loads. Usually, however, the downward acting loads are critical. For the torsion test there is only one condition to consider, that is, the fin and rudder load.

Test methods.—

1. *General.*—The general structural test procedure outlined elsewhere can also apply to fuselage tests.
2. *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. Tail surface loads are applied through the stabilizer attachment fittings and loads representing weights in the fuselage, if used, are laid inside or suspended at their proper locations along the fuselage.
3. *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. See fig. 2–XII.
4. *Towing and launching loads.*—Chap. 1 states that it is unnecessary to investigate launching and towing loads aft of the front spar. However, in testing for these conditions, loads must be applied at points aft of the rear spar to resist

the test load on the towing hook. Care should be taken in testing for these conditions to guard against overloading such portions of the fuselage. For instance, if the side load is resisted only by the front strut fitting and the tail post, 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 ultimate loads for which the fuselage is designed.

In actual flight, the loads on the tow line, especially side loads, are resisted by inertia loads as well as by air loads. Accordingly, this should be given careful consideration in determining the magnitude and location of the fuselage test reactions. 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, the loads being applied through their points of attachment to the fuselage.

Reference is made in Chap. 1 to the resultant effects on the strength of the wing root fittings and adjacent structure from 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 during actual landings.

5. *Deflections.*—Deflection readings should be taken at each increment of load, at various strategic locations, including the wing attachment fittings, the rear end of the fuselage and, for the torsion test, the tip of the fin.

C. Test report.—In all cases the manufacturer making a test is required to submit a complete report covering details of tests. The report should include photographs or drawings of the test setup and the test specimen; photographs of failed parts or sections; records of deflections and readings taken; date of test; identification number of report; serial and model number of glider and signature of responsible witnesses and/or test personnel. In addition, the following points should be covered when applicable:

(a) Substantiation by references or computations of the selection of critical test conditions and loadings.

(b) Loading schedule used in the test.

(c) Description of test setup, with reference to drawing numbers.

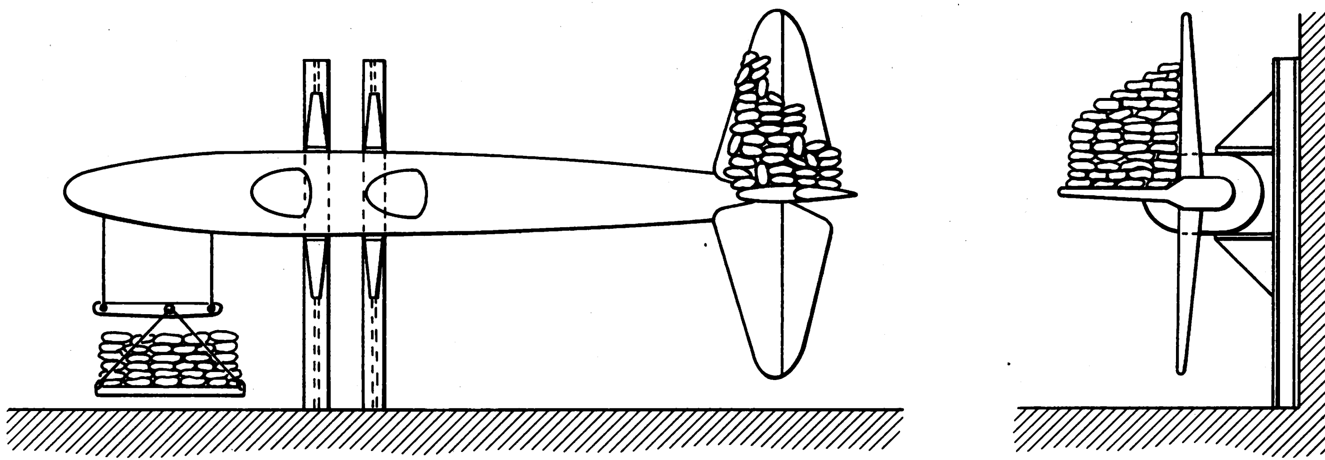


Figure 2–XII. ***Setup for test of aft portion of a fuselage for sidewise acting fin and rudder loads.***

In general, the demonstration of strength by means of flight load tests is not recommended. Any proposals for such tests should be brought to the attention of the regional office concerned for special study and recommendations.

Flutter and vibration prevention tests.—Wings, tail, control surfaces, tabs, wing spoilers, and dive brakes should be demonstrated to be free from flutter at all airspeeds and altitudes within the design speed range up to V_g. Freedom from flutter may be demonstrated either by a formal flutter analysis, by compliance with acceptable simplified flutter prevention criteria or by an acceptable flight flutter test.

Ground vibration tests.—The natural frequencies of all main structural components should be determined by ground vibration tests. These tests should determine at least the following resonant modes of vibration: Wing first symmetrical and unsymmetrical bending and wing torsion, symmetrical rotation of ailerons about their hinge lines, wing spoiler rotation about its hinge line, dive brake rotation about its hinge line, fuselage torsion, fuselage side bending, fuselage vertical bending, fin bending, stabilizer bending, rotation of tabs about their hinge lines.

Simplified flutter criteria.—Studies have indicated that for a conventional airfoil in which the center of gravity of the airfoil section is not too far back, wing flutter could be prevented by designing for a certain degree of wing torsional rigidity and by control surface dynamic balance, whereas empennage flutter could be prevented by providing a degree of control surface dynamic balance. Satisfactory rational analytic methods have been available for a number of years which would permit an engineer to carry through computations to determine the flutter stability of a specific design. In view of the fact that flutter is an aeroelastic phenomenon which is caused by a combination of aerodynamic, inertia and elastic effects, any criteria which does not consider all three effects is bound to have severe limitations. That this is so is evidenced by the fact that in almost all cases where rational analyses have been carried through for specific designs it has been found that the balance requirements specified by the simple criteria have been too severe. Although a rational flutter analysis is to be preferred to the use of the simplified criteria contained herein (since in most cases a better design may be achieved by reducing or eliminating the need for nonstructural balance weights), the application of these criteria to conventional aircraft is believed to be adequate to insure freedom from flutter.

WING FLUTTER CRITERIA

The following wing flutter criteria should be applicable to all conventional gliders which do not have large mass concentrations on the wings.

- *Wing torsional stiffness.*—The wing torsional flexibility factor F defined below should be equal to or less than $\frac{200}{Vg^2}$.

  Where: $F = \int \Theta C_i^2 ds$

  Θ_i = Wing twist at station i, per unit torsional moment applied at a wing station outboard of the end of the aileron. (radians/ft.-lb.)

  C_i = Wing chord length at station i, (ft.)

  ds = Increment of span (ft.)

  V_g = Design speed (IAS) of the glider

Integration to extend over the aileron span only. The value of the above integral can be obtained either by dividing the wing into a finite number of spanwise increments ΔS over the aileron span and summing the values of $\Theta_i C_i^2 \Delta S$ or by plotting the variation of $\Theta_i C_i^2$ over the aileron span and determining the area under the resulting curve.

In order to determine the wing flexibility factor F, a pure torsional couple should be applied near the wing tip (outboard of the end of the aileron span) and the resulting angular deflection at selected intervals along the span measured. The test can best be performed by applying simultaneously equal and opposite torques on each side of the glider and measuring the torsional deflection with respect to the glider center line. The twist in radians per unit torsional moment in ft.-lbs. should then be determined. If the aileron portion of the wing is divided into four spanwise elements and the deflection determined at the midpoint of each element the flexibility factor F can be determined by completing a table similar to table 2–I. Fig. 2–XV illustrates a typical setup for the determination of the parameters C and ΔS.

- *Aileron balance criterion.*—The dynamic balance coefficient K/I should not be greater than the value obtained from fig. 2–XVI wherein K/I is referred to the wing fundamental bending node line and the aileron hinge line. If no knowledge exists of the location of the bending node line the axis parallel to the fuselage center line at the juncture of the wing and fuselage center line at the juncture of the wing and fuselage can be used.

  Wherein: K = product of inertia

  I = mass moment of inertia of aileron about its hinge line

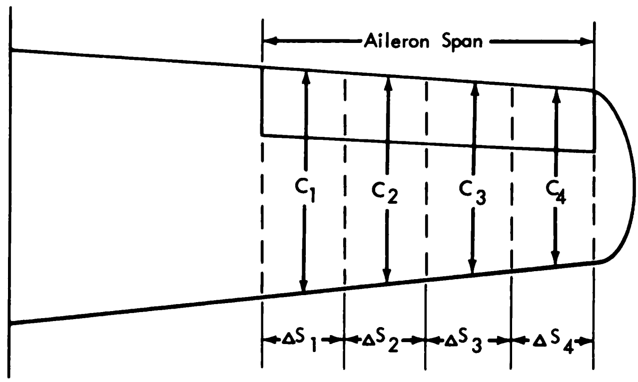


TABLE I

(1)	(2)	(3)	(4)	(5)	(6)
STATION	ΔS	C	C^2	θ	$\theta C^2 \Delta S$
	ft	ft	ft^2	$\frac{rad}{ft\ lb}$	
1 2 3 4					

F= Σcolumn (6)

Figure 2–XV. ***Chordwise extension piece to magnify wing torsional deflectional readings.***

TABLE 1–II

No.	Item	FLIGHT CONDITION I	I_1	II	III	III_1	IV	V
(1)	W = design weight, pounds							
(2)	V = velocity m,.p.h							
(3)	q = .00256 V^2 = .00256 $(2)^2$							
(4)	s = (1)/S							
(5)	q/s = (3)/(4)							
(6)	n_1 = limit wing load factor							
(7)	C_N							
(8)	C_c							
(9)	n_{x_1} = (8) x (5)							
(10)	C_M							
(11)	m_1 = (10) x (5)							
(12)	n_3 = tail load factor							
(13)	n_2 = − (6) − (12)							
(14)	n_{x_2} = − (9)							
(15)	T_2 = (1) x (12)							

- *Free play of ailerons.*—The total free play at the aileron edge of each aileron, when the other aileron is clamped to the wing should not exceed 2.5 percent of the aileron chord aft of the hinge line at the station where the free play is measured.
- *Frequency of ailerons, dive brakes, or spoilers.*—The frequency of the ailerons rotating symmetrically about their hinge lines (both ailerons moving together) should be at least 1.5 times the frequency of the fundamental symmetrical wing bending frequency. The frequency of rotation of dive brakes and/or wing spoilers about their hinge line, for any position from closed to open, should be at least 2 times the wing torsion frequency.

EMPENNAGE FLUTTER CRITERIA

The following empennage flutter criteria are applicable only if the empennage configuration is conventional and consists of a single vertical fin and rudder and a fixed horizontal stabilizer and elevators.

1. *Elevator balance.*—Each elevator should be dynamically balanced to preclude the parallel axis flutter (fuselage vertical bending-symmetric elevator rotation)as well as perpendicular axis flutter (fuselage torsion—antisymmetric elevator rotation). If, however, the antisymmetric elevator frequency is greater than 1.5 times the fuselage torsional frequency the perpendicular axis criterion need not apply.

a. *Parallel axis criterion.*—The balance parameter γ as obtained from fig. 2–XVII should not be exceeded. In fig. 2–XVII the balance parameter γ and the flutter speed parameter V_f are defined as:

$$\gamma = \frac{bS_\beta}{I}$$

$$V_f = \frac{V_g}{bf_h}$$

Where: S_β = Elevator static balance about hinge line (ft.-lbs.)
I = Elevator mass moment of inertia about the hinge line (lb.-ft.^{2})
b = Semichord of the horizontal tail measured at the midspan station (ft.)
V_g = (m.p.h.)
f_h = Fuselage vertical bending frequency (c.p.m.)

b. *Perpendicular axis criterion.*—Need not apply if the antisymmetric elevator frequency is greater than 1.5 times the fuselage torsion frequency. For each elevator the balance parameter λ as obtained from fig. 2–XVIII should not be exceeded. In fig. 2–XVIII the balance parameter λ and the flutter speed parameter V_f are defined as:

$$\lambda = \frac{bK}{SI}$$

$$V_f = \frac{V_g}{bf\alpha}$$

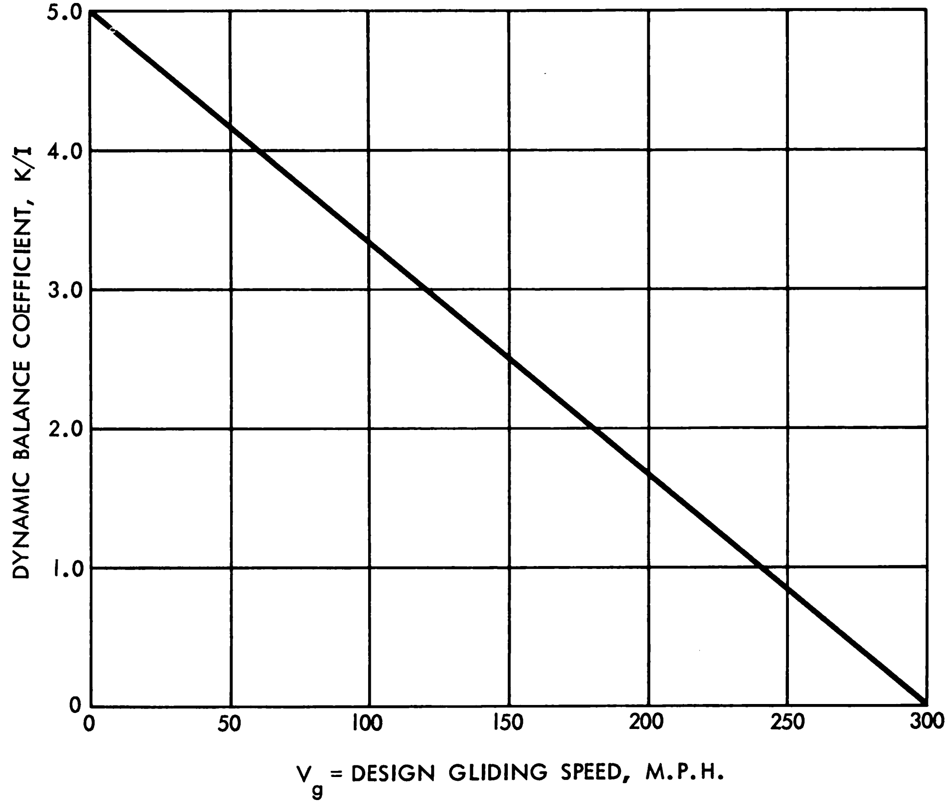


Figure 2—XVI. ***Aileron balance criterion.***

Where: S = Semispan of horizontal tail (ft.)
b = Semichord of horizontal tail at midspan station (ft.)
K = Elevator product of inertia referred to stabilizer center line and elevator hinge line (lb.-ft-^{2})
I = Elevator mass moment of inertia about the elevator hinge (lb.-ft.^{2})
$f\alpha$ = Fuselage torsional frequency (c.p.m.)

