Civil Aeronautics Manual 04

Airplane . . . Airworthiness

Revised July 1, 1944

U. S. DEPARTMENT OF COMMERCE
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Civil Aeronautics Administration
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Civil Aeronautics Manual 04

AIRPLANE AIRWORTHINESS

● Contains all requirements of Part 04 of the Civil Air Regulations, and the CAA interpretation

Revised July 1, 1944

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INTRODUCTORY NOTE

This edition of Civil Aeronautics Manual 04 includes all revisions [1] up to July 1, 1944, and for ease of reference incorporates the Civil Air Regulations Part 04 in smaller type immediately preceding the corresponding section of the Manual.

The Manual material herein is not mandatory and is intended only to explain and to show acceptable methods of complying with the pertinent requirement. Alternative methods of showing compliance may be used at the option of the applicant. The function of the Civil Aeronautics Administration is to examine such technical data and to conduct or witness such inspection and testing as may be necessary to demonstrate compliance with the Regulations.

[1] With the time and personnel available it was found impossible to include certain material pertaining to powerplant installation and to ground oads. This material now incorporated in several releases will be issued as revision sheets to this Manual.

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Airplane Airworthiness

04.0 GENERAL

04.00 SCOPE.

The airworthiness requirements set forth in this Part shall be used as a basis for obtaining airworthiness or type certificates: *Provided*, That (1) deviations from these requirements which, in the opinion of the Administrator, insure the equivalent condition for safe operation and, (2) equivalent requirements of the United States Army or Navy with respect to airworthiness, may be accepted in lieu of the requirements set forth in this Part. Unless otherwise specified, an amendment to this Part will apply only to airplanes for which applications for type certificates are received subsequent to the effective date of such amendment.

The requirements are based on the present development in the science of airplane design. Experience indicates that, when applied to conventional types of design, they will result in an airworthy aircraft. New types of aircraft and new types of construction may, however, incorporate features to which the requirements cannot be applied logically. In such cases it is necessary that the applicant show:

a. That the requirement is not applicable because the airplane is shown to be unconventional with respect thereto.

b. That the objective on which the requirement is based can be shown to have been met.

"Unconventional" refers not only to deviations from the conventional with respect to general design and design details, but also with respect to size. As the requirements of Part 04 have been based largely on experience with airplanes weighing less than 30,000 pounds, they cannot logically be extended to aircraft of considerably greater size. Appendix I, containing suggestions on the trend of the requirements for large airplanes, has therefore been included for the information of designers.

04.01 AIRPLANE CATEGORIES.

At the election of the applicant, an airplane may be certificated under the requirements for a particular category according to the intended use of the airplane. Sections of this Part which affect only one particular category are designated by a suffix added to the appropriate section numbers, as follows:

Normal Category—suffix "N"
Transport Category—suffix "T"
Acrobatic Category—suffix "A"

All sections not designated by a suffix are applicable to all categories, except as otherwise specified.

At this time the only category defined in Part 04 of the Civil Air Regulations identified by means of the suffix system described by the text above is the transport category.

04.02 AIRWORTHINESS CERTIFICATE.

The airworthiness requirements specified hereinafter shall be used as a basis for the certification of airplanes: *Provided*, That an airplane manufactured in accordance with, and conforming to, the currently effective aircraft specifications issued therefor, will be eligible for an airworthiness certificate if the Administrator determines such airplane is in condition for safe operation: *Provided, further*, That an airplane which has not demonstrated compliance with the airworthiness requirements specified hereinafter but which in the opinion of the Administrator, is in condition for safe operation for experimental purposes or for particular activities, will be eligible for an airworthiness certificate.

An airworthiness certificate is a document issued by the CAA certifying that the aircraft is considered airworthy when operated in accordance with the operation limitations (including restrictions, if any) listed on the Aircraft Operation Record attached thereto. There are three types of identification mark designations, the NC, NR, and NX.

NC designation.—This type of identification mark is assigned to those airplanes which fully comply with the airworthiness requirements of the Civil Air Regulations.

NR designation.—This type of identification mark is assigned to those airplanes which comply with the airworthiness requirements of the Civil Air Regulations except in some limited respect but are in condition for safe operation for particular activities. In such cases the lack of compliance with certain of the requirements will be compensated for by operation limitations and restrictions other than those normally employed under the NC designation. Where possible, use should be made of the NC designation; i. e., the NR should not be considered as a means of avoiding the necessity for showing compliance with the usual airworthiness requirements. Further, the NR designation should be applied only to aircraft which are ineligible for the NC designation. In general, NR aircraft will be those used in an industrial operation. However, each case will be handled on its individual merits upon presentation of proper data to the CAA.

NX designation.—This type of identification mark is assigned to those airplanes which have not demonstrated compliance with the airworthiness requirements of the Civil Air Regulations, but which in the opinion of an authorized representative of the Administrator exhibit no apparent unairworthy features and are in condition for safe operation for experimental purposes.

04.03 DATA REQUIRED.

When technical data are submitted as a basis for an airworthiness certificate they should include information which, in conjunction with suitable inspection and test procedure, will enable the Administrator to determine whether the aircraft is eligible for such certificate. All technical data submitted by the applicant for the Administrator's file will be held confidential and will be used only in connection with the airworthiness rating of the airplane or airplanes to which such data apply; provided, however, that the Administrator may at his discretion make such use of the confidential data as is required in the interests of public safety. Access to confidential data will be provided to accredited representatives of the holder of, or applicant for, a pertinent type certificate. Confidential data will not be used for reference purposes in connection with the repair, alteration or remodeling of certificated airplanes by persons other than the holder of the pertinent type certificate without the written consent of such holder unless he is out of business or has given the Administrator blanket permission for such use.

A technical data file for each model airplane for which a type certificate is desired is necessary. This means that a complete file for each model is required to the extent that reference can be made to previously submitted data for a similar model. When the Application for Type Certificate and the Application for Production Certificate forms are submitted, they should refer to the particular models involved. When more than one model is covered by the technical data submitted, separate applications for type certificate should be executed and forwarded for each model.

When data are submitted to a branch office of the CAA, an extra copy of the three-view drawing, main assembly and installation drawings, drawing lists, applications, reports, all electrical system data, including a running load analysis should be included for the Washington office files. Failure to follow this procedure may lead to serious and undesirable delay for the manufacturer in the examination of data requiring the attention of the Washington office.

04.031 Data required for airworthiness certificate. When an airworthiness certificate is sought and a type certificate is not involved, data which are adequate to establish compliance of the aircraft with the requirements listed hereinafter shall be submitted to the Administrator.

General.—When an airworthiness certificate only is desired, the data required is dependent on the particular problems involved in the design concerned. The minimum data needed as a basis for the issuance of an airworthiness certificate for a single airplane for which a type certificate is not sought or has not previously been issued, are as follows for the NC designation:

a. A three-view drawing of the airplane, to a designated scale, specifying the external dimensions, manufacturer's designation, engine model designation, design weight, empty weight, wing and control surface areas, seating arrangements, fuel and oil capacity, baggage capacity (in pounds) and equipment supplied.

b. A complete explanation of the current status of the model airplane involved.

c. Such additional technical data as are deemed necessary by the Administrator. The applicant is free to develop and present any means he can for showing compliance with the specified requirements. Reports on satisfactory strength tests may be substituted for strength analyses. In most cases it is desirable that a personal contact be made to supplement the material presented for consideration.

The above may, and usually will, also apply to an aircraft for which an NR identification mark is desired, but will not in general, apply to an aircraft for which an NX identification mark is desired, since airplanes in this latter designation may be certificated by a CAA representative upon a satisfactory visual inspection only.

Service types.—In the case of service type airplanes, the additional data deemed necessary will include at least the following:

a. A comparison with the service type, describing the differences, if any.

b. Such drawings and technical data as are necessary to substantiate all of the differences in the primary structures described in accordance with *a* above.

c. A copy of the Army or Navy specification(s) pertinent to the basic service type.

d. Summary data, certified to by the Air Corps or the Bureau of Aeronautics, whichever agency is involved, making clear the exact status of its final approval and acceptance of the service type, particularly with respect to gross weight, design speeds, equipment, approved center of gravity range, flutter and vibration, and flight characteristics.

e. One copy each of the complete drawing and equipment lists.

04.032 Data required for type certificate. Data which are adequate to establish compliance of the aircraft with the airworthiness requirements listed hereinafter and which are adequate for the reproduction of other airplanes of the same type shall be submitted to the Administrator. The procedure for submitting the required data, the technical contents of such data, and the methods of testing aircraft with respect to the prescribed airworthiness requirements shall be in accordance with Civil Aeronautics Manual 04, Airplane Airworthiness.

As a basis for type certification the data listed in the following paragraphs should be submitted.

Drawings

1. A set of drawings should be submitted in blueprint form, or equivalent. Drawings should be folded to a size approximately 9″ x 12″, and should contain at least the following information:

a. The manufacturer's designation of the original model to which each drawing applies.
b. All dimensions essential to the reproduction of an identical airplane in respect to structural strength and dimensions.
c. All dimensions essential for checking the structural analysis.
d. Specifications of all materials used in the primary structure, including the guaranteed physical properties in the case of materials the strength properties of which are developed through manufacturing processes, and specifications of all bolts, nuts, rivets and similar standard parts essential to the strength of the structure.
e. Details of the primary structure, seating arrangement, exits, control systems, powerplant installations, equipment installations, and other factors affecting the airworthiness of the airplane, except that adequate photographs may be substituted for drawings of the powerplant installation, including cooling and exhaust systems. Such photographs shall be made from marked negatives indicating the dimensions and materials of the piping and fittings. In any case diagrammatic layouts of the fuel and oil systems should be submitted.
f. Revision blocks stating the nature of the revision and the date it was made. The checking of revised drawings of relatively large size will be expedited if the change letters are printed in two perpendicular margins opposite the revision on the drawing in addition to being included on the revision block. Each change must be adequately described in the revision block of the drawing unless it is so described in a copy of a shop change notice attached to the changed drawing when submitting it to the Administrator for approval.
g. A three-view drawing of the airplane, to a designated scale, specifying only the external dimensions of the airplane (including dimensions and areas of wing and control surfaces) and the airplane and engine model designations. Do not include items of equipment on the three-view drawing, as this serves no useful purpose, the equipment being covered by a separate list. Likewise, all references to performance should be omitted.
h. An electrical wiring diagram and conduit installation drawings containing information pertaining to the rating, manufacturer's name and the model designation of items of electrical equipment.

2. Attention to the following list of frequently omitted items will be of assistance in expediting the work of the Administrator:

a. Complete dimensions and references to all standard parts such as bolts, nuts and rivets used in assembling a given part.
b. Adequate material specifications and bend radii on all shop drawings.
c. Location and details of control system pulleys and of control surface stops.
d. Suitable assembly drawings showing the method of assembly and calling out the detail parts required for all major installations.
e. Adequate drawings and descriptions of the operation retractable landing gear control devices.
f. Drawings to show provision for expansion in oil tanks.
g. Details of measuring devices for fuel and oil tanks.
h. Complete structural drawings of all components.

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

Drawing List

A drawing list should be submitted in duplicate, listing in numerical order or by suitable classification the number and title of each drawing submitted. The drawing list should include references to all drawings originally submitted in connection with applications for airworthiness rating of other models and which apply to the model in question without change. The drawing list should also indicate, by letter, the latest revision of each revised drawing. In preparing the lists it is desirable that the drawings be grouped according to the airplane component concerned such as Wing Group, Fuselage Group, etc. Within each group the drawings should be listed in consecutive order.

In the case of large airplanes the list of drawings becomes very extensive. If the manufacturer uses a straight numerical numbering system it may become necessary to supplement the official drawing list arranged according to consecutive numbers by another list arranged according to components and subassemblies. The latter list will be used only as a ready reference for locating information in the file and need not be kept up to date according to the latest drawing changes. Such supplementary lists need not be submitted in duplicate.

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

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

SAMPLE DRAWING LISTS

I. List when only one new model is involved.

MODEL 10 DRAWING LIST

Drawing No.	Change	Title	Originally submitted for model
		Wing group	
22001	B	Frame assembly, outer wing	10
22002	K	Spar assembly, outer front	10
		Fuselage group	
		Powerplant group—etc.	

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II. List when new model has only minor variations from previously approved basic model (10).

MODEL 11 DRAWING LIST

With the exception of the drawings listed under *a* and *b* below the drawing list of Model 10 applies also to Model 11.

a. Model 10 drawings not pertinent to Model 11. (See arrangement under I above.)

b. Drawings pertinent to Model 11 which are in addition to Model 10 list less group *a* above. (See arrangement under I above.)

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

MODEL 15 DRAWING LIST

Drawing No.	Change	Title	Originally submitted for model
		Wing group	
25001	--------	Frame assembly, outer wing	15
25002	--------	Spar assembly, outer front	15
25003	A	Fitting, front spar, root attachment	10
25004	E	Fitting, front spar, strut, etc	11

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Equipment Lists.

Lists specifying the equipment supplied with each airplane should be submitted. The location, weight and model designation of each item of equipment, including the additional weight necessary for installation should be specified. A recommended form for equipment lists is shown below This list shows a method of handling items in a simplified form which may include a number of related models and which makes it unnecessary to prepare separate lists for each model.

RECOMMENDED FORM FOR EQUIPMENT LISTS

REQUIRED EQUIPMENT

Item No.	Item	Make and model	Horizontal arm from datum [1]	Weights used on models				
				A	B	C	D	E
1	NACA Cowl	Drwg. No. 39700	−48	28.0	28.0	31.0	33.0	33.0
2a	Propeller—Wood	Hartzell 669M	−67	42.0	42.0			
2b	Propeller—Fixed Metal	Curtiss 55501	−66			51.0	51.0	51.0
3	Starter—Direct Electric	Eclipse E-80	−36	20.0	20.0	20.0	20.0	20.0
4	Generator—Engine Driven	Eclipse LV-180	−36			18.0	18.0	18.0
5	Storage Battery	Exide 6-T-S-7-1	−32			36.0	36.0	36.0
6	Battery Box							
7	Position Lights	Grimes A						
8	Landing Lights	Grimes Retractable	9			11.0	11.0	11.0
9	Instruments: a. Compass. b. Altimeter. c. Tachometer.							
10	Safety Belts (5)	Rusco AE-200						
11	Fuel Tanks—Two 35 gal	Drawing No. 39724	30					
12	Oil Tanks—One 5 gal	Drawing No. 39745	−32					
13	Bonding							
14	Oil Cooler	Drawing No. 17091	−48					
15	Wheels (List tires when a special type or size is required).	Hayes 651 M	7	75	75	75	75	75
16a	Carburetor Air Heater	Drawing No. 32001	−40					
16b	Carburetor Air Heater	Drawing No. 32002	−40					

OPTIONAL EQUIPMENT

Item No.	Item	Make and model	Used on models	Total installation weight (net [1] increase over required items)	Horizontal arm from datum [1]
20	Flares—Parachute	International Mark I 3-1½ Minute Electric	All	17.0	106
21	Extra 20 gal. fuel tank (Plus 6 gal. oil tank—No increase).	Drawing No. 39670	C, D, E	12.0	79
22	Special Upholstering:				
	a. Leather	Full Grain	All	30.0	45
	b. Leather Seats only	Full Grain	All	11.0	41
23	Special Instruments:				
	a. Large Compass	Pioneer Straitflight	All	6.0	−13
	b. Thermocouple Installation	Weston 602 (Single Lead)	All	1.5	−13
	c. Etc				
24	Generators:				
	a. Engine driven:				
	1. Bosch	LE 70/12 R5	A, B	13	−34
	2. Bosch	LE 70/12 R5	C, D, E	−6.0	−32
25	Radio Equipment:				
	a. Receivers:				
	1.	RCA-AVR-7 Series	All	24	+9
		(Chassis and Power Supply)		18	−10
		(Controls and Wiring)		6.0	
	b. Compasses:				
	1.	RCA-AVR-8	All	64.0	20
		(Chassis and Power Supply)		43.0	15
		(Hoop Assembly)		10.0	40
		(Controls and Wiring)		11.0	15

[1] Distances measured aft of the datum are positive, those forward are negative.

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

a. The effects of the equipment installation on the aircraft structure. The examiner ascertains that satisfactory analyses and drawings are submitted for such items as batteries, radios, extra fuel tanks, flares, etc.;

b. That items for which approval is required, such as wheels, safety belts, etc., are of an approved type; and

c. The effects of the equipment installation weights on the longitudinal balance of the airplane. (See "Weight and Balance Report," p. 9.)

The following information is often incorrectly or incompletely supplied in preparing the required equipment lists. Careful attention to these details will prevent delays from this source.

a. The model designation of both propeller hub and blades should be specified together with the range in diameter for which approval is desired. Information regarding constant speed control units, etc., should be included.

b. Optional fuel and oil tank installations should be specified with pertinent weights, capacities and locations thereof. When the horizontal arm of the fuel or oil in the tank is different from the arm of the tank installation the list should include both arms.

c. Items which include a number of distributed parts, such as a radio, should be listed with the installation weight and its arm for balance purposes, but the location of the main units should also be given.

d. Wheels, tires and such items should be specified by model designation, name of manufacturer and size, and the weights and horizontal arms should be given.

e. The weights of certain items such as position lights, safety belts, and special throttle controls need not be specified but the list should include the model designation and name of manufacturer in order that it may be determined whether or not they are of an approved type.

f. For special items such as carburetor heaters, oil coolers, etc., which may not have a model designation, the pertinent drawing number should be specified.

g. The weight of items of equipment should be given to the nearest pound.