2. *Rudder balance.*—The value of γ as obtained from fig. 2–XVII and the value λ as obtained from fig. 2–XVIII should not be exceeded; where in figs. 2–XVII and 2–XVIII, $\gamma = \frac{bS_\beta}{I}$, $\lambda = \frac{bK}{SI}$ and:

S = Distance from fuselage torsion axis to tip of fin (ft.)
b = Semichord of vertical tail measured at the 70-percent span position (ft.)
K = Product of inertia of rudder referred to the fuselage torsion axis and the rudder hinge line (lb.-ft.^{2})
$f\alpha$ = Fuselage torsional frequency (c.p.m.)
f_h = Fuselage side bending frequency (c.p.m.)
S_β = Rudder static balance about hinge line (lb.-ft.)
I = Mass moment of inertia of the rudder about hinge line (lb.-ft.^{2}).

TAB FLUTTER CRITERIA

All reversible tabs should be 100 percent statically mass balanced about the tab hinge line. Tabs are considered to be irreversible and need not be mass balanced if they meet the following criteria:

1. For any position of the control surface and tab, no appreciable deflection of the tab can be produced by means of a moment applied directly to the tab, when the control surface is held in a fixed position and the pilots tab controls are restrained.
2. The total free play at the tab trailing edge should be less than 2.5 percent of the tab chord aft of the hinge line, at the station where the play is measured.
3. The tab natural frequency should be equal to or exceed the value given by the lower of the following two criteria

   (a) $f_t = \frac{48V_g}{C_c}\frac{S_t}{S_c}$ c.p.m.

   or

   (b) f = 2,000 c.p.m. for gliders having a design speed of less than 200 m.p.h.

Thus for a glider with a design speed less than 200 m.p.h. if (a) above gave a value in excess of 2,000 c.p.m. it would only be necessary to show a frequency of 2,000 c.p.m. for the frequency criterion.

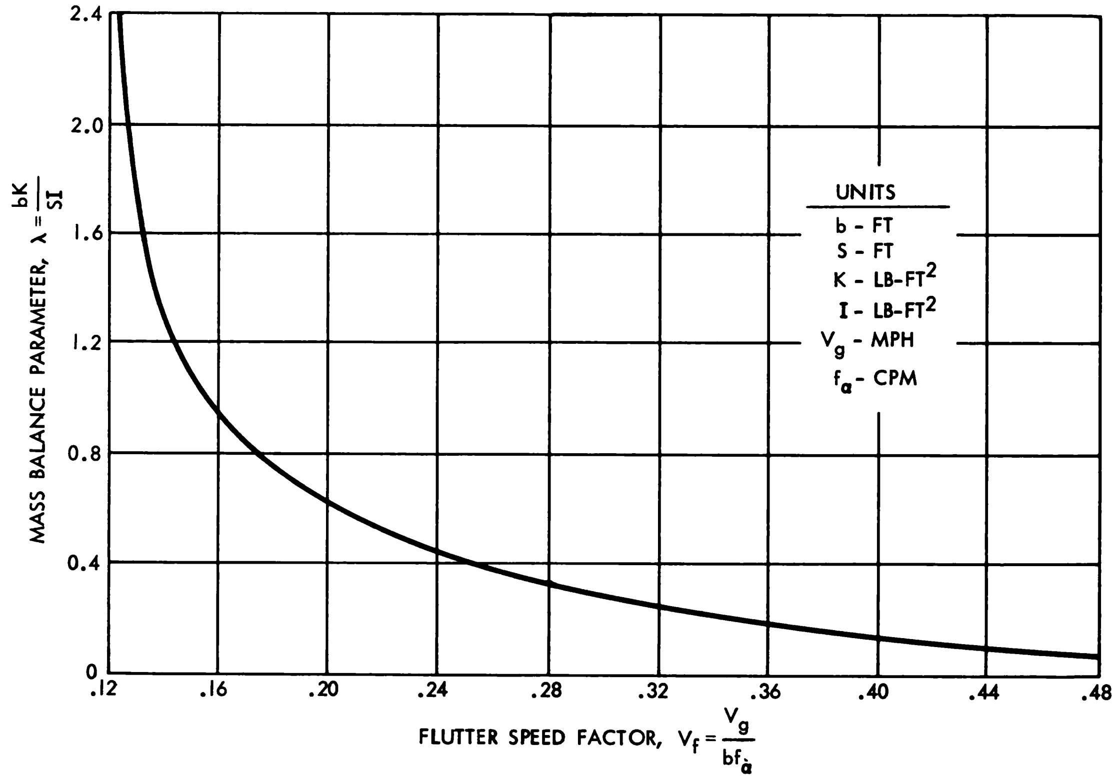


Figure 2—XVIII. ***Elevator and rudder balance criteria perpendicular axes (Fuselage torsion—control surface).***

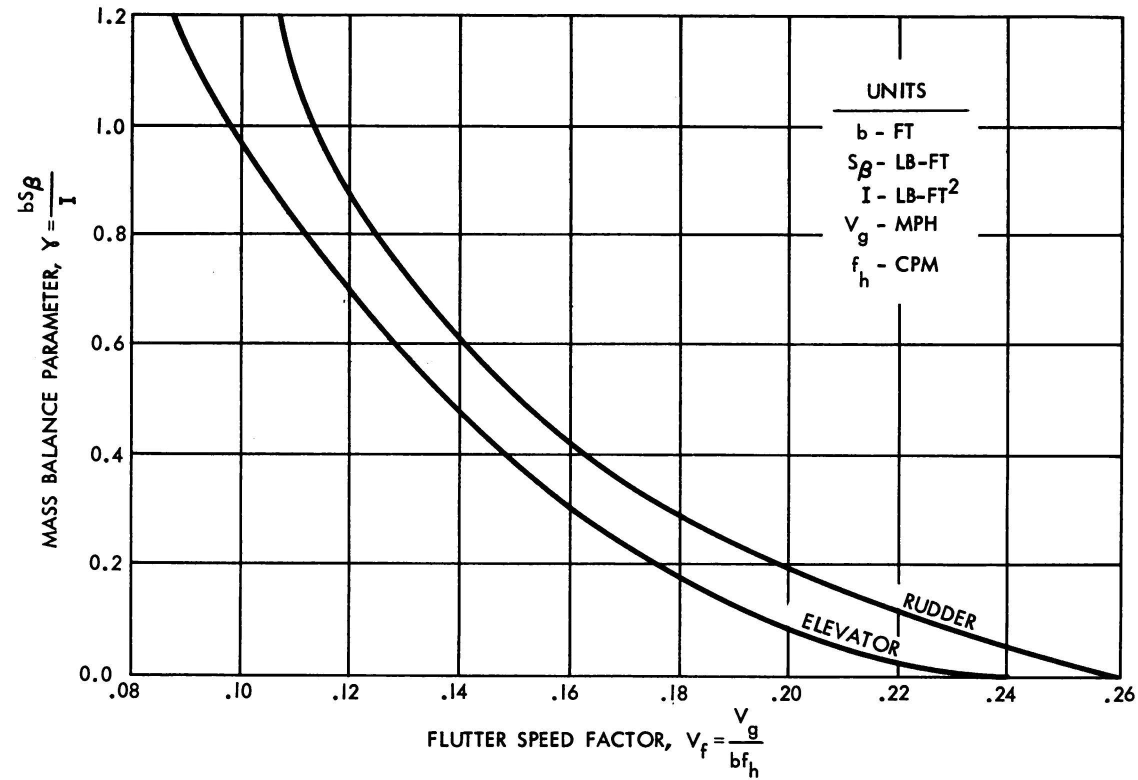


Figure 2–XVII. Elevator and rudder balance criteria parallel axes (fuselage bending—control surface).

Where: f_t = lowest natural frequency of the tab as installed in the glider (c.p.m.)—either tab rotation about the hinge line or tab torsion whichever is lower.

C_c = Chord of movable control surface aft of the hinge line, at the tab midspan position (ft.).

S_t = Span of tab (ft.)

S_c = Span of movable control surface to which tab is attached (both sides of elevator, each aileron and rudder) (ft.)

Particular care should be taken in the detail design to minimize the possibility of fatigue failures which might allow the tab to become free and flutter violently.

Balance weight attachment criteria.—Balance weights should be distributed along the span of the control surface so that the static unbalance of each spanwise element is approximately uniform. However, where a single external concentrated balance weight is attached to a control surface of high torsional rigidity the natural frequency of the balance weight attachment should be at least 50 percent above the highest frequency of the fixed surface with which the control surface may couple in a flutter mode. For example the aileron balance weight frequency should be at least 50 percent above the wing fundamental torsional frequency. The balance weight supporting structure should be designed for a limit load of 24g normal to the plane of the surface and 12g in the other mutually perpendicular directions.

It should be noted that the dynamic balance coefficient K/I can be reduced by (1) reducing K, (2) increasing I or (3) reducing K and increasing I. Since an increase in I results in a reduced control surface natural frequency with possible adverse flutter effects, the primary purpose of ballast weights used to reduce K/I, should be to decrease the product of inertia K and not to increase the mass moment of inertia I.

Flight flutter testing.—Various procedures are available for demonstrating freedom from flutter, namely: Performance of a flutter analysis, application of the simplified criteria as discussed above, when applicable, and performance of a flight flutter test by the applicant. It is not recommended that flight flutter tests be used as a general procedure for substantiating freedom from flutter. The performance of a flutter analysis or the application of the simplified flutter criteria are considered preferred procedures due to the hazards involved in flight flutter tests. It is recommended that flight flutter tests be performed only when allied investigations or engineering evaluations give some assurance that the tests may be performed safely. A flight flutter test program may involve flutter substantiation of two or more control surfaces whose static unbalance values are in excess of the allowable values in the simplified flutter criteria. As a precautionary procedure, it is recommended that tests be performed for one surface at a time with the remaining unsubstantiated control surfaces balanced to at least the degree indicated necessary by the simplified flutter criteria.

(a) *Acceptability.*—Flight flutter tests will be acceptable as substantiation of freedom from flutter when it can be demonstrated by such tests that proper and adequate attempts to induce flutter have been made within the speed range up to V_g, and the vibratory response of the structure during the tests indicates freedom from flutter.

(b) *Records.*—Flight recording instrumentation of either the electrical or photographic type should be installed to provide a permanent record of the control surface and/or fixed surface response to the applied flutter exciting forces, as well as a record of the associated test airspeed.

(c) *Test procedure.*—The following is an outline of an acceptable procedure for demonstrating freedom from flutter by flight tests in which rapid control surface deflections are applied to induce flutter:

(1) The tests should cover the flight speed range with excitation applied at small incremental increases of airspeed up to V_g. The incremental speed increases between 0.8 V_g and V_g should not be more than 5 m.p.h. At lower flight speeds, larger increments of airspeed may be used.

(2) The controls should be deflected to attempt to excite flutter as follows:

- Aileron control to induce wing and aileron flutter.
- Rudder control to induce rudder and vertical tail flutter.
- Elevator control to induce elevator and horizontal tail flutter, and symmetric wing flutter.
- Control surface to which the tab is attached to induce tab flutter.

(3) Attempts to induce flutter should be made by abrupt rotational deflections of the respective control surfaces. These deflections should be obtained by striking the corresponding control with the free hand, or foot, and the disturbed control should be allowed to stabilize without restraint by the pilot. The force applied should be sufficient to produce an impulsive deflection of the control surface of at least 3 degrees.

(4) At each test speed, at least 3 attempts to induce flutter should be made for each of the surfaces being investigated.

(5) A permanent record* at each test speed should be obtained as follows:

- In flutter tests of the rudder, elevator and tab surfaces; a time history of the control surface rotational deflection, and the associated airspeed.
- In flutter tests of the wing and aileron; a time history of the aileron rotational deflection, a time history of the wing vibratory response, and the associated test airspeed.

(6) The tests should consider significant variations in mass and rigidity values which might be expected in service. The aileron and tab control systems should be freed to the extent necessary to be representative of what might be expected in service.

(7) The tests should be conducted at altitudes of approximately 50 to 75 percent of the service ceiling.

(8) The vibration survey should be conducted prior to performing the flight flutter tests. The structural frequency and vibration mode data should be evaluated to determine what structural modes are most likely to be flutter critical.

(9) The resulting oscillations of the wing and/or control surfaces in response to the excitation applied shall be damped with no tendencies to persist at any test speed.

---

* These data can be obtained by installing a control surface position indicating device at the control surface, a vibration pickup in the vicinity of one wing tip to detect the wing response, and a suitable pressure transducer connected to the aircraft's pitot-static system to measure the airspeed with the electrical signals from these instruments connected to a recording oscillograph.

Alternatively, photographic methods using cameras may also be used for recording the flight test data. However, the photographic records obtained should be of a nature that will permit satisfactory evaluation of the degree of control surface deflection applied, the flutter stability and a correlation of the airspeed record with the associated flutter test point.

Chapter 3—DESIGN, CONSTRUCTION AND FABRICATION

The primary structure should not incorporate design details which experience has shown to be unreliable or otherwise unsatisfactory. The suitability of all design details should be firmly established. Products such as bolts, pins, screws, tie-rods, wires, terminals, et cetera, used in the primary structure should be of aircraft standards as established by the SAE or as established by government aircraft standards and specifications or TSO's. Strength values contained in MIL-HDBK-5, ANC-18, and MIL-HDBK-17, and ANC-23 Part II, should be used unless shown to be inapplicable in a particular case. The primary structure should 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 should be such as to produce a uniformly sound structure which should also be reliable with respect to maintenance of the original strength under reasonable service conditions. The workmanship of the primary structure should be of sufficiently high grade to insure proper continued functioning of all parts. All members of the structure should be suitably protected against deterioration or loss of strength in service due to weathering, corrosion, abrasions and other causes. Adequate provision for ventilation and drainage of all parts of the structure should be made. (See also FAA Technical Manual No. 103 "Aircraft Design Through Service Experience.")

PROCESSES

Detailed information on assembly processes will be found in CAM 18, "Maintenance, Repair and Alteration of Airframes, Powerplants, Propellers and Appliances," available from the Government Printing Office, Washington, D.C. Bulletin ANC-19, "Wood Aircraft Inspection and Fabrication" also includes detailed information on the properties and fabrication techniques of various woods and wood materials and associated data on aircraft glues.

WING DESIGN

The more elementary types of gliders usually call for inexpensive wing construction, with simple parts which are easy to construct and assemble. Typical construction of this type is the two-spar wing having a drag truss composed of fore and aft drag struts and single drag and antidrag wires. Where double drag wires, that is, top and bottom drag wires, are employed, the torsional stiffness of the wing is greater than if single drag wires are employed.

If the general planform of the wing is rectangular, and if the spars are parallel, construction is relatively simple. For instance, in the layout of fittings, most of the angles would be right angles, and the problems of load determination would be two dimensional. In wings with rectangular planform, most of the ribs would be of identical construction. In contrast, if the general planform of the wing is tapered, then the construction becomes more complicated, as evidenced by the fact that the spars would necessarily taper in depth and each rib would be different due to the effect of taper. However, there are many advantages to the tapered wings. Usually, they are more efficient structurally and aerodynamically. The wing structures of most gliders of advanced design are currently comprised of a single spar with a D-nose and a special drag strut intersecting the main spar at about the 1/4-span point. The single spar with D-nose type of construction requires careful workmanship, especially in laminating the spar and in attaching the nose cover over the supporting nose-ribs.

The single spar D-nose type of wing construction lends itself to metal as well as wood construction. Metal has the disadvantages that special skill, tools and machines are 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. Also, metal working tools and machinery are more expensive than wood working tools and machinery. The minimum gauges of material usable from the standpoint of handling, workability, and corrosion resistance and, in some cases, the minimum gauges available commercially, often exceed that needed to carry the design loads. Consequently, the structure if of metal may be somewhat heavy. For a particular wing of relatively small thickness, but having a high aspect ratio and high design loading, it is probable that metal construction would be as light and efficient as wood.

Particular care must be used in the design of the wing-to-fuselage attachment fittings of single-spar wings. In some cases a secondary member is 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 rigid 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; otherwise critical loads may be imposed on the fittings due to the dead weight of the wing.

Torsional stiffness.—It is essential that the wing structure have adequate torsional stiffness in order to insure freedom from flutter and other undesirable characteristics. In contrast to metal or plywood covered wings, fabric covered wings usually lack torsional rigidity.

Wing design details.—Wing spars.—Provision 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.

Solid wood spars.—Solid wood spars should be made of Grade A spruce. They may be tapered in depth or thickness, but should not be thinner than ¼-inch at any point.

Built-up wood spars.—Of the built-up types of spars, the box with smooth plywood faces with single upper and lower flanges is the most convenient for attaching ribs. It has half as many flanges to make as the "I" type. In either type, blocking must be provided at all points where fittings are attached. Blocking should 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 vertical sprovide this stiffening effect on the webs. CAM 18 shows details of spar construction.

Top and bottom spar flanges are usually proportioned according to the relative magnitudes of the critical bending loads in each direction such that the corresponding margins of safety will be approximately equal.

Spar webs are usually 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 stiffener.

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

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

Wood leading edge.—Ordinarily 3-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 $\frac{1}{16}$-inch for spruce or mahogany leading edge material is recommended.

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 leading edge covering back past the spar between ribs, to keep the fabric from pulling abruptly over the sharp edge of the spar.

Wood ribs.—Rib spacing under a stressed torsion-nose depends on the thickness of the plywood and the curvature. Usual practice is about 6-inch spacing with 12-inch spacing 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 canning." Roughly, the maximum spacing for nose ribs should not be greater than the nose-rib chord, preferably less.

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.

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 materials 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 glue work is only clamped.

End ribs and corner ribs should be braced or specially designed to provide stiffness against fabric tension loads. A double set of ribs

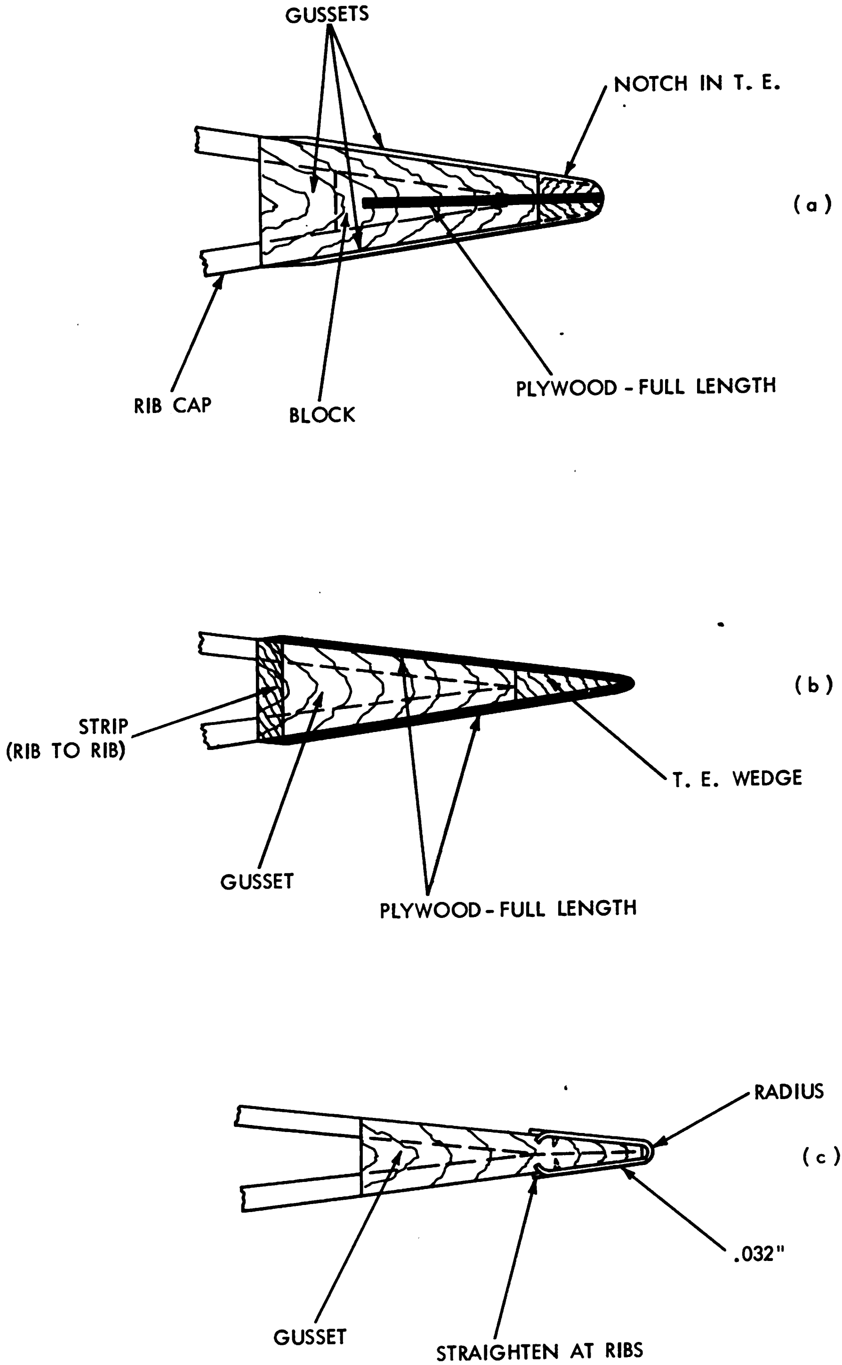
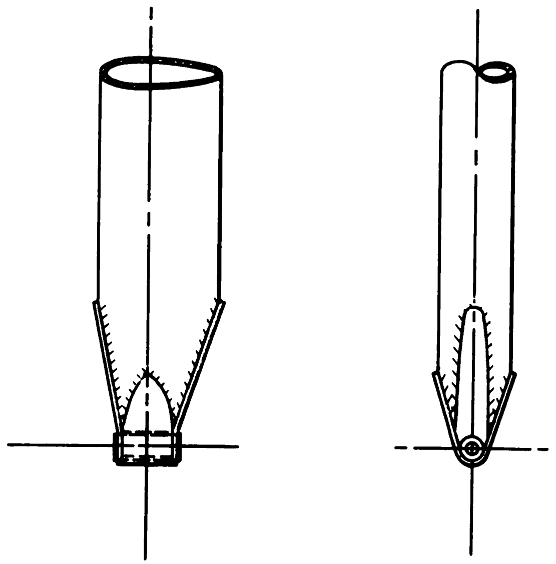
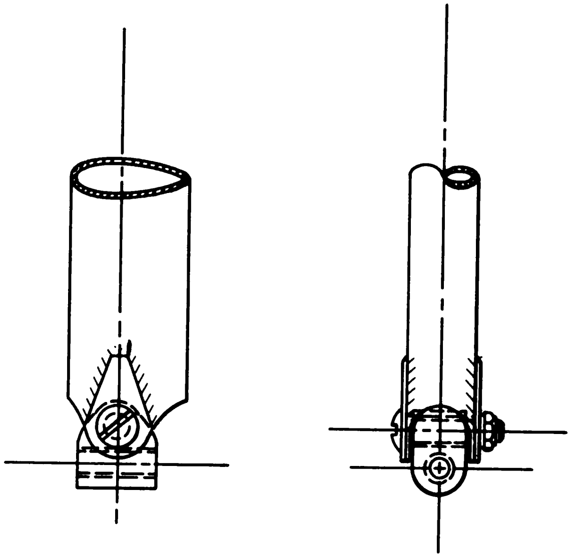


Figure 3–1. ***Trailing edge constructions.***



(a)



(b)

Figure 3–11. ***Typical strut ends.***

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.

Trailing edges.—Typical laminated trailing edges are shown in fig. 3–I. 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 strong enough to withstand considerable rough handling when setting up the glider.

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.