Preliminary Weight and Balance Report

A report should be submitted in which the range of center of gravity locations for which rating is sought is determined versus weight and with respect to suitable reference planes or lines. This report determines the *CG* positions and the weights to be used for design purposes. It should include the Balance Diagram (shown in figure 1) and the Weight Table. If the limits of the final range as determined during the Type Inspection appreciably exceed the design limits, use of the final values should be substantiated insofar as they affect the design computations.

Balance Diagram

The balance report should include a diagram showing the location of the centers of gravity of the component parts of the airplane and its contents, and the location of a suitable reference chord for the wing system, and the location of the assumed center of pressure of the horizontal tail. The locations of these items should be indicated by reference to suitable horizontal and vertical planes.

A suggested form for the balance diagram for an airplane in the one to five place size range is shown in figure 1.

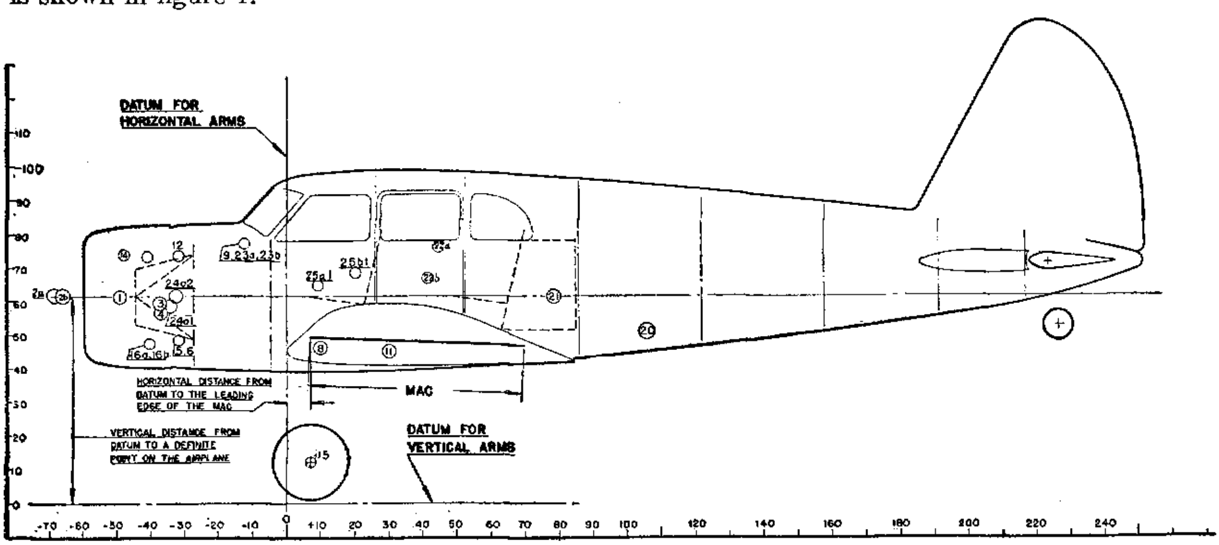

Figure 1.—Sample balance diagram.

The amount of detail necessary in preparing an acceptable balance diagram will vary considerably depending upon the size of the project and the variations in possible loading conditions.

When large variations in the amount of equipment are expected, it may be desirable to use a separate equipment balance diagram. The balance diagram should include the following:

a. Outline of the airplane (side view).

b. Horizontal and vertical scales.—For horizontal arms it is preferable that the datum be chosen at some definite and accessible point on the airplane, such as a point at the leading edge of the wing. This facilitates checking in the field. Distances aft of this point are generally assumed as positive and those forward as negative. For vertical arms the datum may be chosen at some arbitrary location below the extended landing gear, so that all distances are up and positive.

c. Item designations.—These designations (usually numbers) should correspond with the designation used in the weight table and, when possible, with the designations used in the equipment list and weight and balance reports.

d. Item locations.—The various items should be shown in the proper location on the outline noted in (*a*) above. Such location may be indicated by a small circle together with the item designation noted in (*c*) above.

e. Dimensions.—The following should be given: (1) Length of MAC. (2) Horizontal distance from datum to the leading edge of the MAC. (3) Vertical distance from the datum to a definite and accessible point on the airplane such as the centerline of the propeller.

For large airplanes, and especially airline aircraft having many possible loading conditions, a more detailed balance diagram is necessary in order to permit a ready check of many of the component items. The following method is suggested as one possible means of solving the problem satisfactorily for practical use:

a. Prepare an outline drawing (side view), with established vertical and horizontal reference planes located relative to some fixed point on the airplane structure. Include certain additional parallel auxiliary reference planes, called stations, designated by their distance in inches from the established reference planes. If possible, such station designations should agree with designations normally used in specifying stations of the structure and in locating various equipment of structural items shown on major assembly drawings or equipment installations.

b. In cases where it is not practical to show each item as an individual number on the diagram, due to the large number of items involved, the *CG*'s of groups of related items may be determined and each such *CG* shown as a single item on the balance diagram.

Weight Table

The balance report should include a table or list of the weights of all component parts of the airplane and its contents. The weights shown in the table should be broken down and itemized so that they may readily be used in the structural analysis reports of the individual components such as Wing, Fuselage, Engine Nacelles, etc.

Structural Reports

A structural report should be submitted in which the strength of the structure is determined with reference to the strength requirements specified. The structural report should include the computations of the required limit loads and should demonstrate the ability of the structure to develop the required factors of safety with respect to these loads either by analytical methods satisfactory to the Administrator or by reference to authenticated test data, or by a combination of both. Note that the examination of reports can be discontinued in event that they contain errors which render them unsatisfactory. In order to avoid delays in the checking of data it is recommended that all computations be given an independent check by the manufacturer and be signed by both the original computer and the checker.

Structural Analysis

Computations submitted as part of the structural report should include a table, or tables, including the minimum margins of safety computed for all structural members and should bear the signature of the responsible engineer or engineers.

The history of past airplane model designs shows that in practically all cases the original design weight is increased sometime during the life of the project. In order that the approval of such changes may be handled without undue effort and resultant delay it is essential that each structural analyses report contain a table summarizing the minimum margins of safety determined in the body of the report. Such report tables should include the name of the element involved (such as spar), design condition, margin of safety, and page number reference.

Test Reports

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

In the preparation of tests reports submitted to the Administrator as partial proof of a given structure it is essential that they contain as a minimum the following information:

a. Determination of tests loads (including references to pertinent page and number of stress analysis report).

b. Distribution of loads during test.

c. Description and photographs of test set-up. (Detail views are necessary in some cases.)

d. Description of method of testing.

e. Results of tests, including photographs of structures found to be critical.

f. Log of deflection data. (Including sketches to show location of points at which the deflections were measured.)

g. Curves of deflection vs. load for each such point to permit determination of any evidence of permanent set.

h. Signature of Administrator's representative(s) and manufacturing engineer(s) who witnessed the test.

i. Signature of company engineer(s) responsible for test report.

Schedule for Submitting Data

When submitting data for a type certificate for large projects that may require the attention of the Administrator for an extended period of time, it is desirable that information as to the schedule of approximate dates when the data will be received by the Administrator be forwarded at an early date. A sample of a preferred schedule of this nature is shown below.

SAMPLE SCHEDULE FOR SUBMISSION OF MODEL 120 TECHNICAL DATA

1.	Initiate Correspondence Regarding Plans for New Project	July 1, 1943
2.	Conferences and Correspondence re Special or Unconventional Features	August 1, 1943
3.	Structural Research Data	January 1, 1944
4.	Determination of Applied Loads	February 15, 1944
5.	Preliminary Weight and Balance Report	February 15, 1944
6.	Drawing and Equipment Lists	May 1, 1944
7.	Wing Group	May 1, 1944
8.	Engine Mount	May 1, 1944
9.	Landing Gear	May 15, 1944
10.	Tail Wheel	May 15, 1944
11.	Nacelle	May 15, 1944
12.	Tail Group	June 1, 1944
13.	Control System	June 1, 1944
14.	Fuselage	June 1, 1944
15.	Miscellaneous Tests	With Pertinent Group
16.	Control Surface and Control System Proof and Operating Tests; Dynamic Drop Tests	October 1, 1944

NOTE. The above dates represent the best present estimate of the dates at which the reports with assembly and detail drawings necessary for check can be submitted to the Administrator.

Data for transport planes.—For airplanes to be certified in the Transport Category, the following information, in addition to that outlined above, should be submitted at the earliest practicable date:

a. A drawing and sufficient description of the flap control to indicate its compliance with the transport flap control requirements of 04.434–T.

b. A drawing of the trimming controls which will indicate the manner in which they comply with the trim control requirements of 04.439–T.

c. A drawing or diagram of the wheel brake system, together with an indication of the element(s) to be considered "lost" for the purpose of showing compliance with the brake requirements of 04.445–T. Also, a description of the method to be used in demonstrating, with the "lost" element(s) inactive, that at least 50 percent of the normal deceleration during landing may be developed.

d. A statement by the manufacturer of his approved maximum operating pressure or its equivalent for the brake installed in the airplane.

e. A drawing of the pilot's enclosure showing azimuthal and vertical angles of vision with respect to the longitudinal axis of the airplane and the angular orientation of this axis in respect of the horizon for the normal ground attitude of the airplane; the attitude in a climb at the best rate of climb speed with METO power on all engines; and the attitude during a glide at 1.3 V_{s0} with the airplane in the landing configuration and the throttles closed.

f. It is also strongly recommended that during this period a mock-up, which will include all of the crew stations and simulate all of the controls and instruments to be used by the crew as well as the windshield and any windows, be constructed by the manufacturer and made available to the representatives of the CAA for examination.

04.04 INSPECTION AND TESTS.

Authorized representatives of the Administrator shall have access to the airplane and may witness or conduct such inspections and tests as are deemed necessary by the Administrator. Prior to, or at the time of, presentation of the airplane for official flight tests, the applicant for an airworthiness or type certificate shall submit to the Administrator a detailed report of pertinent flight tests conducted, and satisfactory proof of conformity of the airplane with the technical data submitted to the Administrator.

For Type Certification.—The procedure for inspection and tests where type certification is involved is outlined below.

For Airworthiness Certification.—The procedure for inspection and tests where an airworthiness certificate only is desired and type certification is not involved will include such parts of the following provisions as are deemed necessary by the Administrator.

04.05 PROCEDURE FOR TYPE CERTIFICATION.

Acceptable procedures for type certification are outlined in Civil Aeronautics Manual 04.

Examination of Data.

Partial data.—The Administrator will examine partial units of the required technical data provided that each such unit is complete in itself with respect to both analyses and drawings.

Discontinuance of examination.—Examination of any technical data, including drawings, submitted in connection with an application for airworthiness rating, will be discontinued if errors, omissions, or lack of references are found which, in the opinion of the Administrator, render the data unsatisfactory as a basis for proving compliance with the airworthiness requirements. The examination will be continued upon correction of the data to the Administrator's satisfaction. Minor errors and omissions, the effects of which can be readily evaluated, will not constitute cause of discontinuing examination of technical data.

Structural inspection.—An official representative of the Administrator will conduct such inspections of the structure and methods of fabrication as are deemed necessary by the Administrator prior to completion of the airplane and will witness structural tests in compliance with these regulations.

Type inspection authorization.—A type inspection will be authorized upon fulfillment of the following:

a. Completion of examination of the structural report and drawings and correction by the applicant of all errors and omissions which, in the opinion of the Administrator, must be corrected before authorization of the type inspection.

b. Completion, and acceptance by the Administrator, of all structural tests required as part of the structural report or to prove compliance with the requirements herein specified.

c. Submission of the necessary test reports and their acceptance by the Administrator.

Type Inspection Procedure

The type inspection should consist of a ground inspection and a flight test of an airplane built to conform with the technical data previously submitted and approved and on which the authorization of the type inspection was based. The following subparagraphs should be complied with in connection with the type inspection.

Statement of conformity.—The manufacturer should present to a designated inspector of the Administrator a certified statement of conformity, upon a form to be supplied by the Administrator, in which his chief engineer or other responsible technical representative should certify that the airplane submitted for type inspection has been manufactured in accordance with the latest technical data submitted to and approved by the Administrator (including all revisions and additions required by the Administrator in connection with authorization of the type inspection) except for any deviations therefrom, which should be listed and described.

Weight and balance report.—The airplane should be weighed and its balance determined in the presence of an authorized CAA representative, and the manufacturer should submit to such representative a complete report covering the determination of the weights and center of gravity locations for which certification is desired. A datum should be selected which can be easily identified by one unfamiliar with the airplane and which can be included in the aircraft specification.

A recommended form for weight and balance reports is given in Appendix II. This report is based upon the actual weight of the airplane and the loadings as flown in the type tests. 04.7 should be noted in connection with this section.

Applicant's flight test report.—Prior to, or at the time of, presentation of the airplane for flight tests, the applicant should submit to the Administrator's representative a detailed report of flight tests of the airplane involved. This report should be signed by the applicant's test pilot who should certify that the airplane has been flown by him in all maneuvers necessary for proof of compliance with the flight requirements and found to conform therewith, except that for very large airplanes this procedure may be modified as deemed necessary by the Administrator. In order to expedite checking of this report it is advisable that the results of the applicant's flight tests be recorded on a form of the type used by the Administrator in connection with the required flight tests. Copies of this form may be obtained from the Administrator's engineering inspector.

Ground inspection.—Before conducting any flight tests, the CAA representative will complete the ground inspection to determine that all items affecting the safety in flight have been found satisfactory. Prior to presenting the aircraft for certification, it is suggested that the applicant use a form of the type used by the Administrator, and check the aircraft for the purpose of verifying that all items are satisfactory for presentation. Copies of this form may be obtained from a CAA engineering or factory inspector.

Flight tests.—The airplane will be subjected to such flight tests as are necessary to prove compliance with the flight and operation requirements specified in 04.7 and to supply the pertinent information required upon the form specified for flight test reports.

Discontinuance of type inspection.—If during any part of the ground inspection or flight test there is noted any unfavorable characteristic or defect which is considered sufficiently serious by the CAA representative to warrant discontinuing the type inspection until corrective measures have been taken by the applicant:

a. The CAA representative will note each unsatisfactory item upon a form supplied for the purpose, with sufficient detail so that it will be clear to all concerned.

b. One copy of such form will be transmitted to the manufacturer.

c. The manufacturer should advise the Administrator when the aircraft incorporating the required changes will be available for continuance of the type inspection.

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

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

The purpose of the aircraft specification is to describe the airplane and the conditions on which it is certificated as a type for the information of inspection personnel in certificating individual airplanes. The form and contents of these specifications are the responsibility of the Administrator. However, applicants may be given an opportunity to comment on preliminary drafts of the specifications and their comments will be given consideration when the final specifications are drafted.

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

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

04.06 CHANGES.

Changes to certificated aircraft shall be substantiated to demonstrate continued compliance of the aircraft with the pertinent airworthiness requirements.

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

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

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

04.060 Minor changes. Minor changes to airplanes being manufactured under the terms of a type certificate and which obviously do not impair the condition of the airplane for safe operation may be approved by authorized representatives of the Administrator prior to submittal to the Administrator of any required revised drawings. The approval of such minor changes shall be based on the airworthiness requirements in effect when the particular airplane model was originally certificated, unless, in the opinion of the Administrator, compliance with current airworthiness requirements is necessary.

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

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

a. Submit the usual file drawings.

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

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

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

04.061 Major changes. Major changes to airplanes being manufactured under the terms of a type certificate may require the issuance of a new type certificate and the Administrator may, in his discretion, require such changes to comply with current airworthiness requirements.

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

Installation of an engine of a type other than that covered by the original type (or approved type) certificate

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

2. The general procedure to be followed, when the rated power of the engine to be installed exceeds that originally used for design purposes or exceeds the rated power of the engine being

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

3. To expedite handling and to reduce the usual exchange of correspondence to a minimum, the applicant for approval of the change should always supply a complete description of the proposed engine replacement. When an individual airplane is being modified it should be identified in the correspondence as to name of manufacturer, model designation, manufacturer's serial number and identification mark. In addition, a new or revised airplane model designation should be selected to distinguish it from the original model. The current status of the engine to be used, with respect to CAR 13, should be determined prior to the completion of any extensive changes. CAA field inspection personnel are supplied with this information and they will assist in the determination of the status of the engine in question. Copies of the approved engine specification can be obtained from CAA Information and Statistics, Department of Commerce, Washington 25, D. C. If the details of the powerplant installation are affected, note that the pertinent requirements specified in paragraph 1 (e), page 3, and 04.6 call for certain approved file data.

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

5. Changes in engine mount structure and the local effects of an increase in engine weight must, of course, be investigated. The extent of such investigation will depend largely upon the amount of increased power the applicant desires to use in take-off (one minute) and the remaining operations. See 7 below for references to operation limitations. See "Drawings," p. 3 for the information required on drawings submitted covering the changes made.

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

a. Level Flight or Climb: V_L.

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

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

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

a. Maximum, except take-off horsepower, *not to exceed* the output of the originally approved engine installation which is being replaced.

b. Take-off (one minute) horsepower, limited by:

(1) Approved take-off rating of engine. See 04.60, CAR 13 and approved engine specification.

(2) Status of propeller used. See 04.61, CAR 14 and approved propeller specification.

(3) Strength of engine mount structure. See 04.26.

(4) Fuel flow capacity. See 04.625.

(5) Engine cooling requirements. See 04.640.