External brace struts.—

- Struts may be wood, or metal tubing, either round with fairing or streamlined in shape. Material is either steel or aluminum alloy.
- The wooden struts are usually tapered and are made up solid and of square or rectangular section with plywood fairing. The end fittings must be carefully designed to transmit the tension loads into the wood portion.
- Steel struts usually have the end fittings welded on so that the load is carried through the weld in shear. However, in the case of aluminum alloy struts the fittings are riveted or bolted onto the end of the strut. A typical end fitting for a streamline steel strut is shown in fig. 3–I. For a single strut, it is desirable to provide universal end fittings similar to fig. 3–II, b.
- 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.
- In connection with single-apar braced wings employing airfoils which have a large center of pressure travel, it is desirable 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 and socket joints are difficult to design and are not recommended.

Jury struts.—When clamps are used for the attachment of jury to lift struts, the design should be such as to prevent misalignment or local crushing of the lift strut.

Wing fittings.—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 on a shear plate 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.

Bolts for attaching fittings to spars are passed directly through the spar, or through suitable bushings which have for their purpose to increase the bearing area in the wood. Care should be taken not to weaken the spar by too close 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.

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 fig. 3–VII.

Fabric covering.—

- The fabric should be well finished with dope to provide a smooth surface essential to high performance.
- A rough surface is conducive to high 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.
- All handholes through the fabric and holes where controls, et cetera, enter should be properly reinforced by inspection rings or frames.
- When covering a wing with fabric, it is common practice to pass the fabric around the leading edge.

Metal covered wings.—Metal covered wings should be free from buckling or wrinkling of the metal covering. Deflections or deformations at low load factors which may result in fatigue failures also should be avoided.

CONTROL SURFACES DESIGN

a. Trailing edges of control surfaces should be as substantial as the trailing edges of wings. They should hold their shape when subjected to fabric tension loads.

b. The covering of control surfaces should be provided with holes for drainage and "breathing."

c. 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, and that interference cannot occur.

d. The stabilizer on experimental designs should be arranged so that the incidence can be changed if necessary to obtain the proper balance.

e. On wings which have no means of adjustment for 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.

f. If the elevator is unusually low so that it drags in high grass, it may be advisable to cover it with the heavier Grade A fabric instead of the light glider fabric.

g. The leading edge of wood fins and stabilizers usually 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, then ribs are usually built with solid plywood webs.

h. Wood 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 $\frac{3}{16}$ by $\frac{1}{4}$ inches, 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.

i. It will be found very convenient for construction purposes to design the wood 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.

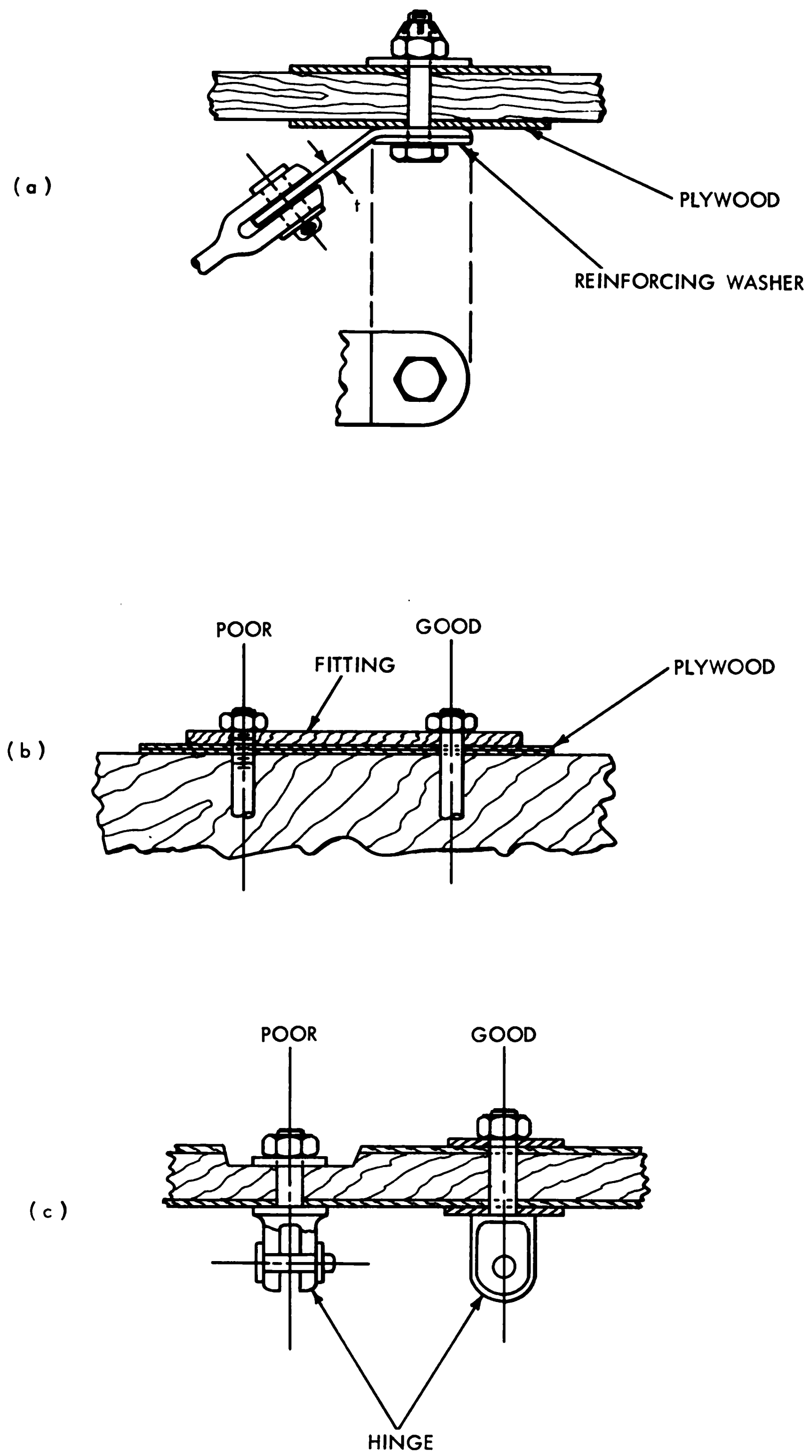


Figure 3–III. ***Typical fittings.***

j. 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 in a fore and aft direction also should be considered. It is desirable to have the control horn at a hinge so that the control load can be reacted through the hinge without bending the beam in the control surface. This also eliminates undesirable flexibility in the control system.

k. Control surface horns are generally constructed of plywood or metal, preferably the latter. They should be attached to the control surface spars near a hinge to avoid bending the spar, and in such a manner as to distribute the loads into the spar without tending to split it.

l. 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 zigzag truss from the horn.

m. Long ailerons often have two control horns in order to provide extra torsional stiffness and keep the surface from feeling "rubbery" under load.

n. 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.

Control surface stops.—Stops are advisable for all control surfaces, particularly adjustable stabilizers and elevator trailing edge tab systems. For these, the stops should be positioned so as to limit the travel to the approved range. In general, stops are advisable for 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. Stops should be installed at the control systems in the cockpit and also at the control surfaces.

Hinges.—

a. Hinges of the strap type bearing directly on torque tubes are advisable only in the case of steel torque tubes which have a multiplying factor of safety as specified in Chap. 1. In other cases sleeves of suitable material should be provided for bearing surfaces.

b. Clevis pins may be used as hinge pins provided that they are made of suitable material and are properly locked.

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

1. Provisions for lubrication should be made if self-lubricated or sealed bearings are not used.
2. 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.
3. Sufficient restraint should be provided in one or more brackets to withstand forces parallel to the hinge center line. Rudders, for instance, may be subjected to high vertical accelerations in ground operation.
4. 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.
5. Piano type hinges are acceptable with certain restrictions. In general, only the "closed" type should be used, that is, 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 exist at 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.

Installation.—

a. Movable tail surfaces should be so installed that there is no interference between the surface or their bracing when any one is held in its extreme position and any other is operated through its full angular movement.

b. 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."

c. 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.

d. 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.

e. 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.

Elevators.—

a. When separate elevators are installed they should be rigidly interconnected.

b. When dihedral is incorporated in the horizontal tail the universal connection between the elevator sections should be rugged and free from play.

Tabs.—

a. Control surface trailing-edge tabs should be statically balanced about their hinge lines, unless an irreversible nonflexible tab control system is used. The installation should be such as to prevent development of any free motion of the tab.

b. If trailing-edge tabs are installed, care should be taken in proportioning areas and relative movements so that the main surface is not aerodynamically overbalanced at any time.

c. Minimum deflections and play are of primary importance in the installation of tabs. Strength of the surface and anchorages should be sufficient to prevent damage or misalignment from handling, particularly thin sheet tabs which are set by bending to the proper position.

CONTROL SYSTEMS DESIGN

a. Rigidity.—It is essential that control systems, when subjected to limit load 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.

b. 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, that is, 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 fair leads are used.

c. 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:

1. The locking device should be so installed as to positively prevent taxiing or taking off, either intentionally or inadvertently, while the lock is engaged.
2. Means should be provided to preclude the possibility of the lock becoming engaged during flights.

Installation.—

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.

a. *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 at least 2 inches travel either way.

b. *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.

c. *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.

Stops.—

a. All control systems should be provided with stops that 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.

b. 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 also should be installed at the surfaces. Stops should be adjustable where production tolerances are such as to result in appreciable variation in range of motion.

Hinges, bearings, and joints.—

a. Hinges.—Control surface hinges of aircraft standard normally are available at the supply houses. In general, hinge pins should be ⅛ 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.

b. *Bearings.*—Bearings should be arranged so that they can be readily inspected and lubricated.

c. *Friction.*—Excessive friction should be especially avoided in the aileron control systems of large high performance gliders. This may sometimes dictate the use of special anti-friction bearings at heavily loaded pivots.

d. *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.

e. *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 fig. 3–IV.

Cables, pulleys and fairleads.—

a. *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 8.212 of MIL–HDBK–5. The 7 x 7 construction is found satisfactory where slight cable bends around pulleys of only 30° or less are encountered. Cable and splices should be made by an approved tuck method such as that of the Army and Navy, except that standard served and soldered splices are acceptable for cables not over 3/32 inch in diameter. However, cables of other sizes with served and soldered fastenings will be satisfactory provided they are not stressed above 50 percent of their rated strength. Approved swaged type terminals are also acceptable. Dimensions for approved splices are given in fig 3–V. It should be noted 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 fig. 3–VI.

b. *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 fig. 3–VII.

c. *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.

d. *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.

e. *Pulleys.*—Standard pulleys of the A–N type usually are lighter than metal pulleys. 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 fig. 3–VIII. 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.

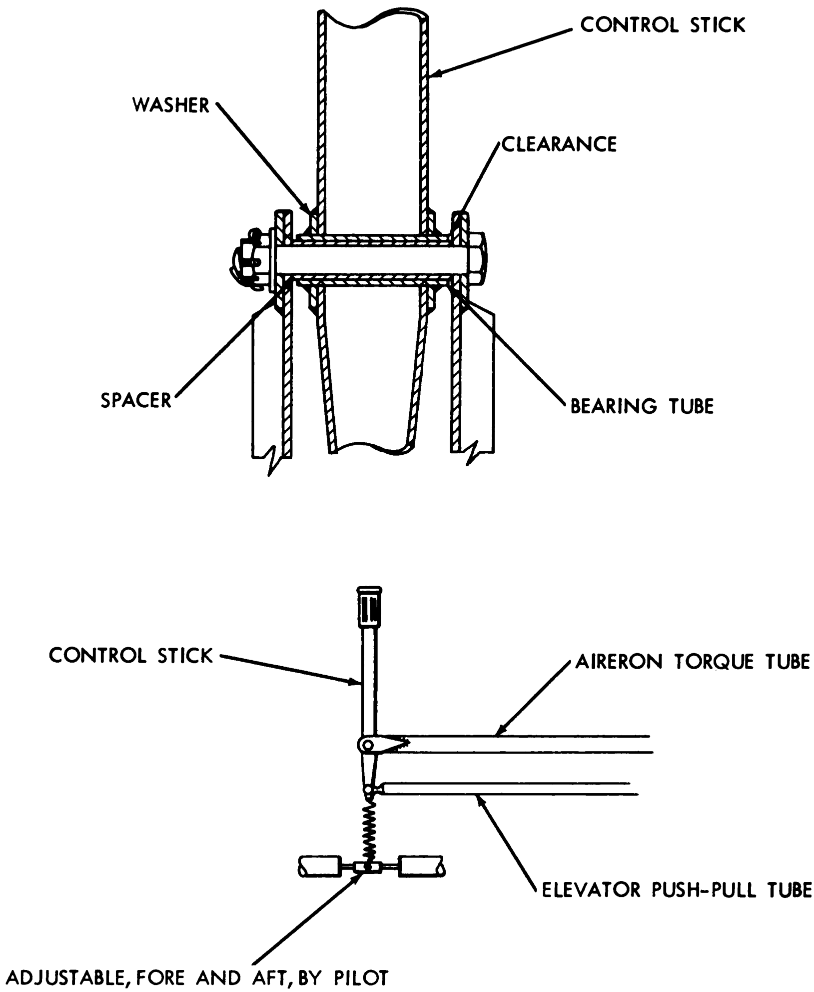
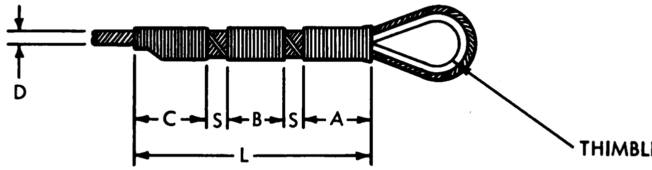