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

Maximum, except take-off horsepower, limited by:

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

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

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

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

a. Assume a power loading of 12 pounds per HP in determining the maneuvering load factors. (See figure 8.)

b. Design the engine mount, adjacent structure, and powerplant installation for the maximum power which might possibly be used in the future.

c. Assume a design level speed (V_L) considered high enough for all future operations. In this connection it should be noted that speed placards refer to "indicated" airspeeds and that the corresponding actual airspeed may therefore exceed the placard speed at altitudes above sea level.

Conversion of approved type landplane or seaplane to approved skiplane status

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

a. Approval of the ski model.

b. Approval of the airplane equipped with approved skis.

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

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

Approval of an airplane equipped with approved skis.—Certain airplane models are already approved with certain specific approved skis installed. The owner of a certificated airplane of some such model wishing to install skis, need only install skis of the model with which airplanes of his model are approved and his airplane will be approved with the skis installed, upon the satisfactory completion of an inspection of the installation by a CAA representative. Should changes in the landing gear be necessary to accommodate the skis, the owner, of course, must make the changes in accordance with the change data approved by the Administrator. If the airplane is of a model which has not been approved with the installation of skis of the particular approved model it is desired to install, the procedure hereinafter outlined should be followed:

a. Technical data showing any changes in the landing gear should be submitted to the Administrator for approval. This is not often necessary, as skis are usually designed to attach to the axles in place of the wheels.

b. Upon approval of the change data, if any, the installation must pass a satisfactory inspection by a representative of the Administrator.

c. During this inspection, the representative will obtain the weight of the ski installation and the weight of the wheel installation which has been replaced.

d. Upon completion of a satisfactory inspection, the representative will witness take-offs and landings, and other demonstrations if deemed necessary, of the airplane equipped with skis. The characteristics of the airplane equipped with skis must be acceptable to the Administrator's representative.

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

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

04.062 Changes required by the Administrator. In the case of aircraft models approved under the airworthiness requirements in effect prior to the currently effective regulations, the Administrator may require that aircraft submitted for original airworthiness certification comply with such portions of the currently effective regulations as are considered necessary.

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

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

04.1 DEFINITIONS

04.100 Weight, *W*. The total weight of the airplane and its contents.

04.101 Design weight. The weight of the airplane assumed for purposes of showing compliance with the structural requirements hereinafter specified.

04.1010 Minimum design weight. Weight empty with standard equipment, plus crew, plus fuel of 0.25 lb. per maximum (except take-off) horsepower, plus oil as per capacity.

04.102 Standard weight. The maximum weight for which the airplane is certificated as complying with all the airworthiness requirements for normal operations.

04.103 Provisional weight. The maximum weight for which the airplane is certificated as complying with the airworthiness requirements as modified for scheduled air carriers in §04.71.

04.104 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 nacelles and through the fuselage to the plane of symmetry.

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

Trailing edge cut-outs may in general be neglected if they do not remove more than one-half the chord.

The application of 04.104 to several typical cases is illustrated in figure 2.

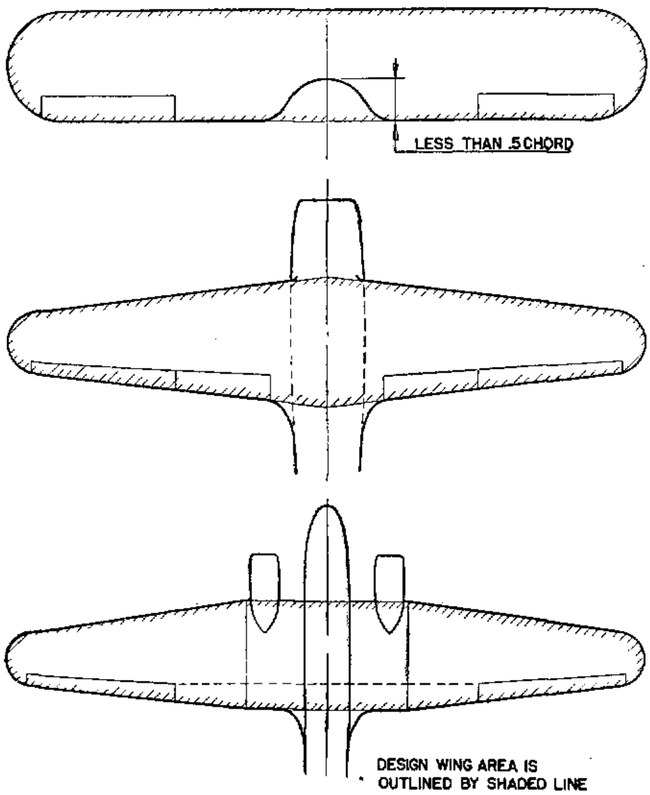


Figure 2.—Typical design wing areas.

04.105 Design Power, *P*. The total engine horsepower chosen for use in determining the manuevering load factors. The corresponding engine output will be incorporated in the aircraft certificate as a maximum operational limitation in all flight operations other than take-off or climbing flight. (See § 04.744.)

For airplanes of less than 2,000 pounds standard weight the design power should not be less than the maximum except take-off rating of the engine installed.

In the case of airplanes having standard weights of 2,000 pounds or more, there are no specific restrictions in regard to the choice of design power.

04.106 Design wing loading, *W/S*. The design weight (§ 04.101) divided by the design wing area (§ 04.104).

04.107 Design power loading, *W/P*. The design weight (§ 04.101) divided by the design power. (See § 04.105 and figure 8.)

04.108 Air density ρ. The mass density of the air through which the airplane is moving, in terms of the weight of a unit volume of air divided by the acceleration of gravity. The symbol ρ_0 denotes the mass density of air at sea level under standard atmospheric conditions and has the value of 0.002378 slugs per cubic foot. (See § 04.130 for definition of standard atmosphere.)

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

04.110 Indicated airspeed, *V*. The true airpseed multiplied by the term $\sqrt{\rho/\rho_0}$. (See § 04.108.)

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

04.111 Design level speed, V_L. The indicated airspeed chosen for use in determining the pertinent structural loading conditions. This value will be incorporated in the aircraft certificate as a maximum operational limitation in level and climbing flight. (See § 04.743.)

In the case of airplanes of less than 2,000 pounds standard weight the high speed in level flight at design power should not exceed the value chosen for V_L. Performance calculations based on actual flight test data will be acceptable to substantiate the value of V_L selected if it is found to be impractical to conduct tests at design power.

For airplanes having standard weights of 2,000 pounds or more, there are no specific restrictions in regard to the choice of design level speed.

04.112 Design gliding speed, V_g. The maximum indicated airspeed to be used in determining the pertinent structural loading conditions. (See §§ 04.211 and 04.743.)

The equation given in 04.211 for the minimum value of V_t provides for the following factors

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

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

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

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

04.114 Design flap speed, V_f. The indicated airspeed at which maximum operation of high-lift devices is assumed. (See §§ 04.211 and 04.743.)

04.115 Maximum vertical speed, V_m. A fictitious value of indicated airspeed computed for unaccelerated flight in a vertical dive with zero propeller thrust.

04.116 Design maneuvering speed, V_p. The indicated airspeed at which maximum operation of the control surfaces is assumed. (See § 04.211.)

The equation given in 04.211 for V_p is intended to provide for the following factors:

a. V_p cannot be less than the minimum speed of level flight.

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

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

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

04.117 Design gust velocity, U. A specific gust velocity assumed to act normal to the flight path. (See § 04.2121.)

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

$q = \frac{1}{2}\rho V_t^2$ (in terms of true airspeed).

$= \frac{1}{2}\rho_0 V^2$ (in terms of indicated airspeed).

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

(See § 04.108 for definition of ρ.)

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

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

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

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

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

04.124 Ultimate load. *A limit* load multiplied by the specified *ultimate* factor (or factors) of safety. (See above definitions and § 04.200.)

04.125 Yield load. *A limit* load multiplied by the specified *yield* factor (or factors) of safety. (See above definitions and § 04.201.)

04.126 Strength test. A static load test in which the *ultimate* loads are properly applied. (See § 04.200 and § 04.3021.)

04.127 Proof test. A static load test in which the *yield* loads are properly applied for a period of at least one minute. (See § 04.201.)

04.128 Balancing loads. Loads by which the airplane is placed in a state of equilibrium under the action of external forces resulting from specified loading conditions. The state of equilibrium thus obtained may be either real or fictitious. Balancing loads may represent air loads, inertia loads, or both. (See § 04.2210.)

04.129 Aerodynamic coefficients, C_L, C_M, CP, etc. The coefficients hereinafter specified are those of the "absolute" (nondimensional) system adopted as standard in the United States. The subscripts N and C used hereinafter refer respectively to directions normal to and parallel with the basic chord of the airfoil section. Other subscripts have the usual significance. When applied to an entire wing or surface, the coefficients represent average values and shall be properly correlated with local conditions (load distribution) as required in § 04.217.

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

04.131 Primary structure. Those portions of the airplane, the failure of which would seriously endanger the safety of the airplane.

This includes such items as control systems, fittings, auxiliary members used to support or strengthen other members carrying direct loads, covering of wings and control surfaces, etc., in addition to the main load carrying structure.

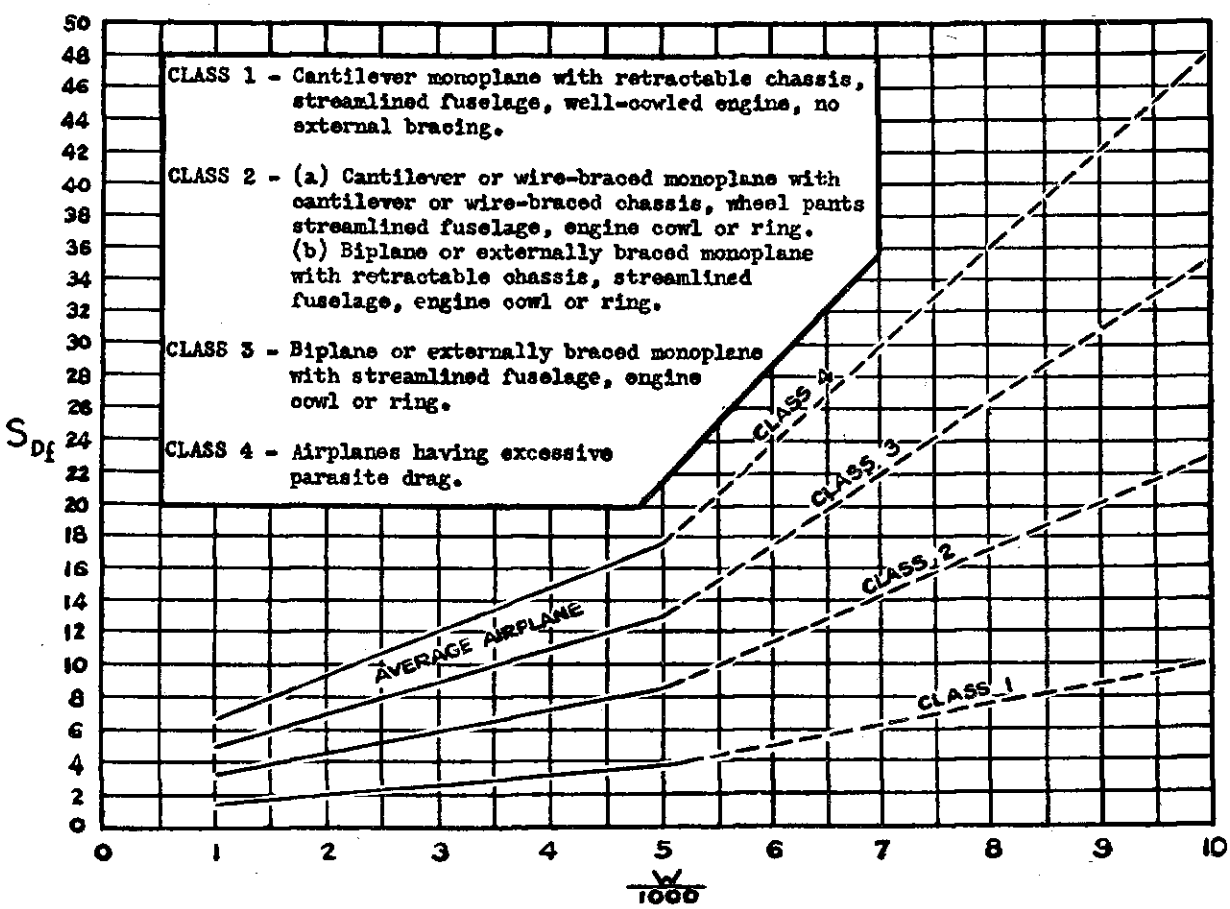


Figure 3.—Variation of fuselage drag area with gross weight.

Aerodynamic center, A. C.[1] The point on the wing chord, expressed as a fraction, about which the moment coefficient is substantially constant for all angles of attack. The theoretical location is at 25 percent of the chord. The actual location may differ from the theoretical location and may be determined from the slope of the moment coefficient curve as outlined under "Computation of Additional Characteristics," page 22.

Drag area.[1] The area of a hypothetical surface having an absolute drag coefficient of 1.0.

Equivalent Drag Area, S_D.[1] The drag area which, at a given value of dynamic pressure, will produce the same aerodynamic drag as the body or combination of bodies under consideration. (Note: $S_D = 1.28\ S_E$, where S_E is the equivalent flat plate area). S_{D_t}=estimated total drag area at high speed, in square feet. When the value of V_L is known or has been estimated, S_{D_t} can be determined by solving Eq. 16 for d. When it is desired to estimate S_{D_t} first in order to compute the value of V_L, the equation $S_{D_t} = S_{D_f} + C_d S_w$ can be used. S_w refers to the total wing area exclusive of the area replaced by the fuselage and C_D can usually be assumed to be the minimum wing drag coefficient. Typical values of S_{D_f} (Drag area of airplane less wing) are given in figure 3.

Margin of safety, M. S.[1] The margin of safety is the percentage or fraction by which ultimate strength of a member exceeds its ultimate load. A linear margin of safety is one which varies linearly with the ultimate load. A nonlinear margin of safety is one which is based on stresses which are not proportional to the ultimate load. A nonlinear margin of safety is not a true measure of the excess strength of a member.

STANDARD SYMBOLS

a—position of aerodynamic center, fraction of chord; subscript "actual."
$a.\ c.$—aerodynamic center.
b—distance between spars, fraction of chord; span of wing.
C—chord, feet; coefficient; constant; subscript, "chord."
CP—center of pressure, fraction of chord.
CG—center of gravity.
D—subscript "drag."
d—drag loading, lbs/sq. ft.
e—unit wing weight, lbs/sq. ft.
f—unit stress, lbs/sq. in; front spar location, fraction of chord; subscript, "fuselage".
g—acceleration of gravity (=32.2 ft/sec.^{2}); subscript "gliding".
HP—horsepower.
h—distance measured perpendicular to MAC, in terms of MAC.
i—subscript "induced".
j—position of wing CG, fraction of chord; factor of safety.
K—a general factor.
L—subscript "lift" or "level".
M—moment, ft./lbs; subscript "moment".
m—slope of lift curve, ΔC_L/radian; moment divided by W; subscript "maximum vertical".
MAC—mean aerodynamic chord.
MS—margin of safety.
N—subscript, "normal force."
n—load in terms of W (net value equals acceleration factor). Without subscript n refers to an applied load normal to the basic wing reference chord. With subscript "x", n refers to an applied load parallel to the basic wing reference chord. (See figure 17.)
o—subscript, "zero lift", "initial", "standard sea level".
P—design power; load, lbs.
p—power loading, lbs./HP.
q—dynamic pressure, lbs./sq. ft.
R—resultant force or reaction, lbs.; aspect ratio; subscript "resultant".
r—rear spar location, fraction of chord.
S—design wing area, sq. ft.
s—wing loading, lbs./sq. ft; subscript, "stall".
S_D—equivalent drag area, sq. ft.
S_E—equivalent flat plate area, sq. ft.
T—tail load, lbs.
t—subscript "tail".
U—gust velocity, ft./sec.
u—subscript "ultimate".
V—airplane speed, mi./hr.
v—airplane speed, ft./sec.
W—total weight of airplane and contents, lbs.
w—unit pressure, lbs. sq. ft.; subscript "wing".

[1] CAA definition.

$\bar{w}$—average unit pressure, lbs./sq. ft.
x—distance measured parallel to *MAC* in terms of *MAC;* subscript.
y—subscript, "yield".
α—(alpha)—angle of attack, radians or degrees.
β—(beta)—flight path angle, degrees.
Δ—(delta)—increment.
η—(eta)—propeller efficiency.
ρ—(rho)—mass density of air.

STANDARD VALUES AND FORMULAS

Air Density

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

Dynamic pressures:

2. $q = 1/2\ \rho_o V_i^2$
$= .00119\ v_i^2$ (where v_i is "indicated" speed, fps.)
$= .00256\ V_i^2$ (where V_i is "indicated" speed, mph.)

Basic airplane parameters:

3. $s = W/S$
4. $p = W/HP$
5. $d = W/S_D$ (The value of "d" should be the same as that used in, or determined from, Eq. 16.)

Aerodynamic coefficients:

6. $C_R = (C_L^2 + C_D^2)^{1/2}$
7. $C_N = C_L \cos\alpha + C_D \sin\alpha$
8. $C_C = -C_L \sin\alpha + C_D \cos\alpha$ (positive rearward)
9. $C_{M_x} = C_N(x - CP)$ (Where x is the distance, from the leading edge, of the point on the chord about which the moment is computed, expressed as a fraction of the chord).