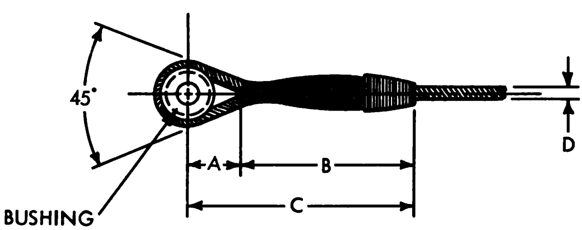
Figure 3–IV. ***Adjustable elevator bungee, control system details.***

(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

Figure 3–V. *Standard cable splice ends.*

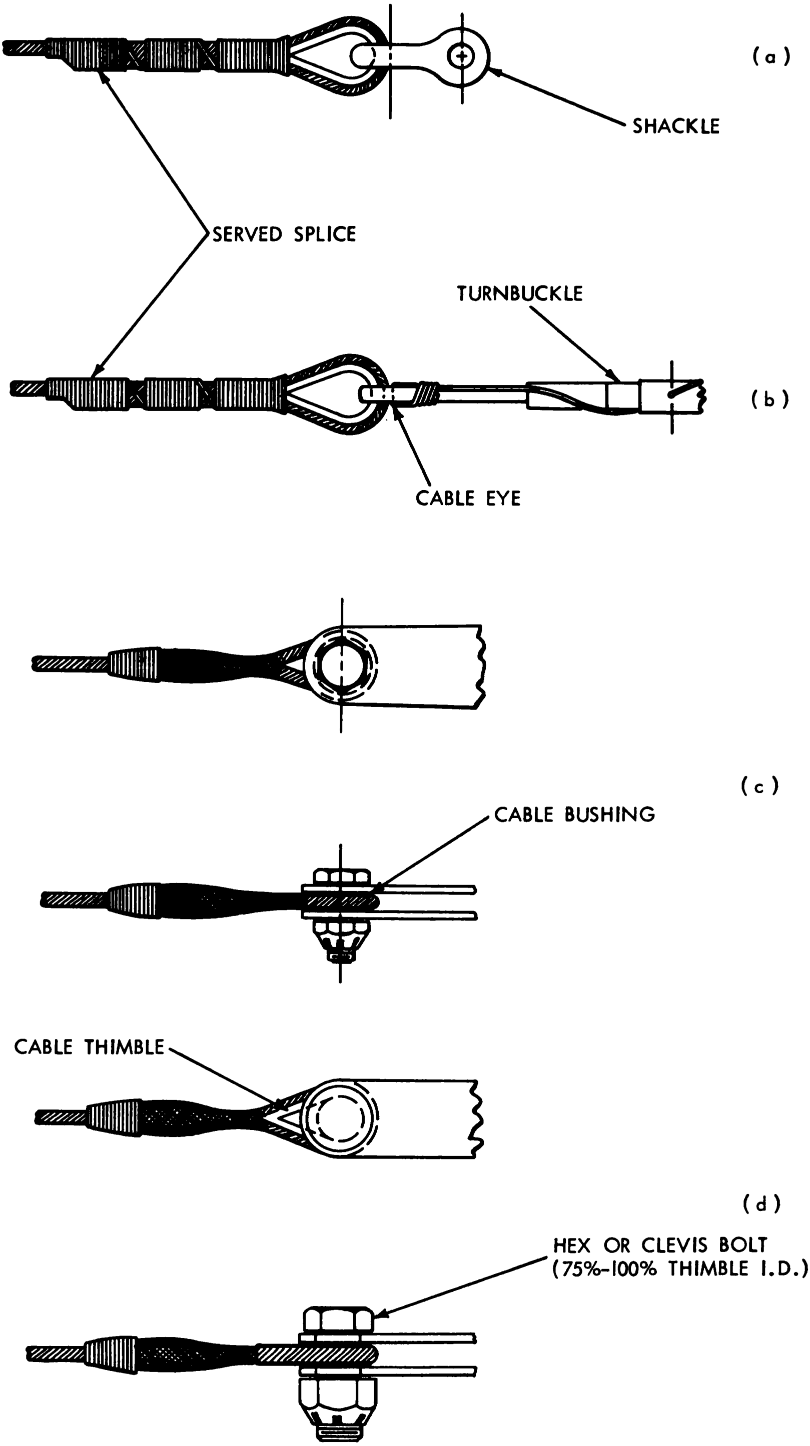


Figure 3–VI. *Cable end attachments.*

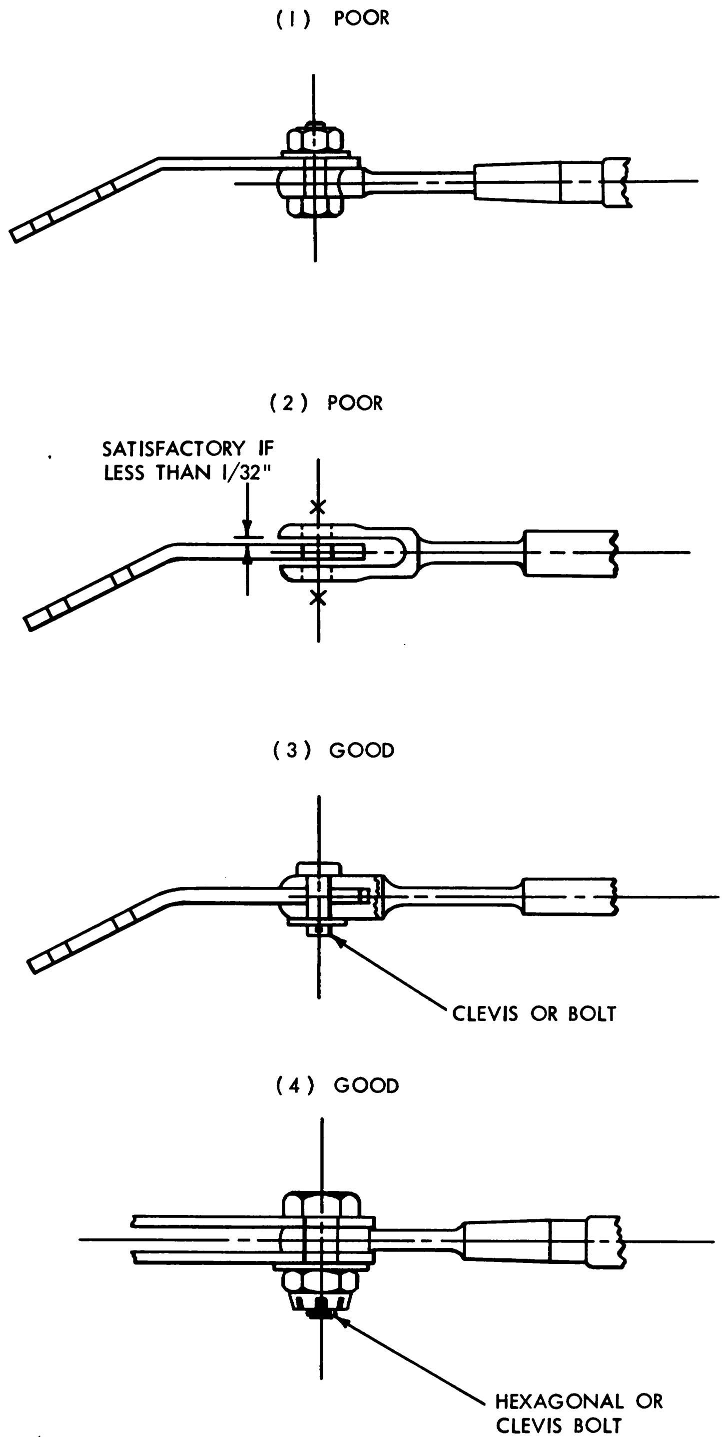


Figure 3–VII. ***Turnbuckle installations.***

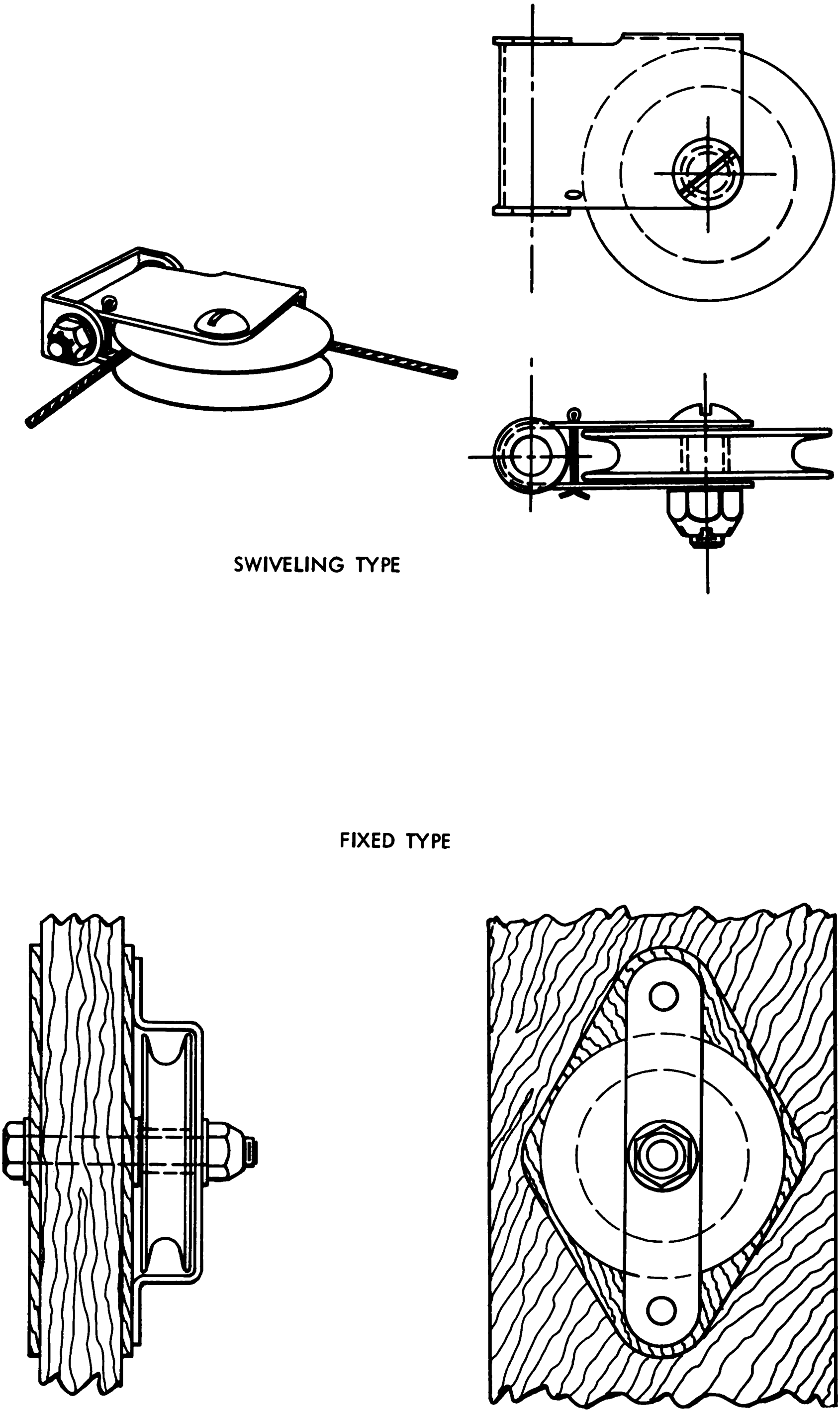


Figure 3–VIII. *Control-pulley brackets.*

Motions and clearances.—

a. *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. Elevators mounted on adjustable stabilizers, in particular, should be checked for this possibility. Pulley guards should be close fitting to prevent jamming from slack cables since wide temperature variations may cause rigging loads to vary appreciably.

b. *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.

c. *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 degrees unless additional friction, a detent, or equivalent means is used. In general, some forms of irreversible mechanism should be incorporated in the system, particularly if the stabilizer is hinged near the trailing edge.

d. *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 throwover 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.

e. *Clearance.*—At the control surfaces themselves ample clearances (5°) should be left beyond the normal deflections, to prevent interferences and possible damage when the surface is slammed over by wind on the ground.

f. *Nose wheel.*—It is essential when a nose wheel steering system is interconnected with the flight controls that care be taken to prevent excessive loads from the nose wheel over-stressing 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.

Single cable controls.—Single cable controls refer to those systems which do not have a positive return for the surface or device being controlled. In general, service experience shows that their use has been satisfactory. 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. If should be noted that it is not the intent of the specified requirement to require a duplication of cables performing the same function.

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 which can be demonstrated to be satisfactory by actual flight tests.

c. Rubber cord should not be used for this purpose.

Flap controls.—

a. 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.

b. 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.

Tab controls.—

a. *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 takeoff and the directions of motion of the control for nose-up and nose-down motions of the glider.

b. Reversibility.—Tab controls should be irreversible and nonflexible, unless the tab is statically balanced about its hinge line.

c. Wear and vibration.—In addition the 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.

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

a. *Direction of operation.*—Proper precautions should be taken against the possibility of inadvertent or abrupt tab operation and operation in the wrong direction.

LANDING GEAR DESIGN

a. 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.

b. 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 takeoff or landing, the spring leaf type is desirable.

Landing skids.—

a. *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.

b. *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 length into the fuselage by means of a flexible boot.

c. *Skid design.*—Skids should be reasonably easy to replace especially on gliders not provided with a wheel. A minimum size of ½ x 2 inches is recommended. For use on coarse-surfaced airports, a metal-faced skid should be used.

MAIN GEAR

The wheels of the main landing gear are most satisfactory for all varieties of operations if located close underneath the average CG 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.

Shock absorption.—

a. 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.

b. 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.

c. 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.

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.

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 materials 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.

Dual wheels.—Although not as common, landing wheels are sometimes used in pairs. If spaced so that the CG of the glider cannot fall outside of the wheel base, the glider will not lie on the wing tip as with a single wheel.

FUSELAGE DESIGN

a. *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.

b. *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 launching and towing loads from the towing hook.

c. *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.

d. *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 takeoff 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.

e. *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.

f. *Provisions for turn-over.*—The fuselage and cabins should be designed to protect the passengers and crew in the event of a complete turnover and adequate provision should be made to permit egress of passengers and crew in such event.

Pilot and passenger compartments.—

a. *Ventilation and visibility.*—The pilots 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.

b. *Seats.*—Seats 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 the applicable TSO for safety belt requirements.) Consideration should be given in the seat installation, to the fact that under some types of operations parachutes will be required. Provisions therefor should be made for parachutes. Shoulder harness may also be necessary in certain operations.

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

Steel tube fuselages.—

a. *General.*—Steel fuselages are built up by welding round or square 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.

b. *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. Usually, the stress analysis can be simplified by elimination of redundant members.

c. *Diagonal braces.*—Diagonal braces are necessary for stability in rectangular or diamond shaped bulkheads having unsymmetrical 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 carry the shear around the open rectangle. This is not necessary when an extra over or under truss serves the same purpose. 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.

d. *Joints.*—Wherever, possible joints should be designed for simplicity. Wherever possible, not over six members should intersect at a joint. This is 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 for tension rather than for column loads, it is 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.

e. *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.

f. *Fairing.*—Truss fuselages can be faired out for better shape which results in lower aerodynamic drag. Strips of wood, tubing, or aluminum alloy sections are used for this purpose. These members should be strong enough to resist the fabric tension and handling loads. The fairing strips are supported by clips on the structure or plywood formers built out from the bulkheads.

Plywood monocoque fuselages.—

a. *General.*—Plywood monocoque fuselages are either of simplified type with flat faces, or of full curved form. The latter type has 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.

b. *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, and in the pilot's bay. A minimum thickness of 1 mm. for birch and 3/64 inch for spruce or mahogany is recommended for fuselage covering.

c. *Bulkheads.*—The bulkheads in this type of fuselage can be made of straight struts fastened at the corners by blocking notched for the longerons and gusseted with plywood.

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

e. *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; however, the natural periods of vibration 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 vibration tests conducted with the glider on the ground.

f. *Concentrated loads.*—The loads from main fittings should be well distributed to 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 fig. 3–IX.

Crash protection.—

a. *General.*—The fuselage should be designed to give reasonable assurance that each occupant provided with the proper belts and harness will not suffer serious injury during minor crash conditions as a result of contact of any vulnerable part of his body with any penetrating or relatively solid object.

If the characteristics of the glider make a turnover reasonably probable, the fuselage, in combination with other portions of the structure, should be designed to afford protection of the occupants in a complete turnover.

Compliance Suggestion

PERSONNEL COMPARTMENT

Relatively rigid structural members or rigid mounted items of equipment which might be struck by the head, arm, knees, et cetera, should be padded. Padding preferably should be of foam polystryrene or unicellular polyvinal chloride material rather than ordinary foam or sponge rubber, which are ineffective. Heavy transversal

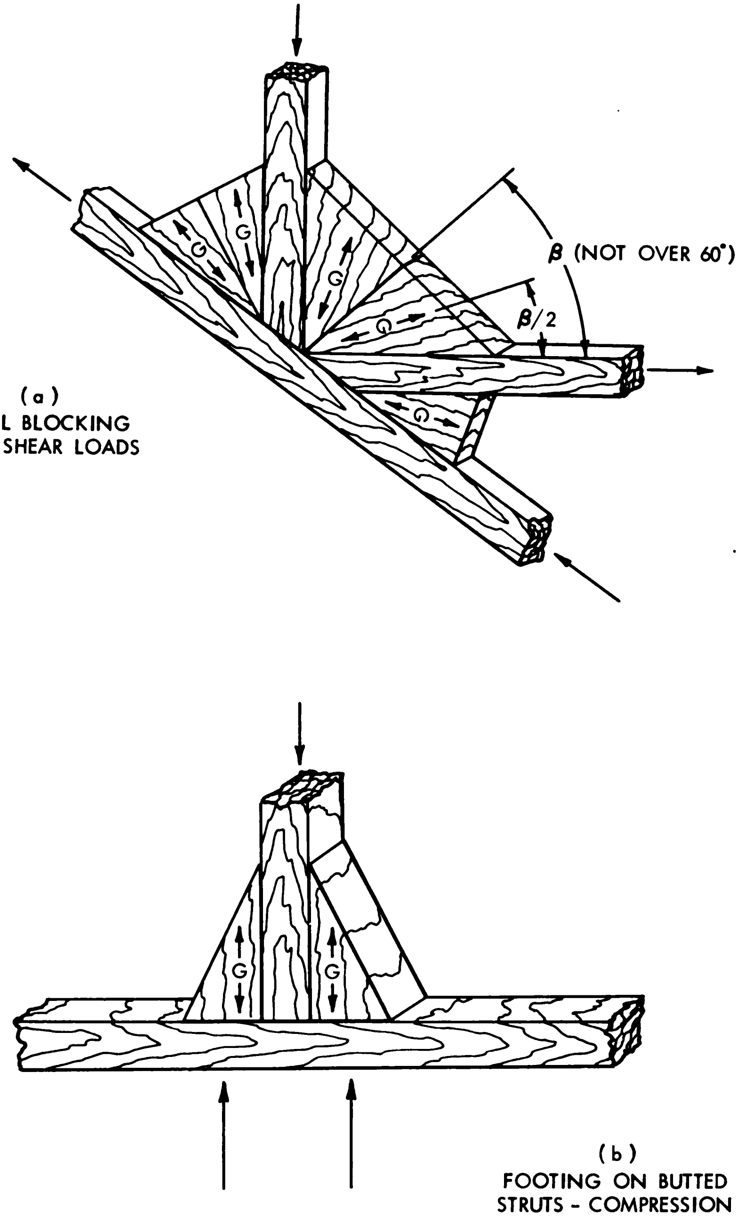


Figure 3—IX. ***Installation of corner blocks.***

braces or other structure behind a light weight instrument panel should be positioned several inches behind the ductile skirt of the instrument panel. The backs of forward positioned seats in tandem gliders, often bearing solid tubing members in the backrest, should be designed to pivot forward so that the occupant in the rear seat would not contact the solid member due to the body pivoting about the seatbelt. There should be a similar clearance for the front occupant. Instrument panels should be smooth, with top edge curved in a substantial radius. It is preferred that openings for instruments should have beveled instead of sharp edges. Also, seats should be adequate to withstand downward loads without failure of the seat or injury to the pilot. Control wheels and control sticks should be free of sharp edges, hooks or projections which might cause injury in a minor accident.

Compliance Suggestion

NOSE STRUCTURE

As far as practicable, the nose structure ahead of the cockpit should be designed to progressively collapse from the extreme nose towards the rear, upon head-on impact. The structure should be progressively stronger from the nose towards the rear. Such shock absorption minimizes the effects of sudden deceleration encountered in a head-on impact.

TOW CABLE RELEASE MECHANISMS

Operation tests on cable release mechanisms are covered in Chap. 2. Also, Chap. 1 covers limit loads to be applied to the operating handle. Releases should be designed to accommodate towline rings having a minimum diameter of 1½ inches except in cases of special design. 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 towline or towline 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 towline and/or parachute.

b. The operating handle in the cockpit should be so located, designed, and placarded that it cannot be confused with the spoiler, brake or other control handles.

c. Cables or wires used to operate the release should be protected from wear and abrasion due to normal operation, and from possible damage caused 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.

Chapter 4—EQUIPMENT

The equipment installed should be consistent with the type of operation for which certification is sought. The requirements specified in this section are basic equipment recommendations. Additional equipment should be installed for special cases if necessary or if specified in other sections of the Civil Air Regulations, such as CAR, Part 43 for instrument requirements.

Each item of equipment specified for gliders should be of satisfactory type and design. It should be properly installed and should function satisfactorily. Items of equipment for which approval is required should have been certificated in accordance with the provisions of previous regulations, or they should show compliance with pertinent Technical Standard Orders.

INSTRUMENTS AND EQUIPMENT

The following equipment for gliders with Standard Airworthiness Certificates should be installed for the particular category of operation specified. (For required instruments refer to Section 43.30 of CAR Part 43.)

a. *Contact (day) flight rules.*—

1. Instruments listed in CAR 43.30 (a), as applicable.
2. Approved safety belts for passengers and pilot(s).
3. Log-book.
4. Rigging instructions. Such instructions should be in the form of a sketch, or listed data, and should include sufficient information to facilitate proper rigging of wings, control systems, et cetera.

NOTE.—Items 3 and 4 need not be carried in the glider.

b. *Contact (night) flight rules.*

1. Equipment specified in paragraph a, above, and instruments listed in CAR 43.30 (b) as applicable.
2. Forward and rear position lights.
3. Adequate source and supply of electrical energy for the instruments, electrical and radio equipment installed.
4. Instrument lights, incorporating a suitable dimming means.
5. If fuses are used, one spare of each rating should be carried.

c. *Instrument flight.*

1. Equipment specified in paragraphs a and b of above except item b, 2, and instruments listed in CAR 43.30 (c) as applicable.
2. Two-way radio communications system appropriate to the available ground facilities.
3. Generator or battery of adequate capacity for powered gliders; battery or air pressure system for nonpowered gliders.

INSTALLATION RECOMMENDATIONS

The following recommendations apply to the installation of specific items of equipment:

INSTRUMENTS

The following recommendations are considered applicable to the installation of instruments when such instruments are required for particular operations.

Airspeed indicator.—This instrument should be so installed as to indicate true airspeed at sea level with maximum practicable accuracy, but in no event should 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.

Magnetic compass.—This instrument should be compensated so that the deviation in level flight does not exceed 10 degrees on any heading. A suitable calibration placard should be provided and should be located where it is not seriously affected by electrical disturbances and magnetic influences.

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

Gyroscopic instruments.—All gyroscopic instruments installed in gliders intended for operation under instrument flight rules should derive their energy from a power source of sufficient capacity to maintain their required accuracy in accordance with the manufacturers' recommendations.

Safety equipment installation—Safety belts.—Safety belts should be so attached that no part of the anchorage will fair at a load lower than that corresponding with the ultimate load factors equal to those specified in Chap. 1, multiplied, by a factor of 1.33.

ELECTRICAL EQUIPMENT INSTALLATION

Electrical equipment, if installed, should be free from hazards in themselves, in their method of operation and in their effects on other parts of the glider. Items of equipment which are essential to the safe operation of the glider should be designed and installed to insure that they will perform their intended functions reliably during all reasonably foreseeable operating conditions.

Storage battery.—The capacity of storage batteries should be sufficient to supply all connected loads in probable combinations and for probable durations. If corrosive electrolyte can be spilled during flight or during servicing, means should be provided to prevent hazardous corrosive effects on adjacent glider structure or equipment.

Master switch.—If electrical equipment is installed, a master switch should be provided which will disconnect the storage battery from all loads at a point adjacent to the battery.

Circuit protective devices.—Protective devices (fuses or circuit breakers) should be installed in the circuits to all electrical equipment. Protective devices in circuits essential to safety in flight should be accessible and identified so that they may be reset (or replaced, in the case of fuses) during flight. Sufficient spare fuses should be provided for this purpose. Circuit breakers should be of the tip-free type.

Instrument lights.—Instrument lights, when required, should provide sufficient illumination to make all instruments and controls easily readable. The lights should be installed in such manner that direct rays or bright reflections are shielded from the pilot's eyes.