Forces, unit loadings, and couples:

10. $F_x = C_x S q$ (Where x may be R, L, D, N, C, or M)
11. $F_D = S_D q$
12. $M = F_M C$ (torque or couple)
$= C_M S q C$
13. $\bar{w} = C_N q$
14. $n = F/W$
15. $F_{pr} = 375\eta HP_a/V_a$ (propeller thrust, pounds)

Speeds:

16. $V_{L_a} = 52.7(\eta d/p_a)^{1/3}\ (\rho_o/\rho_a)^{1/3}$ (mph) = actual air speed at air density ρ_a.
17. $V_s = 19.76(s/C_L \max)^{1/2}$ (mph) = indicated stalling speed.
18. $V_m = 19.76\ (d)^{1/2}$ (mph) = indicated theoretical maximum vertical speed.
19. $V_i = V_a(\rho_a/\rho_o)^{1/2}$ where V_i = indicated air speed.
V_a = actual air speed.
ρ_o = standard density of air at sea level.
ρ_a = density of air in which V_a is attained.
20. $\Delta C_L = m\ (U/v)$ = change in C_L due to gust.
21. $\Delta n = \Delta C_L\ (q/s)$ = change in load factor due to gust.

AERODYNAMIC COEFFICIENTS

A. General

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

2. The subscripts "L" and "D" refer to directions normal to and parallel to the relative wind, while the subscripts "N" and "C" refer to directions respectively normal to and parallel to the basic wing chord. Subscript "R" refers to the direction of the resultant force. These factors are illustrated in figure 4a and b. When the planes of the drag truss and lift trusses do not coincide respectively with the planes of the basic chord and the plane of the normal forces, a correction is necessary before the coefficients can be used directly in the wing analysis method outlined in "Proof of Wings," page 59. The corrected coefficients are obtained by resolving the resultant

force coefficients into components in the plane of the lift truss and drag truss, as shown in figure 4c. The effect on the chord coefficient may be considerable, but the correction for C_N will usually be negligible.

3. The moment coefficient may be considered to be of the same nature as the force coefficients if the force to which it corresponds is applied as a couple at the leading and trailing edges of the wing chord, as shown in figure 4. A positive moment coefficient requires an upward force at the leading edge, as shown. The conversion of center of pressure position into a moment coefficient about any given point can be easily accomplished by means of Eq. 9, page 19. It should be noted that the center of pressure and the moment coefficient are alternative in nature and can not both be used at the same time.

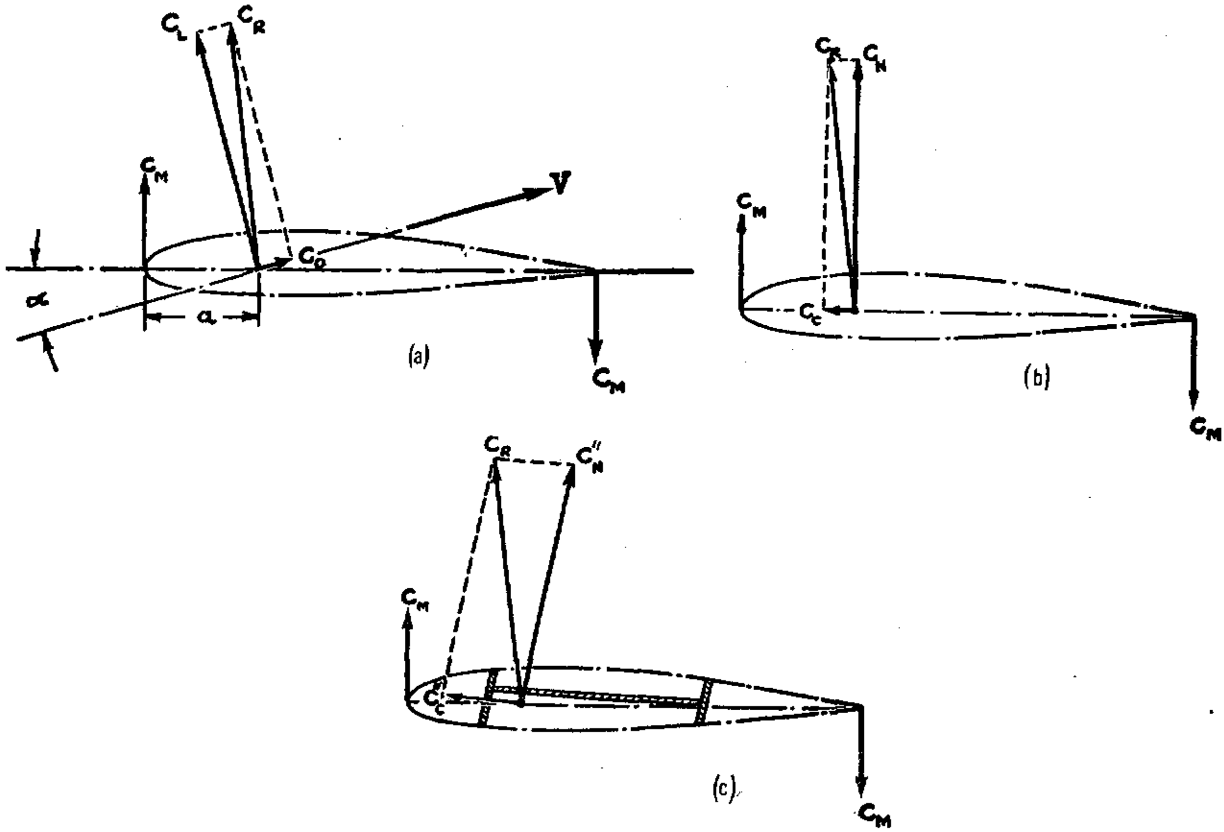


Figure 4.—Illustration of airfoil force coefficients.

B. Determination of Corrected Airfoil Characteristics

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

2. *Aspect ratio corrections.*—The methods of correcting for aspect ratio are well defined and are outlined in various text books and reports. The following equations may be used in this connection.

$R = (kb)^2/S$

Where R=aspect ratio,
k =Munk's span factor for biplanes (for monoplanes $k=1.0$),
b =span of longest wing, and
S=design wing area.

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

Where K=correction factor.

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

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

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

$C_D = C_{D_6} + 0.318\ KC_L^2$ (Items 5 to 7, Table I).

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

$C_D = C_D$ for given C_L when aspect ratio is R.

$$m = m_6\left[\frac{4}{3+6/R}\right]$$

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

m = slope of lift curve when aspect ratio is R.

C. Computation of Additional Characteristics

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

a. The normal force coefficient, C_N, can be determined from Eq. 7, page 19. The steps involved are shown as items 8 to 12 of Table I. It will be found that C_N is almost identically equal to C_L for small values of the latter. This may not be true, however, for unconventional or modified airfoils, such as those equipped with flaps.

b. The chord coefficient, C_C, is determined from Eq. 8, page 19. The steps are outlined as items 13 to 15 of Table I.

c. The moment coefficient about the aerodynamic center, C_{M_a} is not given in some airfoil reports. In some cases the moment coefficient about the quarter chord point $C_{M_{c/4}}$ is plotted

TABLE I. —Computation of Airfoil Characteristics

1	C_L	−1.0	−.8	−.6	−.4	−.2	0	.2	.4	.6	.8	1.0	1.2	1.4	1.6	1.8	2.0	2.2
2	α_6																	
3	$\Delta\alpha = 18.24 + KC_L$																	
4	α = ② + ③																	
5	C_{D_6}																	
6	$\Delta C_{D_i} = .318\ KC_L^2$																	
7	C_D = ⑤ + ⑥																	
	$C_{D_{ext.}}$ = ⑲ + ⑳																	
8	cos α = cos ④																	
9	sin α = sin ④																	
10	C_L cos α = ① × ⑧																	
11	C_D sin α ⑦ × ⑨																	
12	C_N = ⑩ + ⑪																	
13	C_L sin α = ① × ⑨																	
14	C_D cos α = ⑦ × ⑧																	
15	C_C = ⑭ − ⑬																	
16	$CP = CP_6$																	
	$CP_{ext.} = a - C_{M\alpha}$/⑫																	
17	$C_{M_{c/4}}$ = (.25 − ⑯) × ⑫																	
18	C_{M_a} = ⑰ + (a − .25) × ⑫																	
19	C_{D_i} = ⑥/KR																	
20	C_{D_o} = ⑦ − ⑲																	
21	ΔC_{L_u} (Biplane)																	
22	ΔC_{L_l} (Biplane)																	
23	C_{L_u} = ① + ㉑																	
24	C_{L_l} = ① + ㉒																	

R = $K = 1/R - 1/6$ = a = ΔC_{L_u} ΔC_{Ll} =

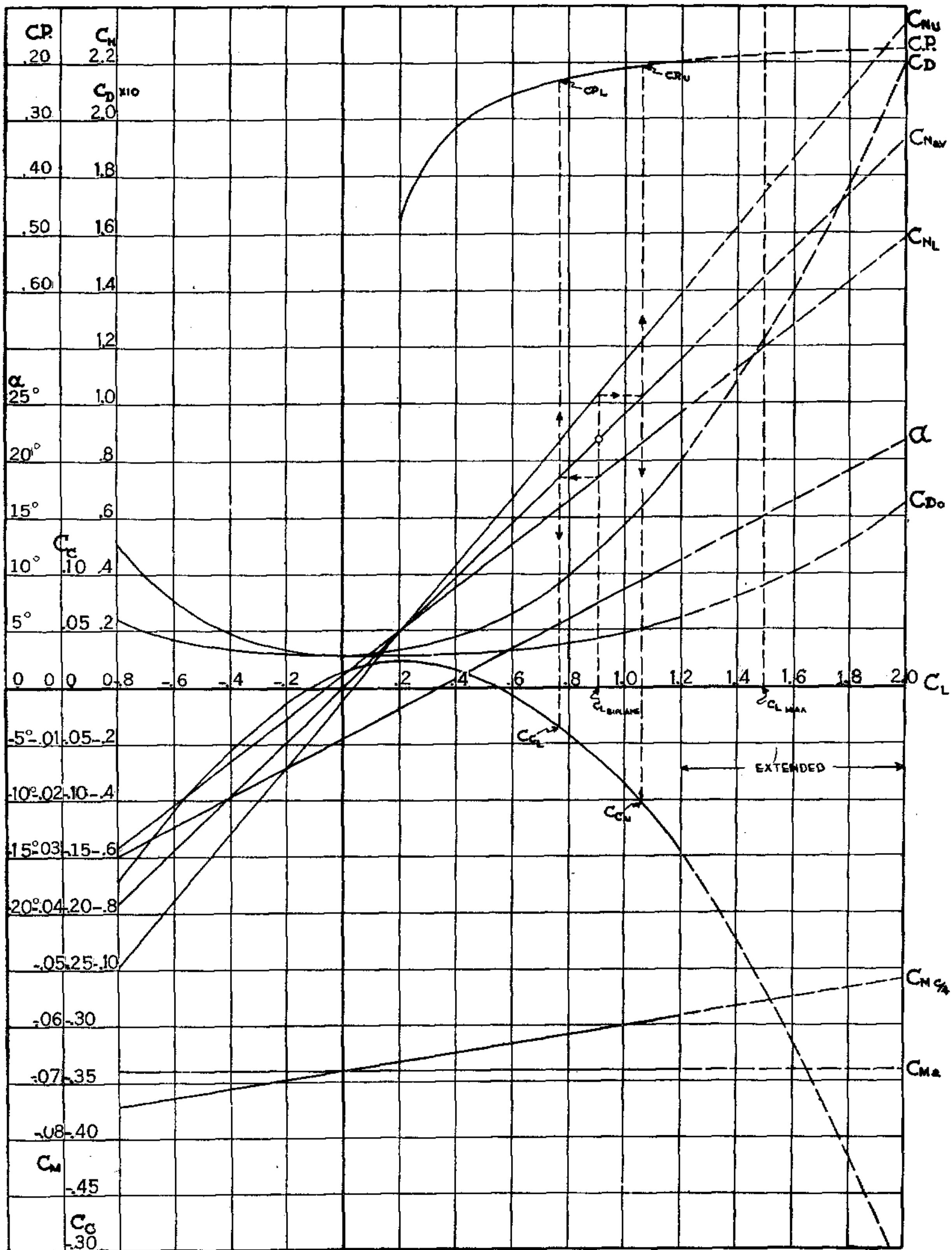


Figure 5.—Corrected airfoil characteristics.

against C_L. In such cases a straight line can be drawn to fit the $C_{M_{c/4}}$ curve as closely as possible (see figure 5). The average value of C_{M_a} is then obtained from the straight line where $C_L=O$. The position of the aerodynamic center can then be obtained by the following equation:

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

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

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

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

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

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

D. Extension of Characteristic Curves

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

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

b. Determine the induced drag coefficient as outlined in item 19 of Table I. R and K are defined on page 20.

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

d. Extend the C_D curve by determining the values for item 7 of Table I, as indicated.

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

f. The extended values of C_N and C_C are determined as indicated under items 8 to 15 of Table I, using the extended values of C_D.

g. The CP values should be extended by means of the equation:

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

using the extended values of C_N.

E. Biplane Effects

1. The effects of biplane interference can be conveniently accounted for by a suitable modification of the corrected airfoil characteristic curves illustrated in figure 5. The modification of the various characteristics for each wing can be carried out as follows, referring to Table I:

a. Lift coefficients.—The individual lift coefficients for each wing should be determined for the useful range of average lift coefficient, C_L, (item 1 of Table I). Appendix III, comprises the acceptable method and calls attention to the limitation in the application of NACA Report No. 501. This method derives increments which are added to and subtracted from the average lift coefficient. Items 21 and 24 are provided in Table I for this purpose.

b. Normal force coefficients.—The corrected normal force coefficients for each biplane wing are plotted on figure 5. These values can be determined from the original curve of average normal force coefficient by using the corrected values of C_L under items 23 and 24, Table I.

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

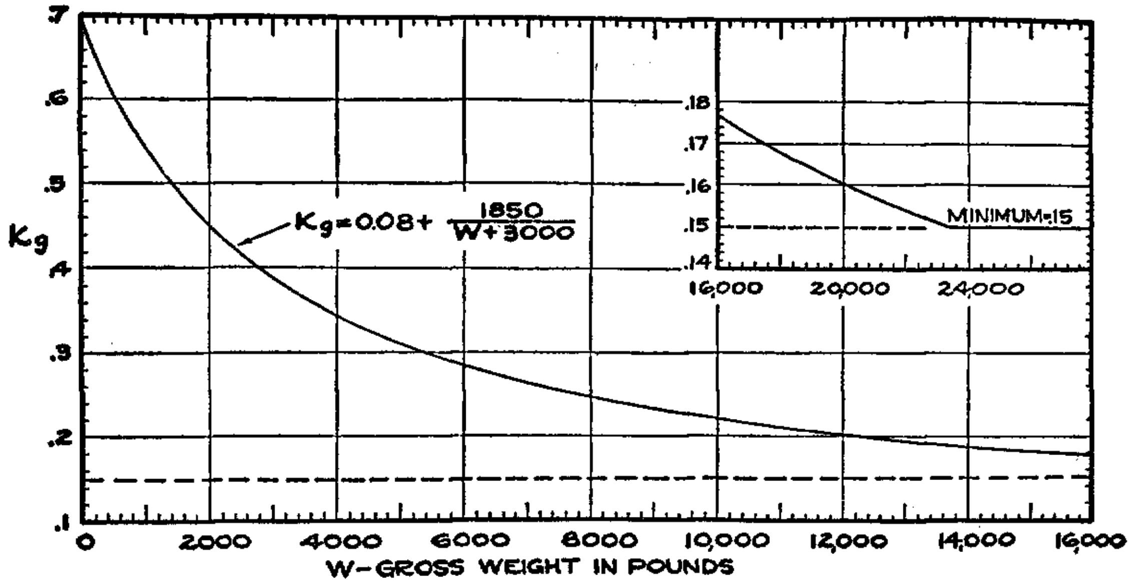


Figure 6.—Gliding speed factor.

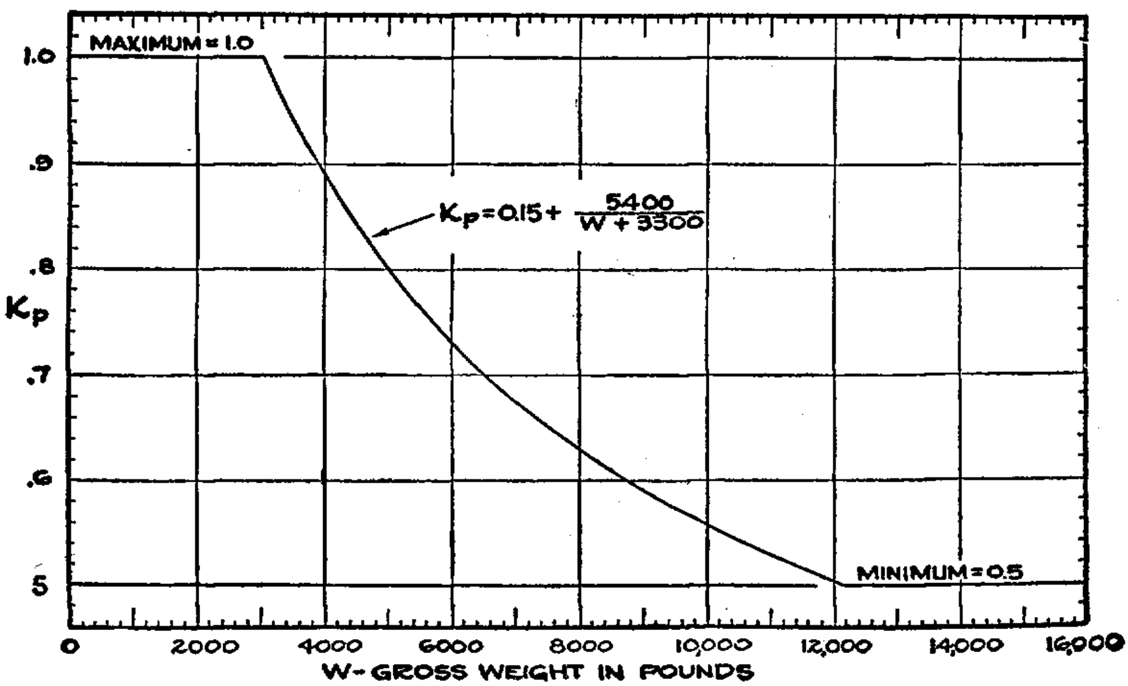


Figure 7.—Pull-up speed factor.