Position and anti-collision lights.—Position lights and anti-collision lights, when required, should conform to the standards in CAR 3.700 to 3.705.

MISCELLANEOUS EQUIPMENT INSTALLATION

Seats.—Seats or chairs, even though adjustable, should be securely fastened in place, whether or not the safety belt load is transmitted through the seat.

Chapter 5—FLIGHT CHARACTERISTICS

The following recommendations are for gliders for which a type inspection authorization has been issued. The glider should meet the recommendations set forth below under all critical combinations of weight and center-of-gravity positions within the range of each for which certification is sought.

Controllability.—It should be possible, under all conditions of operation probable for the type, to make a smooth transition from one flight condition to another, including turns and slips without requiring an exceptional degree of skill, alertness or strength on the part of the pilot and without danger of exceeding the limit load factor.

Trim.—The means used for trimming should be such that after being trimmed and without further pressure upon, or movement of, either the primary control or its corresponding trim control, the glider will maintain (1) lateral and directional trim in unaccelerated forward flight at the speed for minimum sink, (2) longitudinal trim within a speed range from 1.1 V_s to 1.6 V_s.

Stability.—The glider should be longitudinally, directionally and laterally stable when trimmed within the speed range specified above. Suitable stability and control "feel" (static stability) should be required in other conditions normally encountered in service if flight tests show such stability to be necessary for safe operation. When trimmed, as indicated in the previous paragraph the characteristics of the control forces and the friction within the control system should be such that:

a. A pull on the longitudinal control should be required to obtain and maintain speeds below the specified trim speed and a push required to obtain and maintain speeds above the specified trim speed. This should be set at any speed which can be obtained without excessive control force except that such speeds need not be greater than the appropriate maximum permissible speed or less than the minimum speed in steady and stall flight.

b. The airspeed should return to within 10 percent or 5 m.p.h. of the original trim speed, whichever is greater, when the control force is slowly released from any speed within the limits defined in paragraph *a*, above.

c. Any short period oscillations occurring between stalling speed and maximum permissible speed should be heavily damped when primary controls are (1) free, and (2) in a fixed position.

d. In straight steady sideslips the aileron and rudder control movement and force increase steadily, but not necessarily in constant proportions, as the angle of sideslip is increased. At greater angles of sideslip up to that at which full rudder-control is employed, the rudder pedal forces should not reverse.

Compliance Suggestion

STABILITY AND CONTROL SPEEDS

Stability will be measured in the free-control condition. Although stability and trim are closely connected, they should not be confused. A nose-heavy glider is not necessarily unstable. If 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.

Normal flight conditions as used herein refer to normal maneuvers and speeds between V_s and V_g (placard).

Stalls.—Stall demonstrations should be conducted by reducing the speed at approximately one m.p.h. per sec. until a stall results as evidenced by an uncontrollable downward pitching motion of the glider, or, until the longitudinal control reaches the stop. It should be possible to prevent more than 15° roll or yaw by normal use of the controls during recovery, and there should be no uncontrollable tendency for the glider to spin.

When stalled during a coordinated 30° banked turn it should be possible to recover to normal level flight without encountering an excessive loss of altitude, uncontrollable rolling characteristics or uncontrollable spinning tendencies.

Spins.—Spin demonstrations should include recovery from three turn spins conducted with (1) controls held in the position normal for spins; (2) with controls crossed; and (3) with ailerons applied in the direction of rotation. It should be possible to recover from such spins in not more than one turn by applying the controls in a manner normal for recovery and without exceeding either the limiting airspeed or the limiting positive maneuvering load factor for the glider. It should not be possible to obtain uncontrollable spins by any possible use or position of the controls.

Compliance Suggestion

SPIN CHARACTERISTICS

a. In some cases, gliders are designed purposely with the ability to spin, since this would offer a means of making a safe exit from turbulence; however, the use of dive brakes or spoilers may eliminate the necessity for such a maneuver.

b. Gliders should have good control below the stall speed. There should be no tendency to fall off to one side after an accidental stall in straight flight. If such tendency exists, it should be possible to prevent it by normal use of the controls.

c. Gliders that will not spin may fall into a spiral dive. Since the speed in this maneuver is not limited as it is in the spin, the results may be serious. High performance gliders, due to their clean design, may build up dangerous speeds in spiral dives with great rapidity. Hence, there should be adequate control to recover from this condition, as well as to prevent it.

Ground handling characteristics.—All gliders should be readily controllable during takeoffs with all launching methods approved for the type. There should be no uncontrollable ground looping or porpoising tendency during landing.

Flutter and vibration.—All parts of the glider should be demonstrated to be free from flutter and excessive vibration at all speeds up to at least the minimum value of V_g permitted for which compliance with structural loading requirement has been proved.

Compliance Suggestion

TAIL BUFFETING

Tail buffeting at speeds close to the stall or in turns in the low speed range will not be considered objectionable, provided that it does not occur regularly at speeds normally used, and further provided that its magnitude is small and not detrimental.

FLIGHT TESTS

Compliance with the foregoing flight recommendations should be demonstrated by means of suitable flight test of the glider.

Test pilot qualifications.—The applicant should provide a person holding an appropriate airman certificate to make the flight test, but the FAA representative may pilot the glider during such parts of the tests as he may deem advisable.

Parachutes.—Parachutes should be worn by members of the flight test crew when deemed necessary by the FAA representative.

Instrument calibration.—The applicant should submit a report overing all the computations and tests required in connection with calibration of flight instruments. The FAA representative 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.

Compliance Suggestion

INSTRUMENT CALIBRATION

In the average glider flight test, the only instrument that must be accurately calibrated is the airspeed indicator. It must be possible to accurately determine the readings at the placard V_g, V_{taw}, V_{sf} and V_s.

Loading conditions.—The loading conditions used in flight tests should be such as to cover a normally expected range of loads and CG positions, and will be those for which the glider will be certified.

Compliance Suggestion

TEST CONDITIONS

a. In flight testing two-place gliders, the weight of the passenger and equipment should be simulated by ballast such that the normal loading conditions are represented as closely as possible.

b. In cases where the pilot is too heavy to make flight tests at extreme aft CG positions, such as might be encountered in a single-place ship with a light pilot, dumpable ballast may be added to the tail, provided the allowable gross weight is not exceeded. Since this procedure may cause the glider to enter a flat spin, extreme caution should be observed.

c. Tests will be considered unsatisfactory if tail heaviness or unfavorable spinning characteristics are evident. Tests should include CG positions resulting from the following loading conditions:

1. *Single-place gliders.*—Pilot weight 100 lbs., all equipment in rearmost position.
2. *Two-place gliders.*—100-lb. person in front seat, 220-lb. person in rear seat and all equipment in rearmost position. (It is assumed that the rear seat is aft of the cg.)
3. *Two-place side-by-side gliders.*—Same as 1, except that the CG position should be satisfactory with a 100-lb. pilot and a 100-lb. passenger without the use of removable ballast. If the above conditions cannot be met, the glider should be placarded to prevent an unfavorable combination of pilot and passenger weights.

Ballast.—Ballast may be used to comply with performance recommendations for longitudinal stability and balance provided that the place for carrying such ballast is properly designed for the weights involved and adequate means are provided for securing the ballast and for informing operating personnel of its proper use. In cases where ballast is permanently installed, it should be plainly marked, "Do Not Remove."

Compliance Suggestion

USE OF WEIGHTS

Use of weights for balancing is not recommended except in the case of two-place gliders when the absence of a passenger will cause the CG to fall outside the certified limits. In such cases a standard weight or set of weights should be supplied with the glider. A standard means of installation of these weights should be provided and plainly marked.

Maximum airspeed.—The flight tests should include steady flight at the design gliding speed (V_g) for which compliance with the structural loading requirement has been proved, except that the speed need not exceed the value of V_g specified. When high-lift devices having nonautomatic operation are employed, the tests should also include steady flight at the design flap speed V_f, except that they need not involve speeds in excess of 1.67 V_{sf}. In cases where the high-lift devices are automatically operated, the tests should cover the range of speeds within which the devices are operative.

Airspeed indicator calibration.—The airspeed indicator, of the glider type, should be calibrated in flight unless the instrument is satisfactorily calibrated by other means and properly located on the glider. A satisfactory method of calibration should be used.

Compliance Suggestion

INSTALLATION CALIBRATION

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

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

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

OPERATION LIMITATIONS

Center of gravity limitations.—The maximum variation in the location of the center of gravity for which the glider is certificated to be airworthy should be established. Suitable loading charts or schedules should be provided when necessary to assure that the glider is always properly loaded.

Towing limitations.—Gliders not aircraft towed during official flight tests should not be aircraft towed in service. A suitable placard should be provided to so inform operating personnel.

Airspeed limitations.—The maximum certified airspeed (V_{NE}) shall be limited to .90 V_D unless dive brakes are installed which will limit the terminal velocity of the glider to the V_{NE} speed selected by the designer in which case V_{NE} shall not exceed .95V_D. The maximum certified auto-winch tow speed should be limited to a value of at least 5 m.p.h. less than either the design automobile-winch tow speed (V_{taw}) or the maximum value attained in the official flight tests, whichever is lower. The maximum certified aircraft tow speed should be limited to a value of at least 10 percent 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 should be limited to 5 m.p.h. less than either the design flap speed (V_f) or the maximum value attained in the official flight tests, whichever is lower. Means should be provided to affect such limitations or suitable placards should be used to inform the operating personnel thereof.

Equipment limitations.—Gliders which are not equipped as specified in Chap. 4 should not be used for any type of operation other than visual-contact day flying. A suitable placard should be provided to inform the operating personnel of these limitations.

Chapter 6—GLIDERS WITH POWER FOR SELF-LAUNCHING

The requirements of this chapter are applicable to gliders with power for self-launching, based upon the premise that power is intended to be used for takeoff, climb and intermittent use thereafter.

Flight requirements.—The applicable flight and operational limitation requirements of either CAR 3, Acrobatic Category, or the flight and operational limitation recommendations contained herein are acceptable, provided that a glider with power for self-launching should meet the additional following conditions at the most critical loading and include the following information:

a. *Power-off condition.*—It should be demonstrated that the glider has a rate of sink not in excess of 5 ft./sec.

b. *Power-on condition.*—It should be demonstrated that the glider has with take-off power at sea level; (1) a rate of climb not less than 200 ft./min., and (2) an angle of climb of at least 1:17.

c. *Airspeed and other information.*—A placard or other suitable marking should be permanently affixed in easy view of the pilot which contains the following information:

1. Best airspeed —m.p.h. for climb (speed at which the best rate of climb was determined).
2. Best airspeed — m.p.h. for obstacle clearance (speed at which the best angle of climb was determined).
3. Best airspeed — m.p.h. for minimum sinks (speed at which the minimum rate of sink was determined with full load).
4. Fuel consumption at m.c.p. — gal./hr.

Structural requirement.—The applicable structural requirements of either CAR 3, Acrobatic Category or the structural recommendations in the handbook are acceptable.

Powerplant requirements.—The following requirements are applicable to all gliders having a powerplant for self-launching:

Scope and general design.—The powerplant installation should be considered to include all components of the powered glider that are necessary for propulsion. It should also be considered to include all components which affect their safety of operation between normal inspections or overhaul periods.

a. Reciprocating engine installations should comply with the provisions of this subpart and such other requirements as may be deemed necessary by the Federal Aviation Agency.

b. All components of the powerplant installation should be constructed, arranged and installed in a manner which will assure their continued safe operation between normal inspections or overhaul periods.

c. Accessibility should be provided to permit such inspection and maintenance as is necessary to assure continued airworthiness.

d. Electrical interconnections should be provided to prevent the existence of differences of potential between major components of the powerplant installation and other major portions of the glider which are electrically conductive.

ENGINES AND PROPELLERS

Engines.—All engines, including all essential accessories, should be subjected to the block tests and inspections prescribed in the following subparagraphs.

a. *Calibration test.*—The engine should be subjected to such calibration tests as are necessary to establish its power characteristics and the conditions for the endurance test specified in the following subparagraphs. The results of the power characteristics calibration tests should constitute the basis for establishing the characteristics of the engine over its entire operating range of crankshaft rotational speeds, manifold pressures, fuel/air mixture settings and altitude. Power ratings should be based upon standard atmospheric conditions.

b. *Detonation test.*—A test should be conducted to establish that the engine can function without detonation throughout its range of intended conditions of operation, using the lowest grade of fuel specified·

c. *Endurance test.*—The endurance test of an engine with a representative propeller should include a total of 50 hours of operation and shall consist of the following: The runs should be performed in such periods and order as are found appropriate by the Federal Aviation Agency for the specific engine. During the endurance test the engine power and crankshaft rotational speed shall be controlled within ±3 percent of the specified values.

1. A one-hour run should be conducted, consisting of alternate periods of five (5) minutes at take-off power and speed and 25 minutes at maximum continuous power and speed.
2. At the end of the one-hour period, the engine will be stopped and cooled for a (30) thirty-minute period.
3. The procedures outlined in sections (1) and (2) above will be alternately continued until the engine has accumulated a total of fifty (50) operating hours.