04.2 STRUCTURAL LOADING CONDITIONS

04.20 GENERAL STRUCTURAL REQUIREMENTS.

04.200 Strength. The primary structure (see § 04.131) shall be capable of supporting the *ultimate* loads (see § 04.124) determined by the loading conditions and *ultimate* factors of safety hereinafter specified, the loads being properly distributed and applied.

04.201 Deformations. The primary structure shall be capable of supporting without detrimental permanent deformations, for a period of at least one minute, the *yield* loads (see § 04.125) determined by the loading conditions and *yield* factors of safety hereinafter specified, the loads being properly distributed and applied. Where no *yield* factor of safety is specified a factor of 1.0 shall be assumed. In addition, temporary deformations which occur before the yield load is reached shall be of such a nature that their repeated occurrence will not weaken or damage the primary structure.

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

In determining the permanent deformations the effects of slippage or jig deflection may be deducted if properly measured.

04.202 Stiffness. The primary structure shall be capable of supporting the *limit* loads (see § 04.120) determined by the loading conditions hereinafter specified without deflecting beyond whatever limits may be hereinafter prescribed or which may be deemed necessary by the Administrator for the case in question.

04.203 Proof of strength and rigidity. No general requirements, but see § 04.3 for specific requirements.

04.204 Materials, fabrication, protection, etc. No general requirements, but see § 04.4 for specific requirements.

04.21 FLIGHT LOADS.

04.210 General. The airworthiness rating of an airplane with respect to its strength under flight loads will be based on the airspeeds and accelerations (from maneuvering or gusts) which can safely be developed in combination. For certain classes of airplanes the acceleration factors and gust velocities are arbitrarily specified hereinafter and shall be used for those classes. The airspeeds which can safely be developed in combination with the specified acceleration factors and gusts shall be determined in accordance with the procedure hereinafter specified and shall serve as a basis for restricting the operation of the airplane in flight. (See § 04.743.)

04.211 Airspeeds. (See §§ 04.109 to 04.116 for definitions.) The design airspeeds shall be determined as ollows:

V_L (See § 04.111)

V_g shall not be less than $V_L + K_g(V_m - V_L)$, except that it need not be greater than either $V_L + 100$ miles per hour or 1.5 V_L, whichever is lower. K_g is specified on figure 6. V_m is defined in § 04.115. A special ruling may be obtained from the Administrator if the design gliding speed thus determined is greater than 1.33 V_L and appears to be unnecessarily high for the type of airplane involved.

V_f shall not be less than $2V_{sf}$. V_{sf} is defined in § 04.113.

V_p shall not be less than $V_{sf} + K_p(V_L - V_{sf})$, except that it need not be greater than V_L. K_p is specified in figure 7. (See §§ 04.2220, 04.2223 and 04.2230 for exceptions for multi-engine airplanes.)

04.212 Load factors. The flight load factors specified hereinafter shall represent *wing* load factors. The *net* load factor, or acceleration factor, shall be obtained by proper consideration of balancing loads acting on the airplane in the specific flight conditions.

04.2120 Maneuvering load factors. The limit maneuvering load factors specified hereinafter (see figure 8) are derived largely from experience with conventional types of airplanes and shall be considered as minimum values unl ess it can be proved, to the satisfaction of the Administrator, that the airplane embodies features of design which make it impossible to develop such values in flight, in which case lower values may be used subject to the approval ofthe Administrator.

04.2121 Gust load factor. The gust load factors shall be computed on the basis of a gust of the magnitude specified, acting normal to the flight path, and proper allowance shall be made for the effects of aspect ratio on the slope of the lift curve. The gust velocities specified shall be used only in conjunction with the CAA gust formulae below.

The following formula for the load factor added in encountering a gust should be used for wings:

$\Delta n = \dfrac{KUVm}{575\ (W/S)}$ where Δn=limit load factor increment.

K=gust factor, see figure 9.

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

V=indicated airspeed, miles per hour.

W/S=wing loading (04.106).

m=slope of lift curve, C_L per radian, corrected for aspect ratio.

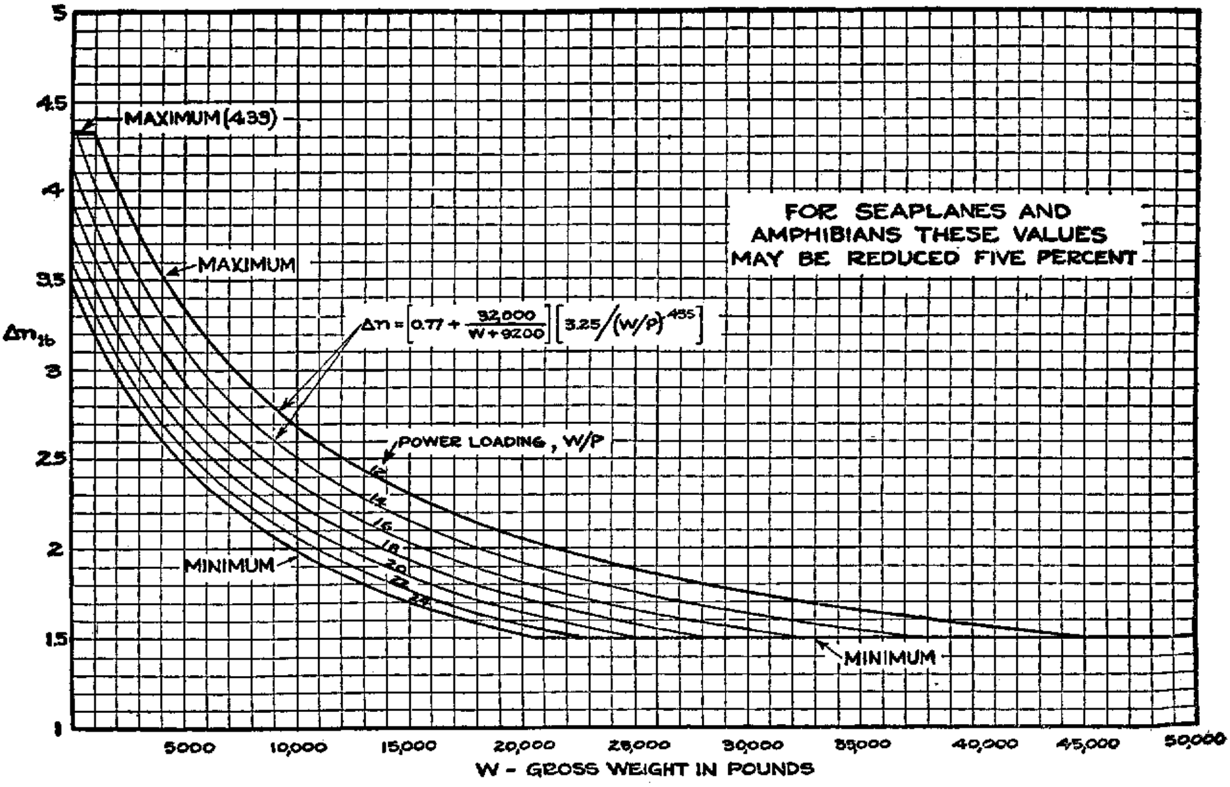


Figure 8.—Maneuvering load factor increment, Conditions *I* and *III*.

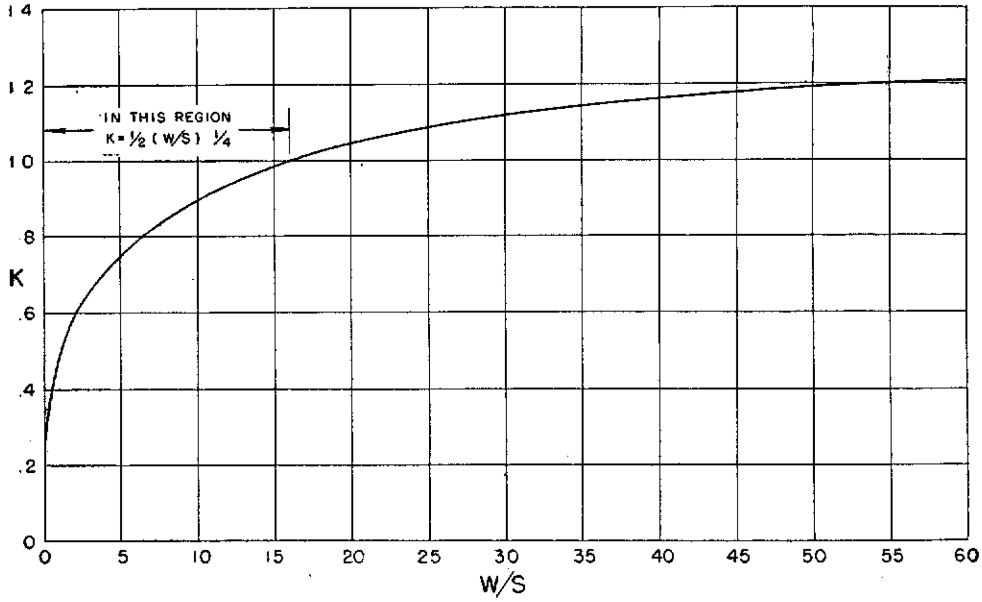


Figure 9.—Variation of gust factor with wing loading.

04.2122 Factors of safety. The minimum factors of safety are specified for each loading condition. See also § 04.27 for multiplying factors of safety required in certain cases.

04.213 Symmetrical flight conditions (flaps retracted).

04.2130 General. The following flight conditions, together with Table II, shall be considered as representing the minimum number of conditions required to cover a suitable range of symmetrical flight loadings.

TABLE II.—Symmetrical Flight Conditions (flaps retracted)

	Condition	I (p. 27)	II (p. 28)	III (p. 28)	IV (p. 29)	V (p. 30)	VI (p. 30)
1	Design speed (see *Airspeeds*, p. 25)	V_L	V_L	V_g	V_g	V_L	V_g
2	Gust velocity, U, *fps* [1] [2]	+30	−30	+15	−15	------	------
3	Δn (*a*) Gust [3]	P. 25	P. 25	P. 25	P. 25	$-0.5\Delta n\mathrm{I}a$	------
	Δn (*b*) Maneuvering	Fig. 8	------	Fig. 8	------	$-0.25\Delta n\mathrm{I}b$	------
4	*Limit* load factor, n, When (*b*), above, gives two values of Δn, use larger	$1+\Delta n_{\mathrm{I}}$	$1+\Delta n_{\mathrm{II}}$	$1+\Delta n_{\mathrm{III}}$	$1+\Delta n_{\mathrm{IV}}$	$-1+\Delta n_{\mathrm{V}}$	------
5	Minimum value of n	2.50	None	2.50	None	−1.5	None
6	Minimum yield factor of safety, j_y	1.0	1.0	1.0	1.0	1.0	1.0
7	Minimum ultimate factor of safety, j^u	1.5	1.5	1.5	1.5	1.5	1.5

[1] Feet per second.
[2] + means upward, − means downward.
[3] May be limited by maximum dynamic lift coefficient obtainable under sudden changes of angle of attack.

04.2131 Condition *I* positive high angle of attack. The factors given in Table II and figure 8 for this condition shall be used. To provide for flight conditions critical for the front lift truss or its equivalent the aerodynamic characteristics C_N, CP (or C_M), and C_c shall be determined as follows:

(*a*) $C_{N_I}=\dfrac{n_I(W/S)}{q_L}$ (q_L is dynamic pressure corresponding to V_L; see §§ 04.111 and 04.118.)

(*b*) C_c' = value corresponding to C_{N_I}, or value equal to $-.20\ C_{N_I}$, whichever is greater negatively.

(*c*) CP' = most forward position of the center of pressure between $C_L=C_{N_I}$ and $C_{L\,max.}$; when C_{N_I} exceeds $C_{L\,max.}$, the CP curve shall be extended accordingly.

(*d*) For biplane combinations the CP of the upper wing shall be assumed to be 2.5 percent of the chord forward of its nominal position.

(*e*) C_M' = moment coefficient necessary to give the required CP' in conjunction with C_{N_I}.

This condition is illustrated in figure 10. It is primarily designed to represent conditions at which the highest positive acceleration or load factor is likely to be obtained and is based on either a gust or maneuvering condition. The maneuvering load factor increments given in figure 8 are semi-empirical and are based largely on past experience. They represent the highest increments of acceleration which are to be expected during maneuvers.

As it is possible to develop the limit load factor for Condition *I* in various flight attitudes, a definite range of values of C_L is included, as indicated in figure 10. This corresponds to the assumption that the limit load factor will be developed at speeds somewhat below the V_L, the lowest speed being that associated with the value of $C_{L\,\max}$. The modified flight conditions, which are explained in succeeding sections, are intended to provide for the effects of this assumption and are so specified as to require a minimum amount of investigation.

It will be noted that in Condition *I* a value for the CP is specified, instead of the moment coefficient. If it is desired to find the moment coefficient to be used in Condition *I*, the values of CP and C_{N_I} can be inserted in equation 9, page 19. In the case of a biplane, the proper correction should first be made to the upper wing CP.

The arbitrary assumption of $C_C=-.20\ C_N$ is based on an average figure for C_C at $C_{L\max}$ and an adjustment of the design speed to give the applied load factor required. If the gust condition causes the value of C_L to exceed $C_{L\,\max}$, the chord coefficient will usually be greater negatively than the arbitrary value specified.

04.21310 Condition I_1 (positive high angle of attack modified). The smaller of the two values of C_c specified in § 04.2131 (*b*) and the most rearward CP position in the range specified in § 04.2131 (*c*) shall also be investigated when Condition *I* is critical for the rear spar (or its equivalent) or if any portion of the front spar (or its equivalent) is likely to be critical in tension. Only the wings and wing bracing need be investigated for this condition.

In Condition *I*, the value of C_N required to produce the specified limit load factor at the high speed of the airplane will usually be considerably less than that corresponding to $C_{L\,\max}$. Condition *I* is designed to be critical for the front spar in bending and compression. For this reason arbitrary values of C_C and CP are assigned, which ordinarily represent a pull-up to the limit load factor at a speed lower than V_L. In certain cases, however, the actual accelerated condition at V_L may be critical for some portions of the structure, in which case it should be checked. The characteristics used for Condition I_1 are illustrated on figure 11. This condition applies in any event to the following cases:

Front spar.—When the tension flange or chord member of a front spar is designed for low

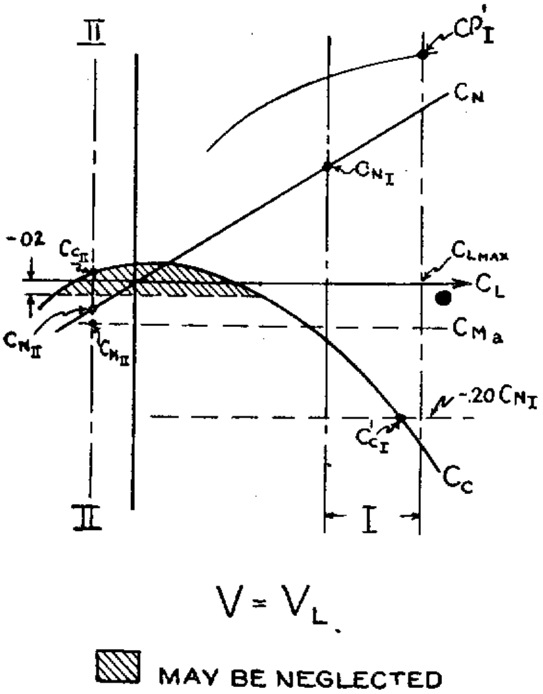


Figure 10.—Conditions *I* and *II*.