Propellers.—Propellers installed in certificated powered gliders should be a type which will properly restrain the engine to a speed not exceeding its maximum permissible takeoff speed during takeoff and initial climb at best-rate-of-climb speed with the engine operating at full throttle or the throttle setting corresponding to maximum permissible takeoff manifold pressure.

A propeller may be approved as a part of the installation if it satisfactorily withstands the following test.

a. Fixed-pitch wood propellers of a particular pitch, diameter, and manufacture should withstand without failure two (2) hours of operation at takeoff r.p.m.

b. Fixed-pitch metal propellers and adjustable-pitch propellers should be subjected to such additional tests as may be required by the Federal Aviation Agency.

Propeller clearance.—

a. Ground clearance should be not less than 7 inches in taxi and takeoff attitudes.

b. One-inch radial clearance between the blade tips and the glider structure, or whatever additional radial clearance is necessary to preclude harmful vibration of the propeller or glider.

c. Adequate positive clearance should be provided between any other rotating parts of the propeller and stationary portions of the glider.

FUEL SYSTEM

The fuel system should be constructed and arranged in a manner to assure the provision of fuel to each engine at a flow rate and pressure adequate for proper engine functioning under all normal conditions for which engine operation is intended.

Arrangement.—Fuel systems should be arranged as to permit any one fuel pump to draw fuel from only one tank at at ime. Gravity feed systems should not supply fuel to any one engine from more than one tank at a time unless the airspaces are interconnected in such a manner as to assure that all interconnected tanks feed equally.

Fuel tanks.—Fuel tanks should be capable of withstanding, without failure, any vibration, inertia, fluid, or structural loads to which they may be subjected in operation. Flexible fuel tank liners should be of an acceptable type.

Fuel tank tests.—Fuel tanks should be capable of withstanding the following pressure and vibration tests without failure of leakage.

a. Conventional metal tanks and nonmetallic tanks whose walls are not supported by the glider structure: A pressure of 3.5 p.s.i. for the first tank of a specific design.

b. Nonmetallic tanks, the walls of which are supported by the glider structure. Tanks constructed of an acceptable basic tank material and type of construction and with actual or simulated support conditions should be subjected to a pressure of 2 p.s.i. for the first tank of a specific design.

c. Fuel tanks which are subject to engine vibration shall, together with supports, be subjected to the following vibration test when mounted in a manner simulating the actual installation. The tank assembly should be vibrated for 25 hours at a total amplitude of not less than 1/32 of an inch while filled 2/3 full of water. The frequency of vibration should be 90 percent of the rated takeoff speed of the engine unless some other frequency within the normal operating range of speeds of the engine is more critical, in which case the latter speed shall be employed and the time of the test shall be adjusted to accomplish the same number of vibration cycles.

Fuel tank installation.—

a. The method of supporting the tanks should not be such as to concentrate the loads resulting from the weight of the fuel in the tanks. Materials employed for supporting, or padding the supporting members of the tank should be nonabsorbent or shall be treated to prevent absorption of fuel.

b. Fuel tank compartments should be ventilated and drained to prevent accumulation of flammable fluids or vapors. Compartments adjacent to the tanks should be treated in a like manner.

c. Fuel tanks should not be located on the engine side of the firewall. Not less than ½-inch of clear airspace should be provided between the fuel tank and the firewall.

d. Fuel tank capacity should not exceed 5 gallons. Tanks may be installed in personnel compartments if adequate ventilation and drainage are provided and it can be demonstrated that the location of the tank will in no way interfere with the operation of any part of the glider or the normal movements of light personnel. In all other cases, fuel tanks should be isolated from personnel compartments by means of fume and fuelproof enclosures.

Fuel system lines and fittings.—Fuel lines should be installed and supported in a manner which will prevent excessive vibration and will be adequate to withstand loads due to fuel pressure and accelerated flight conditions. Lines which are connected to components of the glider between which relative motion might exist should incorporate provisions for flexibility. Flexible hose should be of an acceptable type.

Fuel sumps.—Each fuel tank should be provided with a drainable sump having a capacity of not less than 0.10 percent of the tank capacity or 1/32 gallon, whichever is the greater. It should be acceptable to dispense with the sump if the fuel system is provided with a sediment bowl permitting ground inspection and drainage. The capacity of the sediment chamber should not be less than one (1) ounce.

Fuel vents.—Fuel tanks should be vented from the top portion of the tank. Vent outlets should be located and constructed as to minimize the possibility of the vent icing or becoming obstructed by other foreign matter and shall discharge clear of the glider. Fuel should not siphon or drain from the vent during normal operation or when the engine is in the fully retracted position.

OIL SYSTEM

Each engine should be provided with an independent oil system capable of supplying the engine with an ample quantity of oil at a temperature not exceeding the maximum which has been established as safe for continuous operation except that two-cycle engines using a single tank to supply a fuel/oil mixture to the engine need not comply with this section.

Oil tanks.—Oil tanks shall be capable of withstanding without failure any vibration, inertia, fluid, or structural loads to which they may be subjected in operation. Flexible oil tank liners shall be of an acceptable type.

Oil tank tests.—

a. Oil tanks should be subjected to the same tests as fuel tanks except pressure test at 5 p.s.i.

b. For a two-cycle engine using a common tank to supply a fuel/oil mixture to the engine, the pressures applicable to fuel tank installation should apply.

Oil tank installation.—Oil tank installation should comply with the requirements of fuel tank installations.

Oil system lines and fittings.—Oil system lines should comply with the same installation requirements as used for the fuel system lines and fittings except that the inside diameter of the engine oil inlet and outlet lines should not be less than the diameter of the corresponding engine oil pump inlet and outlet.

Oil breather lines.—

a. Engine breather lines should be arranged so that condensed water vapor which might freeze and obstruct the line cannot accumulate at any point.

b. Breathers should discharge in a location which will not constitute a fire hazard in case foaming occurs and so that oil emitted from the line will not impinge on the pilot's windshield. The breather should not discharge into the engine air induction system. Drainage through the breather should not be possible when engine is fully retracted.

COOLING

The powerplant cooling provision should be capable of maintaining the temperature of all powerplant components, engine parts and engine fluids (oil and coolant), at or below the maximum established safe values under critical conditions of ground, takeoff, and climb operation.

Cooling test.—The cooling test to determine that the maximum temperatures which the engine can withstand will not be exceeded should be conducted by starting and running the engines on the ground until normal engine warmup temperature has been stabilized. The glider shall then proceed with normal takeoff and climb at best-rate-of-climb speed. The power used should be takeoff power and speed and maximum continuous power and speed. The climb should continue for approximately 5 minutes after the temperature peak or dropoff, to assure that stabilization had been reached.

INDUCTION SYSTEM

The engine air induction system should permit supplying an adequate quantity of air to the engine under all conditions of operation.

Induction system ducts.—Induction system ducts should be provided with drains which will prevent the accumulation of fuel or moisture in all normal ground and flight attitudes. Drains should not discharge in a location which will constitute a fire hazard. Ducts which are connected to components of the glider between which relative motion may exist should incorporate provisions for flexibility.

Induction system screens.—Where induction system screens or filters are employed, they should be located upstream from the carburetor. It should not be possible for fuel to impinge upon the screen.

EXHAUST SYSTEM

The exhaust system should be constructed and arranged in such a manner as to assure the safe disposal of exhaust gases without the existence of a fire hazard or contamination of the air in personnel compartments with carbon monoxide. Unless other suitable precautions are taken, all exhaust system components should be separated from adjacent flammable portions of the glider by means of fireproof shields. Exhaust gases should not be discharged within dangerous proximity of any fuel or oil system drains. All exhaust system components should be ventilated to prevent the existence of points of excessively high temperature.

Exhaust manifold.—Exhaust manifolds should be made of fireproof, corrosion-resistant materials, and shall incorporate provisions to prevent failure due to their expansion when heated to operating temperatures. Exhaust manifolds should be supported in a manner adequate to withstand all vibration and inertia loads to which they might be subjected in operation. Portions of the manifold which are connected to components between which relative motion might exist should incorporate provisions for flexibility.

FIREWALL AND COWLING

Firewalls.—All engines, fuel burning heaters, and other combustion equipment which are intended for operation during takeoff or flight should be isolated from the remainder of the glider by means of firewalls, shrouds, or other equivalent means.

Firewall construction.—

a. Firewalls and shrouds should be constructed in such a manner that no hazardous quantity of liquids, gases, or flame could pass from the engine compartment to other portions of the glider.

b. Firewalls and shrouds should be constructed of fireproof material and protected against corrosion.

c. Where the engine, during operation, is situated so that the probable path of an engine fire would not contact the glider structure, firewalls and shrouds may be constructed of fire-resistant materials.

Cowling.—

a. Cowling should be constructed and supported in such a manner as to be capable of resisting all vibration, inertia, and air loads to which it may normally be subjected. Provision should be made to permit rapid and complete drainage of all portions of the cowling in all normal ground and flight attitudes. Drains must not discharge in locations constituting a fire hazard.

b. Cowling should be constructed of fire-resistant material. All portions of the glider lying behind openings in the engine compartment cowling should also be constructed of fire-resistant materials for a distance of at least 24 inches aft of such openings. Portions of cowling which are subjected to high temperature due to proximity to exhaust system ports or exhaust gas impingement should be constructed of fireproof material.

Retractable engine assembly.—

a. Engine assemblies which retract into any part of the glider should be installed in such a manner as to avoid the possibility of fire occurring or other hazardous effects due to residual heat, with the engine in the retracted position.

b. Fuel or oil should not drain from the engine, its components, or items of equipment installed on the engine assembly when the engine assembly is in the retracted position.

POWER PLANT CONTROLS AND ACCESSORIES

Power plant controls.—

a. *Throttle controls.*—A throttle control should be provided to give independent control for each engine. Throttle controls should afford a positive and immediately responsive means of controlling the engine.

b. *Ignition switches.*—Ignition switches should provide control for each ignition circuit on each engine.

c. *Mixture controls.*—If mixture controls are provided, a separate control should be provided for each engine. The controls should be grouped and arranged in such a manner as to permit both separate and simultaneous control of all engines.

Location.—All powerplant controls should be located so as to provide convenience in operation including provisions to prevent inadvertent operation. The controls should be located and arranged so that when the pilot is seated it will be readily possible for him to obtain full and unrestricted movement of each control without interference from either his clothing or the cockpit structure.

Markings.—All powerplant controls should be plainly marked as to their function and method of operation. Controls for fuel tank selector valve should be marked to indicate the position corresponding to each tank and to all existing cross feed positions. When more than one fuel tank is provided, and if safe operation depends upon the use of tanks in a specific sequence, the fuel tank selector controls should be marked adjacent to or on the control to indicate to the pilot the order in which the tanks must be used.

Construction.—Controls should maintain any necessary position without constant attention by the flight personnel and shall not tend to creep due to control loads or vibration. Flexible controls should be of an acceptable type. Controls should have adequate strength and rigidity to withstand loads without failure or excessive deflection.

Movement.—The direction of movement of controls should be as follows:

a. Throttle: Forward to increase forward thrust.

b. Propeller: Forward to increase r.p.m.

c. Mixture: Forward or upward for rich.

ACCESSORIES

Powerplant accessories.—Engine-driven accessories should be of a type satisfactory for installation on the engine involved and shall utilize the provisions made on the engine for the mounting of such units. Items of electrical equipment subject to arcing or sparking should be installed so as to minimize the possibility of their contact with any flammable fluids or vapors which might be present in a free state.

EQUIPMENT

Required basic equipment.—The following table shows the basic equipment required for type and airworthiness certification of a glider:

Flight instruments.—

a. Airspeed indicator.

b. Altimeter.

c. Magnetic direction indicator.

Powerplant instruments.—For each engine or tank.

a. Tachometer.

b. Fuel quantity indicator.

c. Oil temperature indicator (not for two-cycle engines).

d. Oil pressure indicator (not for two-cycle engines).

e. Powerplant instrument lines carrying flammable fluids or gases under pressure should be provided with restricted orifices or other safety devices at the source of the pressure to prevent escape of excessive fluid or gas in case of line failure.

Chapter 7—MISCELLANEOUS RECOMMENDATIONS

GENERAL

Standard weights.—In computing weights the following standard values should be used: (Also see Chap. 6.)

Pilot and passengers	170 lbs. per person
Parachutes	20 lbs. each
Water	8.5 lbs. per gallon

Leveling means.—Adequate means should be provided for determining when the glider is in a level position.

It is most advisable to provide a leveling means in order that rigging may be checked, and so that necessary load tests may be conducted. There is no standard definition of level position for gliders. However, any arbitrary reference may be used, such as a longeron or a line between bolt holes in one wing root fitting. All rigging instructions should require only the more common and inexpensive measuring devices, such as straight edges, ordinary scales, and spirit levels. Only reasonable accuracy in any measurement normally is necessary.

☆ U.S. GOVERNMENT PRINTING OFFICE: 1961 O - 61702