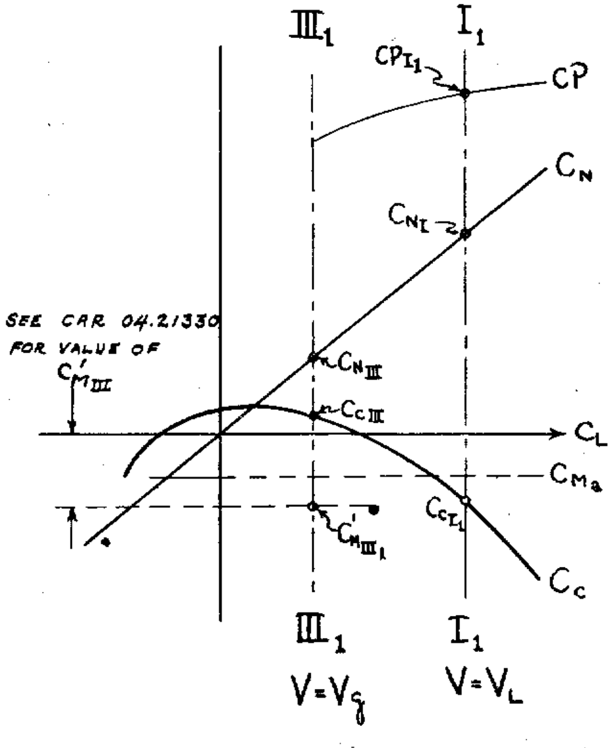


Figure 11.—Conditions I_1 and III_1.

margins of safety in Condition *I*, the smaller forward chord component which occurs in Condition I_1 may permit the net tension load to become greater than that computed for Condition *I* and thereby result in negative margins of safety.

Rear spar and rear lift truss.—When a wing section having a small negative or a positive moment coefficient is employed, it is possible for the rear spar to receive its greatest beam loading when the limit load factor for Condition *I* is developed at the speed V_L.

04.2132 Condition *II* (negative high angle of attack). The factors given in Table II for this condition shall be used with the following provisions:

(*a*) $C_{N_{II}}=\frac{n_{II}(W/S)}{q_L}$.

(*b*) C_c=actual value corresponding to $C_{N_{II}}$.

(*c*) When C_c is positive or has a negative value smaller than 0.02 it may be assumed to be zero.

(*d*) C_M=actual value corresponding to $C_{N_{II}}$.

This condition represents the effects of encountering a downward gust of 30 feet per second while flying at the speed V_L. The coefficients to be used are illustrated in figure 10. The assumption of a zero chord coefficient in certain cases is not a requirement, but is permitted in order to simplify the analysis.

04.2133 Condition *III* (positive low angle of attack). The factors given in Table II for this condition shall be used, with the following provisions:

(*a*) $C_{N_{III}}=\frac{n_{III}(W/S)}{q_g}$ (q_g is dynamic pressure corresponding to V_g; see §§ 04.118 and 04.112).

(*b*) C_c=actual value corresponding to $C_{N_{III}}$.

(*c*) When C_c is positive or has a negative value smaller than 0.02 it may be assumed to be zero.

(*d*) C_M=actual value corresponding to $C_{N_{III}}$.

This condition represents an upward acceleration of the airplane at its design gliding speed V_g. The coefficients to be used are shown graphically in figure 12. As in Condition *I*, the applied load factor is considered to be produced by either a gust or a maneuver. As the speed V_g is the speed at which the airplane will be flown least, and not at all in very turbulent air, the gust load factor formula is based on a gust of 15 feet per second and the arbitrary value of the limit acceleration required is less than that for Condition *I*. This is further justified by the fact that for a conventional 2-spar wing, the value of the limit load factor affects the rear spar load much less than the values of speed and moment coefficient used and is therefore relatively unimportant. For other types of wings, the values of speed and moment coefficient are again usually the more important with respect to torsional loading, the maximum beam loading being obtained from Condition *I*.

04.21330 Condition III_1 (positive low angle of attack, modified). If the moment coefficient of the airfoil section at zero lift has a positive value or a negative value smaller than 0.06 the effects of displaced ailerons on the moment coefficient shall be accounted for in Condition *III* for that portion of the span incorporating ailerons. To cover this point it will be satisfactory to combine 75 percent of the loads acting in Condition *III* with the loads due to a moment coefficient of $-0.08-C_{M_{III}}$ acting over that portion only of the span incorporating ailerons. The design dynamic pressure for the additional moment forces shall be equal to $0.75q_g$. Only the wings and wing bracing need be investigated for this condition.

This condition is included to provide for the use of ailerons during a pull-up or gust. It should be noted that a relatively small downward aileron deflection is sufficient to change the moment coefficient from a very small negative or from a positive value to the value specified. The effect of the displaced ailerons on high-moment airfoils is proportionately small and for that reason no corrections are required for such airfoils. In general the condition is critical only for the rear spar and the rear lift truss. This requirement is not applied to Condition *IV*, as the down-load on the front spar is not as sensitive to changes in aileron position.

04.2134 Condition *IV* (negative low angle of attack). The factors given in Table II for this condition shall be used, with the following provisions:

(*a*) $C_{N_{IV}}=\dfrac{n_{IV}(W/S)}{q_g}$.
(*b*) C_c=actual value corresponding to C_{NIV}.
(*c*) When C_c is positive or has a negative value smaller than 0.02 it may be assumed to be zero.
(*d*) C_M=actual value corresponding to C_{NIV}.

This condition, which is illustrated in figure 12, represents the effects of encountering a "down" gust of 15 feet per second while flying at the design gliding speed, V_g.

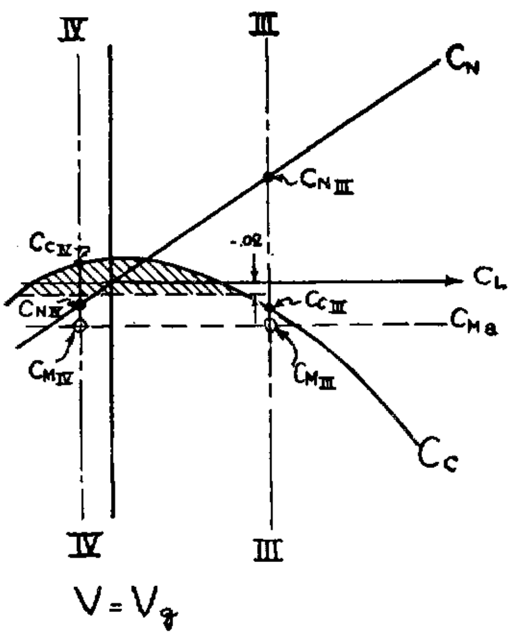


Figure 12.—Conditions *III* and *IV*.

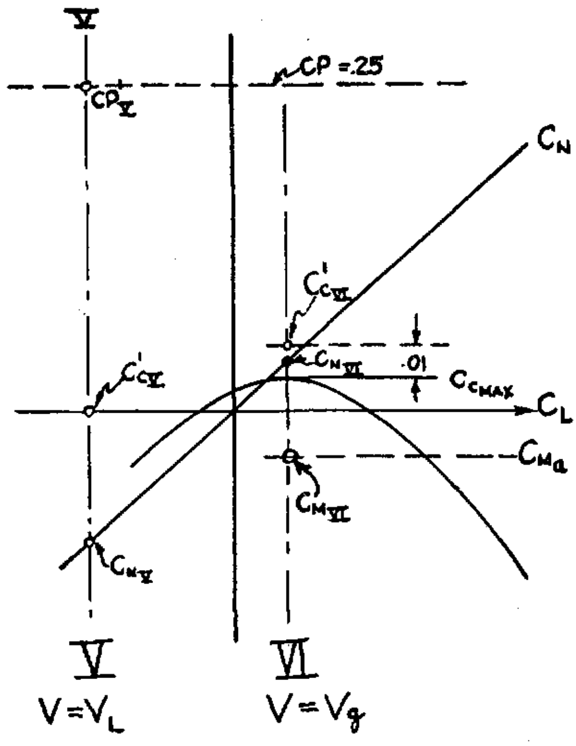


Figure 13.—Conditions *V* and *VI*.

04.2135 Condition *V* (inverted flight). The factors given in Table II for this condition shall be used, with the following provisions:

(a) $C_{NV}=\frac{n_V(W/S)}{q_L}$.
(b) $C_c'=0$.
(c) $CP'=25\%$.
(d) Only the rear (or single) lift truss system of externally braced wing structures need be investigated for this condition.

An airfoil which has a negative moment coefficient always tends to produce an up-load on the rear spar. It will usually be found, therefore, that none of the basic flying conditions produce any considerable down load on the rear spar (or any considerable "stalling" moment about the elastic axis of a wing). At large negative angles of attack, however, the moment coefficient about the aerodynamic center approaches zero and may even reverse in sign. This means that the *CP* approaches or lies behind the aerodynamic center. Condition *V* therefore represents such a condition, which is likely to be developed only in inverted flight. The applied load factors represent either a gust load factor, which may be produced while flying inverted, or a pull-up load factor based on the corresponding value for Condition *I*. For simplicity of the value of C_c is assumed to be zero. See figure 13 for illustration of this condition.

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

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

04.2136 Condition *VI* (gliding). The factors given in Table II shall be used for this condition with the following provisions:

(a) C_{NVI} = value corresponding to C_c max. (positive).
(b) $C_c'=C_c$ max. (positive) + 0.01.
(c) C_M = actual value corresponding to $C_{N_{VI}}$
(d) The drag of nacelles and other items attached to the wings shall be conservatively estimated and properly included in the investigation of this condition.
(e) Only the wings and wing bracing need be investigated for this condition.

This condition which is illustrated graphically in figure 13, is equivalent to the assumption that while flying at a speed V_g a small negative gust changes the value of C_C to $C_{C_{max}}$. The increment of 0.01 is added to account for surface roughness and protuberances.

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

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

04.2140 General. When flaps or other auxiliary high-lift devices are installed on the wings the design conditions shall be suitably modified to account for their use in flight. The modifications shall be based on the intended use of such devices and the aerodynamic characteristics of the wing. The following conditions, together with Table III, shall be considered as representing the minimum number of conditions required to cover a suitable range of symmetrical flight loadings in cases where the flaps are used only at relatively low airspeeds.

TABLE III.—Symmetrical Flight Conditions (flaps extended)

	Condition	VII (p. 31)	VIII (p. 31)	IX (p. 31)
1	Design speed (see *Airspeeds*, page 25)	V_f	V_f	V_f
2	Gust velocity, *U*, fps [1] [2]	+15	−15	
3	Δn [3]	P. 25	P. 25	
4	*Limit* load factor, *n*	$1+\Delta n_{VII}$	$1+\Delta n_{VIII}$	
5	Minimum value of *n*	2.00	None	None
6	Minimum yield factor of safety, j_y	1.0	1.0	1.0
7	Minimum ultimate factor of safety, j_u	1.5	1.5	1.5

[1] Feet per second.
[2] + means upward, − means downward.
[3] May be limited by maximum dynamic lift coefficient obtainable under sudden changes of angle of attack.

For internally braced monoplane wings equipped with trailing edge flaps, no stress analysis of the wing structure as a whole need be submitted for the flaps Conditions *VII* and *VIII*, provided that the average value of C'_M used in design Conditions *III* and *IV* equals or exceeds the quantity

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

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 weighting).

V_f is the design speed with flaps extended, as specified in 04.211.

V_s is the design speed used in Conditions *III* and *IV*, as specified in 04.211.

When the above condition is substantiated, no balancing computations for the extended flap conditions need be submitted and 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 from conditions *VII* and *VIII* below. The strength of special wing ribs used with split flaps, and the effect of flap control forces, should also be investigated. Reference should be made to current NACA reports and notes for acceptable flap data.

04.2141 Condition *VII* (positive gust, flaps deflected). The factors given in Table III for this condition shall be used, with the following provisions: (*a*) The most critical deflection of the flap shall be investigated. (*b*) The magnitude and distribution of normal, chord and moment forces over the wing shall correspond to that which would be obtained in developing the specified *limit* gust load factor at the specified airspeed.

04.2142 Condition *VIII* (negative gust, flaps deflected). The factors given in Table III for this condition shall be used, with the following provisions: (*a*) The most critical deflection of the flap shall be investigated. (*b*) The magnitude and distribution of normal, chord and moment forces over the wing shall correspond to that which would be obtained in encountering the specified *limit* gust load factor at the specified airspeed.

04.2143 Condition *IX* (dive, flaps deflected). The factors given in Table III for this condition shall be used, with the following provisions: (*a*) The most critical deflection of the flap shall be investigated. (*b*) The load factor and the magnitude and distribution of normal, chord and moment forces over the wing shall correspond to the angle of attack at which the greatest rearward chord loads are produced on the wing structure. (*c*) Only the wings and wing bracing need be investigated for this condition.

04.215 Unsymmetrical flight conditions.

04.2150 General. In the following unsymmetrical flight conditions, the unbalanced rolling moment shall be assumed to be resisted by the angular inertia of the complete airplane.

As an alternative procedure Conditions I_u, III_u, and V_u can be analyzed by modifying Conditions *I*, *III* and *V* respectively so that 100 percent of the airload is assumed to act on one wing and 70 percent on the other, provided that the angular inertia of the wings is neglected. For airplanes over 10,000 pounds standard weight the latter factor may be increased linearly with standard weight up to 80 percent at 25,000 pounds. The effects of wing nacelles and landing gear may, however, be considered in computing the angular inertia.

When the procedure above is followed, the approximate method of applying adjustments directly to the wing reactions may be used if desired. This method obviates the necessity for an additional determination of the beam load.

The use of more rational loading conditions than those specified here will be permitted if they are shown to be applicable. Such loading conditions should be based on studies giving consideration to unsymmetrical entries into gusts, to gusts affecting one wing only, and to maneuvering with ailerons.

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

04.2151 Condition I_u. Condition *I* (§ 04.2131) shall be modified by assuming 100 percent of the air load acting on one wing and 40 percent on the other. For airplanes over 1,000 pounds standard weight the latter factor may be increased linearly with standard weight up to 80 percent at 25,000 pounds.

04.2152 Condition III_u. Condition *III* (§ 04.2133) shall be modified as described for Condition I_u above.

04.2153 Condition V_u. Condition *V* (§ 04.2135) shall be modified as described for Condition I_u above.

04.216 Special flight conditions.

04.2160 Gust at reduced weight. The requirements for gust conditions (excepting tail surface gust conditions) under any loading between minimum and maximum design weight shall be met by primary structure critically loaded thereby.

It should be noted that a decrease in airplane gross weight will increase the gust load factor. This may cause critical loads to be developed in parts of the structure supporting dead weight. This should be thoroughly investigated in the case of airplanes having a widely variable loading. In other cases it can usually be demonstrated that the gust at reduced weight condition is critical only for the forward portion of the fuselage, the engine mount, and the attachments of items of dead weight.

When engine nacelles or other large items of dead weight are attached to the wing structure, they should be checked for the load factor due to the combined linear and angular accelerations resulting from the loadings of Condition I_u determined at the minimum design weight.

04.2161 Lift-wire-cut. For wings employing wire bracing in the lift truss, Conditions *I* and *III* shall be investigated, using load factors nI and $nIII$ of one half the values specified for these conditions and assuming that any lift wire is out of action. This requirement does not apply to parallel double lift wires, for which case see § 04.273.

04.2162 Drag-wire-cut. Drag struts in double-truss systems shall be designed to withstand the loads developed when the drag wire of the upper system in one bay and the drag wire of the lower system in the adjacent bay are each carrying their *limit* loads from any flight condition, the remaining wires in these two bays being assumed to be out of action. The minimum *ultimate* factor of safety shall be 1.5.

04.2163 Unsymmetrical propeller thrust. The structure shall incorporate an *ultimate* factor of safety of 1.5 against failure due to loads caused by maximum (except take-off) power applied on one side of the plane of symmetry only, when power on the other side is off and the airplane is in unaccelerated rectilinear flight.

04.2164 Wing tanks empty. If fuel tanks are supported by the wing structure, such structure and its bracing shall also be investigated for Conditions *I*, *II*, *III*, and *IV* with wing tanks empty. The design weight may be reduced by 0.9 pounds per certified maximum (except take-off) horsepower.

The specified weight reduction has particular application to cases in which the maximum authorized weight is based on full pay load and a fuel load of 0.15 gallons (0.9 pounds) per certified maximum (except take-off horsepower) in accordance with 04.740. In all other cases the reduction in weight may equal the weight of fuel that can be carried simultaneously with full pay load.

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

A. Span Distribution

1. For wings having mean taper ratios (see *c* below) equal to or greater than 0.5, the span distribution should be determined as follows:

a. If the wing does not have aerodynamic twist (i. e., if the zero lift lines of all sections are parallel), the span distribution of normal force coefficient (C_N) should be assumed to vary in accordance with figure 14a and b, which 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 figure 14a.

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

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

d. Acceptable methods of determining a rational span distribution are given in Army-Navy-Commerce publication ANC 1 (1), "Spanwise Air Load Distribution" (obtainable from the Superintendent of Documents, Washington, D. C., for 60 cents), in NACA Technical Report No. 572, in NACA Technical Report No. 585, in NACA Technical Note No. 606, and in Appendix IV.

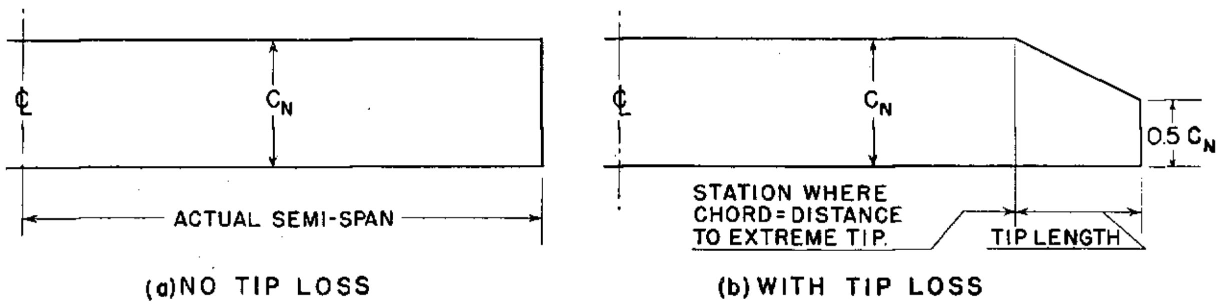
2. For all wings having mean taper ratios less than 0.5, the span distribution should be determined by rational methods, unless it is shown that a more severe distribution has been used.

3. The effects of nacelles on the normal force coefficient may, in general, be neglected. Their effects on chord loads are outlined in "C."

4. The effect of trailing edge cut-outs which remove less than 50 percent of the chord may be neglected when figure 14 is used.

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

6. When figure 14 is used, the chord coefficient should be assumed to be constant along the span, that is, it should be assumed that tip loss does not affect the chord coefficient.



NOTE: ABOVE FIGURES APPLY ONLY TO WINGS WITHOUT AERODYNAMIC TWIST BUT HAVING MEAN TAPER RATIOS GREATER THAN 0.5

Figure 14.—Span distribution of C_n for wings.

B. Chord Distribution

1. The approximate method of chord loadings outlined on pages 69 to 71 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. The methods outlined in ANC-1(2) "Chordwise Air Load Distribution of Airfoils" (obtainable from the Superintendent of Documents, Government Printing Office, Washington 25, D. C.) are acceptable for this purpose. It should be understood that when a rational pressure distribution is used for high angle of attack (Condition *I*), it should be determined so as to correspond to $C_{L\ max.}$ The value of dynamic pressure, q, to be used in conjunction with $C_{L\ max.}$ for the purpose of determining the load applied on the rib should be determined as follows:

$$q=\frac{(C_{n_1})(q_1)}{C_{L\ max.}}$$

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

3. *Effects of auxiliary devices.*—When a design speed higher than required is used in connection with wing flaps or other auxiliary high-lift 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 and fixed slots.

C. Special Loadings

1. *Parasite drag.*—The drag of large items attached to the wing cellule (such as nacelles) should be estimated and considered in conjunction with the conditions in which the addition of such a drag load may result in a critical load in any member(s).

2. *Propeller thrust.*—The propeller thrust from a nacelle may be neglected in the detailed analysis of the wing structure, with the following exceptions:

a. When the nacelle location is such as to produce large local loads on the wing structure (nacelle above wing, etc.)

b. When, in multi-engined airplanes, nacelles are located at a considerable distance from the plane of symmetry, in which case the wing attachment structure should be analyzed for the case of full power applied on one side only.

D. Determination of Point of Application of the Resultant Air Loads on a Wing

1. A general method is outlined in Army-Navy-Civil Publication ANC-1 (3)[1], "Determination of the Points of Application of the Resultant Airloads on a Wing", 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.

2. A simplified version of the method presented in ANC-1 (3) is presented by Table IV and is applicable to those cases where the span distribution outlined in A1 is employed. In order that the procedure may be entirely clear, the following instructions are presented:

a. In general, the summation of all forces acting upon a wing can be expressed as a single resultant force acting at a certain point and a couple, or moment of air forces, about this point. If the point is so chosen that, at constant dynamic pressure, the moment of the air forces does not appreciably change with a change in the angle of attack of the airfoil, the point can be considered as the mean aerodynamic center of the wing. The resultant force can be resolved into the normal and chord components and represented by the average coefficients C_N and C_C, while the moment may be 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. The choice of a reference axis system is an arbitrary matter; any rectangular system of coordinates will be satisfactory. However, for the present case, it has been found convenient to choose a set of rectangular axes fixed in the airplane. The longitudinal or X-axis may be taken parallel to the line of thrust, parallel to the wing chord, perpendicular to the projection of the plane of the wing beams on the plane of symmetry, or any similar line in the plane of symmetry. The normal or Z-axis lies in the plane of symmetry perpendicular to the X-axis. The lateral or Y-axis is perpendicular to the plane of symmetry. The origin may be taken at any point in the plane of symmetry, preferably not too far from the wing. The resultant single force can then be expressed by its components in the XZ plane and the moment as a couple about the Y-axis. Forces which act up, aft, and toward the left wing tip are considered positive. Positive moments act counter-clockwise about their respective axes when viewed from the origin. (See figure 15.)

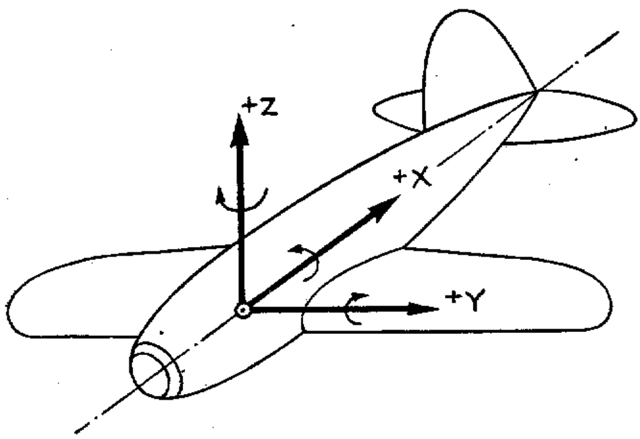


Figure 15.—Convention of axes, forces and moments.

c. An explanation and clarification of the procedure to be followed in filling out Table IV is as follows:

(1) The semi-wing should be divided into a convenient number of strips with a number or letter assigned to each and listed in column 1.

(2) The distance from the plane of symmetry to the centroid of the area of each strip is listed in column 2.

(3) Column 3 contains the width of the strip.

(4) The mean chord of each strip is listed in column 4.

(5) The area of each strip is found in column 5 by multiplying column 3 by column 4. The sum of this column should equal half the wing area as indicated below the table. This is a check on the correctness of the strip areas and nothing more should be done until good agreement is obtained.

(6) The data for column 6 is taken from the assumed span distribution curve, figure 14. 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.

(7) Column 7 contains the product of column 5 and 6. The sum of this column when divided by the sum of column 5 will give the value K_b, which is the ratio of the mean effective C_N to the value of C_{N_0} (at the root).

[1] To be published soon.

TABLE IV.—Determination of Point of Application of Resultant Airloads on a Wing

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

$\bar{y} = \frac{\Sigma(8)}{\Sigma(7)} =$ $\quad C_M = \frac{\Sigma(15)}{\Sigma(13)} =$

Check:
Wing Area $= \frac{2}{144}\Sigma(5) =$

$\bar{x} = \frac{\Sigma(10)}{\Sigma(7)} =$ $\quad K_b = \frac{\Sigma(7)}{\Sigma(5)} =$

$\bar{Z} = \frac{\Sigma(12)}{\Sigma(7)} =$

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

(8) Column 8 contains the product of columns 2 and 7. The sum of this column when divided by the sum of column 7 gives the lateral coordinate of the centroid of the semi-wing as indicated below the table.

(9) In column 9 is listed the distance of each section aerodynamic center aft of the YZ plane.

(10) Column 10 contains the product of column 7 and 9. The sum of this column when divided by the sum of column 7 gives the X-coordinate of the centroid as indicated below the table.

(11) In column 11 is listed the distance of each section aerodynamic center above the XY plane.

(12) Column 12 contains the product of columns 7 and 11. The sum of this column when divided by the sum of column 7 gives the Z-coordinate of the centroid as indicated below the table.

(13) Column 13 contains the product of columns 4 and 5. The sum of this column, when divided by the sum of column 5, gives the length of the mean aerodynamic chord. If a location is wanted for the mean aerodynamic chord, it should be drawn on the wing so that its aerodynamic center coincides with the centroid of lift determined by means of the preceding columns.

(14) Column 14 is a list of the section moment coefficients.

(15) Column 15 contains the product of columns 13 and 14. For wings that have no twist, the sum of this column when divided by column 13 gives the average moment coefficient for the wing.

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

E. Resultant Forces on Biplanes

1. The mean aerodynamic center location and the value of the mean aerodynamic chord for each wing panel can be found as outlined above. When wing flaps or other auxiliary high lift devices are used the mean effective moment coefficient for each wing panel should also be obtained. For a given flight condition, the values of C_N for each wing can be determined from Table IV. The

location of the mean aerodynamic center of the biplane and the determination of the resultant forces and moments can be accomplished as follows, referring to figure 16:

a. The mean aerodynamic center of the biplane cellule lies on a straight line connecting the mean aerodynamic centers of the two wing panels. The location on the line is determined from equation (*a*), figure 16.

b. Assuming that the mean effective moment coefficient is the same for each wing panel, the value of the mean aerodynamic chord for the biplane is determined from equation (b), figure 16.

c. If the mean effective moment coefficients for the two wing panels are different in value, the effective moment coefficient for the biplane can be determined from equation (c), figure 16.

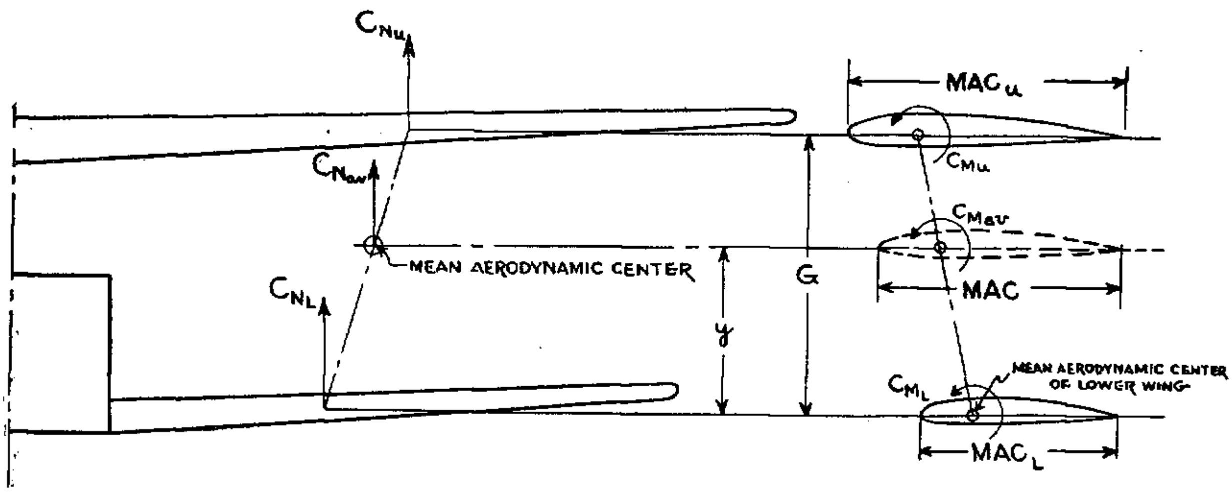
2. The mean aerodynamic center of a biplane, as determined above, is based on the relative values of the normal forces acting on each wing. When the average normal force coefficient for the entire biplane is near zero, the relative loading on the wings varies over a wide range and the mean aerodynamic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

a. A single location may be assumed for the mean aerodynamic center for all the balancing conditions.

b. When the investigation of two different span distributions is required, the more nearly constant span distribution may be used in determining the mean aerodynamic center and *MAC*.

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

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



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

(b) $$MAC=\frac{(MAC)_u S_u+(MAC)_L S_L}{S_u+S_L}$$

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

Figure 16.—Resultant forces on a biplane.

BALANCING LOADS

A. General

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

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

B. Balancing the Airplane

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

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

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

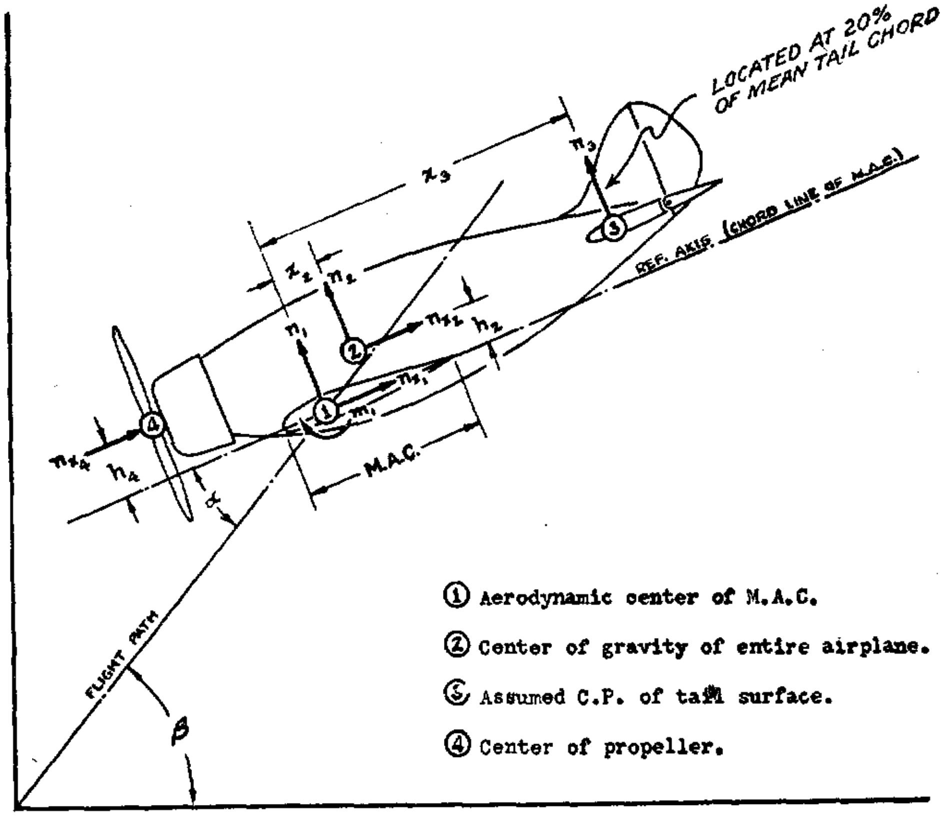
c. Assuming that it is possible for a load to be acting in the opposite direction on the elevator, it is recommended that the center of pressure of the horizontal tail be placed at *20 percent of the mean chord of the entire tail surface.* This arbitrary location may also be considered as the point of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

d. In figure 17 the external forces are assumed to be acting at four points only. The assumption can generally be made that the fuselage drag acts at the center of gravity. When more accurate data are available, the resultant fuselage drag force can of course be computed and applied at the proper point. In cases where large independent items having considerable drag (such as nacelles) are present, it is advisable to extend the set-up shown in figure 17 to include the additional items.

2. As shown in figure 17, a convenient reference axis is the basic chord line of the mean aerodynamic wing chord. (The basic chord line is usually specified along with the dimensions of the airfoil section.) The determination of the size and location of the *MAC* is outlined in "D" page 34. In determining the vertical location of the aerodynamic center of the *MAC* (point 1 of figure 17) the vertical position of the *MAC* in relation to the wing root chord, or other similar reference line, should be considered.

3. A tabular form will simplify the computations required to obtain the balancing loads for various flight conditions. A typical form for this purpose is shown in Table V. In using figure 17 and Table V the following assumptions and conventions should be employed:

a. If known distances or forces are opposite in direction from those shown in figure 17, a negative sign should be prefixed before inserting in the computations. For instance, in the case of a high-wing monoplane, h_2 will have a negative sign. Likewise n_{x4} will be either negative or zero in all cases. The direction of unknown forces will be indicated by the sign of the value obtained from the equations. A negative value of n_3 will usually be determined from the balancing process, indicating a down load on the tail. For conditions of positive acceleration the solution should give a negative value for n_2, as the inertia load will be acting downward. The convention



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

β = gliding angle, degrees.

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

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

x = horizontal distance from ① (positive rearward).

h = vertical distance from ① (positive upward).

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

Figure 17.—Basic forces in flight conditions.

for m_1 corresponds to that used for moment coefficients; that is, when the value of C_m is negative m_1 should also be negative, indicating a diving moment.

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

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

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

4. *Computation of Balancing Loads*.—In Table V the computation of balancing loads is indicated for typical flight conditions. The equations are based on the fact that the use of the average force coefficients in connection with the design wing area, mean aerodynamic chord, and mean aerodynamic center will give resultant forces and moments of the proper magnitude, direction and location. Provision is made in the table for obtaining the balancing loads for different gross weights. The table may be expanded to include computations for several loading conditions, special flight conditions, or conditions involving the use of auxiliary devices. It should be noted that a change in the location of the *CG* will require a corresponding change in the values of x_2 and h_2 on figure 17.

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

5. The following explanatory notes refer by number to items appearing in Table V:

(3) The wing loading, s, should be based on the design wing area.
(5) n_1=limit load factor required for the condition being investigated. (See 04.21.)
(8) Determine C_C as specified in 04.21. See also eq. 8, page 19.
(10) Propeller thrust, F_{pr}, should be determined from eq. 15, page 19 for conditions at V_L. For conditions at V_g assume $n_{x_4}=0$.
(11) The value of C_m' is specified in 04.21. For a biplane see "E," page 35. See also "D," page 34 in cases involving wing flaps.
(13) The net tail load factor, n_3, is found by a summation of moments about point (2) of figure 17, from which the following equation is obtained:

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

NOTE: The above explanatory notes apply only when the set-up shown in figure 17 is used. If a different distribution of external loads or a different system of measuring distances is employed, the computations should be correspondingly modified.

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

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

TABLE V.—Balancing Computations

(See fig. 17 for symbols)

No.	Item	V_L=mph		V_g=mph	
		I	II	III	IV
1	W=gross weight, pounds				
2	q=.00256 V^2				
3	s=①/S				
4	q/s=②/③				
5	n_1=applied wing load factor				
6	C_N=⑤/④				
7	C_L corresponding to C_N				
8	C_C				
9	n_{x_1}=⑧ x ④				
10	n_{x_4}=F_{pr}/①				
11	C_m=design moment coefficient				
12	m_1=⑪ x ④				
13	n_3=tail load factor				
14	n_2= − ⑤ − ⑬ = net load factor				
15	n_{x_2}= − ⑨ − ⑩ = chord load factor				
16	T = ① × ⑬ = tail load				
17	C_{mt}=moment coefficient of airplane less tail				
18	ΔC_m=⑰ − ⑪				
19	Δm_1=⑱ x ④				
20	Δn_3=⑲/(x_3-x_2)				
21	ΔT=① x ⑳				
22	T'=⑯ + ㉑				

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

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

04.22 CONTROL SURFACE LOADS.

04.220 General. In addition to the flight loads specified in § 04.21 the primary structure shall meet the requirements hereinafter specified to account for the loads acting on the control surfaces. The following loading conditions include the application of balancing loads (§ 04.128) derived from the symmetrical flight conditions and also cover the possibility of loading the control surfaces and systems in operating the airplane and by encountering gusts. See also § 04.27 for multiplying factors of safety required in certain cases.

The requirements for the design of control surfaces are based on the two separate functions of control surfaces: balancing and manoeuvering. The requirements are specified so as to account also for the effects of auxiliary control devices, gust loads, and control forces.

The average unit loading normal to any surface is determined by the force coefficient C_N and the dynamic pressure q, as shown by eq. 13, page 19. When dealing with tail surfaces, it is customary to specify the value of C_N for the entire surface, including both the fixed and movable surfaces. The total load so obtained is then distributed so as to simulate the conditions which exist in flight. In the case of ailerons, flaps or tabs, the value of C_N is usually determined only for the particular surface, without reference to the surface to which it is attached.

The average unit loading is usually assumed to be constant over the span. On account of the nature of the chord distribution curves specified in figures 18, 19, 20, it will be simpler to assume that the unit loading at the hinge line is constant over the span.

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

04.221 Horizontal tail surfaces.

04.2210 Balancing. *The limit* load acting on the horizontal tail surface shall not be less than the maximum balancing load obtained from Conditions *I, II, III, IV, VII*, and *VIII*. In computing these loads for tail surface design the moments of fuselage and nacelles shall be suitably accounted for. The factors given in Table VI shall be used, with the following provisions:

(*a*) For Conditions *I, II, III*, and *IV*, P (in figure 18) = 40 percent of net balancing load. (This means that the load on the fixed surface should be 140 percent of the net balancing load.) In any case P need not exceed that corresponding to a *limit* elevator control force of 150 pounds applied by the pilot.

(*b*) For Conditions *VII* and *VIII*, P may be assumed equal to zero.

TABLE VI.—Loading Conditions for Horizontal Tail Surfaces

Condition	Balancing (p. 40)	Maneuvering (p. 41)	Damping (p. 42)	Tab effects (p. 42)
1 Design speed (see *Airspeeds*, page 25)		V_p		V_L
2 Force coefficient, C_N		−.55 (down) +.35 (up)		
3 Average *limit* pressure, p. s. f.[1]		$C_N q_p$ [2]		
4 Chord distribution	Fig. 18	Fig. 19	Fig. 20	Fig. 19 [3]
5 Span distribution	Constant C_N	Constant C_N	Constant C_N	Constant C_N [2]
6 Minimum average *limit* pressure p. s. f.[1]		15		
7 Special requirements	None	None	None	None.
8 Minimum yield factor of safety, j_y	1.0	1.0	1.0	1.0
9 Minimum ultimate factor of safety, j_u	1.5	1.5	1.5	1.5

[1] Over entire horizontal tail.
[2] q_p is the dynamic pressure corresponding to V_p see p. 16 (04.118).
[3] Refers to main surface, disregarding tab; uniform pressure distribution may be assumed over tab.

The balancing loads should be applied to the horizontal tail surfaces, as the ailerons and the vertical tail surfaces are used only to a small extent for balancing purposes. The use of the vertical tail surfaces for balancing a multi-engined airplane having one engine dead is provided for in 04.2220.

An acceptable method for accounting for fuselage and nacelle moments in the determination of the balancing tail loads is given in "B6" page 39, and Table V. When wind tunnel tests have been used in this process the tail loads T′ in item 22 of this table may be used for design purposes. When, however, the −0.01 moment increment has been used in lieu of wind tunnel tests to account for fuselage and nacelle moments, the balancing loads to be used for design purposes should be taken as either item 16 or item 22 in Table V, whichever are most severe. This is to allow for a possible range of fuselage and nacelle moment coefficients.

The chord distribution illustrated in figure 18 is intended to simulate a relatively high angle of attack condition for the stabilizer, in which very high unit loadings can be obtained near the leading edge. The opposite loading required for the elevator in the balancing condition provides for the control force which the pilot might need to exert to hold the airplane in equilibrium.

In figure 18, the load from the elevator is shown as a concentrated load acting at the elevator hinge line. The hinge moment is, of course, resisted by the control system and therefore does not affect the stabilizer. It will be noted in 04.2210 (*b*) that the opposite elevator load, P, may be assumed equal to zero when the balancing load is obtained with flaps deflected (Conditions VII and VIII). This is based on the improbability of the pilots having to push on the elevator control in order to obtain balance with flaps down.

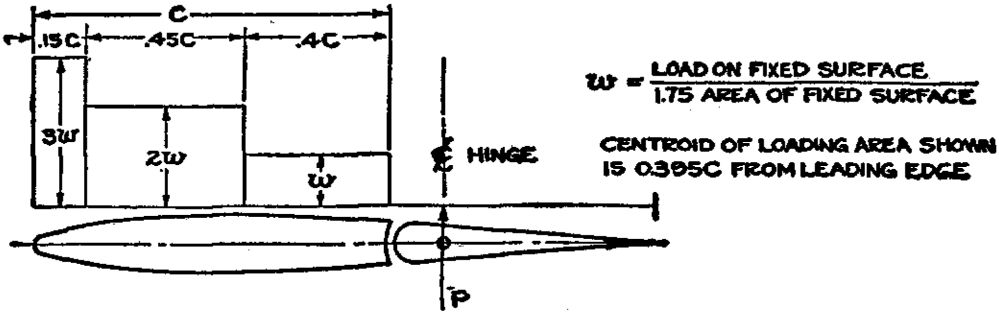


Figure 18.—"Balancing" distribution—horizontal tail.

04.2211 Maneuvering (horizontal surfaces). The factors and distributions specified in Table VI and figure 19 for this condition shall be used, together with the following provisions:

(*a*) The *limit* unit loading in either direction need not exceed that corresponding to a 200 pound force on the elevator control (see Table IX).

(*b*) The average *limit* unit loading shall not be less than 15 pounds per square foot (see Table VI).

The requirements for maneuvering loads are intended mainly to place the determination of such loads on a speed-force coefficient basis, to specify values which agree substantially with previous practice, and to provide for the effects of increasingly greater airplane speeds. It should be understood that the method is designed for application to conventional airplanes and that in determining the maneuvering loads the designer should consider the type of service for which the airplane is to be used.

The design values of C_N represent coefficients which can be attained by deflecting the control surfaces, the highest value representing the largest deflection of the movable surface expected at the design speed. Lower values are used for up loads on the horizontal tail surfaces and for the vertical tail surfaces, as the corresponding control forces are expected to be less in these cases. The numerical values of the coefficients are coordinated with the value of the factor K_p in the equation for design speed and do not represent the maximum coefficients which can be obtained with conventional control surfaces. Higher values may be desirable in certain cases, depending on the purpose of the airplane.

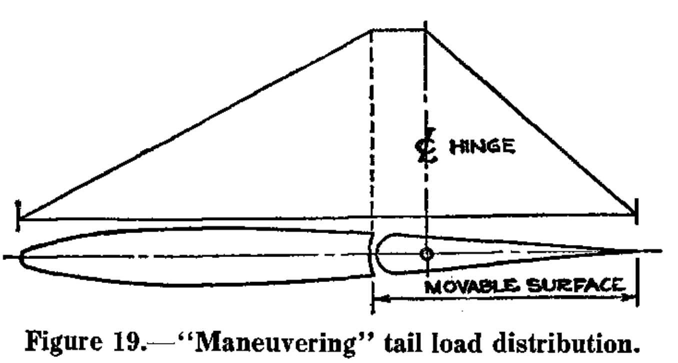


Figure 19.—"Maneuvering" tail load distribution.

The chord distribution shown in figure 19 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.

04.2212 Damping (horizontal surfaces). The total *limit* load acting down on the fixed surface (stabilizer) in the maneuvering condition (§ 04.2211) shall be applied in accordance with the load distribution of figure 20, acting in either direction. The load acting on the movable surface in the maneuvering condition may be neglected in determining the damping loads.

When a control surface is deflected suddenly the full maneuvering load tends to build up immediately, after which the airplane begins to acquire an angular velocity. This angular motion causes the direction of the relative air stream over the fixed surface to change, which causes the air load on this surface to build up in a direction such as to oppose the angular rotation of the airplane. This load is concentrated near the leading edge of the fixed surface and is commonly referred to as the damping load. It is provided for 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.

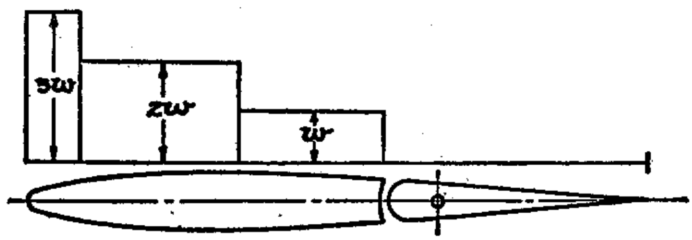


Figure 20.—"Damping" tail load distribution (see figure 18 for dimensions).

04.2213 Tab effects (horizontal surfaces). When a tab is installed so that it can be used by the pilot as a trimming or assisting device, a *limit* up load over the tab corresponding to the dynamic pressure at V_L and the maximum tab deflection shall be assumed to act in conjunction with the *limit* down load specified in § 04.2211, disregarding the provisions of § 04.2211 (*a*) applied over the remaining area. If the control force necessary to balance the resulting loads on the elevator and tab exceeds 200 pounds (Table IX), the loadings over the areas not covered by the tab may be reduced until the control force is equal to this maximum *limit* value.

The loading condition specified above is diagrammatically illustrated in figure 21. This condition represents the case of the tab load and the control force both acting so as to resist the hinge moment due to the air load on the movable surface. For convenience, the distances and moments can be computed for the neutral position of the movable surface and tab. Actually, the tab load will tend to decrease slightly when the movable surface is deflected, but this effect, being small and difficult to determine rationally, can be neglected.

04.222 Vertical tail surfaces.

04.2220 Maneuvering. The factors given in Table VII and figure 19 for this condition shall be used, with the following provisions:

(*a*) If the propeller axes are not in the plane of symmetry, the design speed shall not be less than the maximum speed in level flight with any engine inoperative.

(*b*) The *limit* unit loading in either direction need not exceed that corresponding to the maximum *limit* control force (Table IX) except as modified by paragraph (*c*) following.

(*c*) In any case the average *limit* unit loading shall not be less than the minimum pressure specified in Table VII for this condition.

The comments on page 41 in regard to horizontal surfaces also apply, in general, to the vertical surfaces.

It is specified that the value of V_p shall not be less than the level flight speed with one engine dead. This is based on the assumption that the unbalanced yawing moment present in such a condition will be balanced by the vertical tail surfaces. In some cases it may be advisable to increase the value of the normal force coefficient to account for features such as engines which are

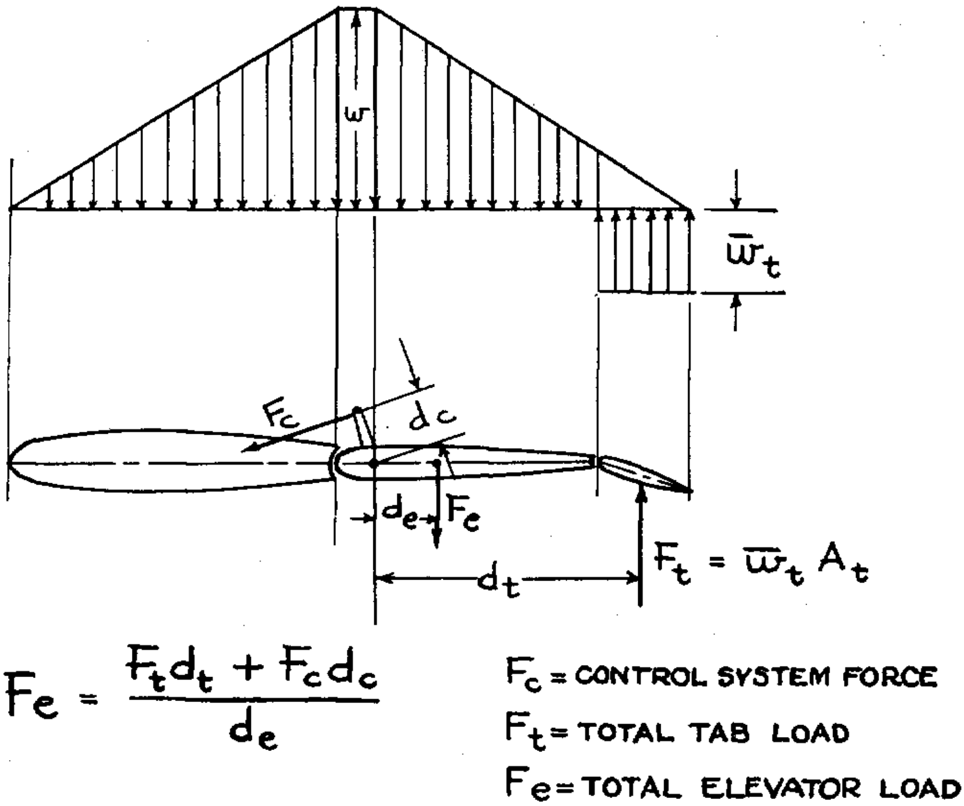


Figure 21.—Tab loading condition.

relatively far from the plane of symmetry. In estimating the speed with one engine dead the following approximate equation may be used:

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

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

04.2221 Damping (vertical surfaces). The total *limit* load acting on the fixed surface (fin) in the maneuvering condition shall be applied in accordance with the load distribution of figure 20, acting in either direction. The load acting on the movable surface in the maneuvering condition may be neglected in determining the damping loads.

The comments on page 42 in regard to horizontal surfaces also apply, in general, to the vertical surfaces.

04.2222 Gusts (vertical surfaces). The gust conditions specified in Table VII shall be applied, using the following formulae and provisions:

(*a*) The gust shall be assumed to be sharp-edged and to act normal to the plane of symmetry in either direction.

(*b*) The average *limit* unit pressure, $\overline{w}$, developed in striking the gust shall be determined from the following formula:

$\overline{w} = UVm/575$, where
$\overline{w}$ is in pounds per square foot,
U is in feet per second
V is in miles per hour and
m=slope of lift curve, C_L per radian, corrected for aspect ratio.

The aspect ratio shall not be taken as less than 2.0 in any case.

(*c*) This condition applies only to that portion of the vertical surface which has a well-defined leading edge.

(*d*) The chord distribution extending over the fixed and movable surfaces shall simulate that for a symmetrical airfoil, except that the distribution in figure 20 may be used where applicable.

TABLE VII.—Loading Conditions for Vertical Tail Surfaces

	Condition	Maneuvering (p. 42)	Damping (p. 43)	Gust (p. 43)	Tab effects (p. 44)
1	Design speed (see *Airspeeds*, p. 25)	[3] V_p		V_L	[4] V_L
2	C_N or gust	C_N=0.45		U=30 *fps*	
3	Average *limit* pressure, *p. s. f.*[1]	[2] $C_N q_p$		(*b*) p. 43	
4	Chord distribution	Fig. 19	Fig. 20	[5] Fig. 20	[6] Fig. 19
5	Span distribution	Constant C_N	Constant C_N	Constant C_N	[6] Constant C_N
6	Minimum average *limit* pressure, *p. s. f.*[1]	12			
7	Special requirements	(*b*) p. —	None	None	None
8	Minimum yield factor of safety, j_y	1.0	1.0	1.0	1.0
9	Minimum ultimate factor of safety, j_u	1.5	1.5	1.5	1.5

[1] Over entire vertical tail.
[2] $_{pp}$ is the dynamic pressure corresponding to V_p see p. 16 (04.118)
[3] See (*a*), p. 04.2220 for exception.
[4] See (*a*), p. 04.2223 for exception.
[5] See (*c*), p. 04.2222.
[6] Refers to main surface, disregarding tab; uniform pressure-distribution may be assumed over tab).

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

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

b. The chord distribution specified in figure 20 is applicable to those cases in which the mean chords of the effective fin and rudder areas are of approximately the same magnitudes. When this figure is used it should be noted that $\overline{w}$ refers to the average limit pressure over the total effective area of the vertical surface. The total load acting is therefore equal to $\overline{w}$ times the total effective area. This load is, however, applied to the fin only, in accordance with the specified distribution.

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

04.2223 Tab effects (vertical surfaces). When a tab is installed on the vertical movable tail surface so that it can be used by the pilot as a trimming device the *limit* unit loading over the entire vertical tail surfaces shall not be less than that corresponding to the maximum deflection of the tab together with simultaneous application of the following control force in a direction assisting the tab action:

For airplanes with all propeller axes in the plane of symmetry, zero.
For airplanes with propeller axes not in the plane of symmetry, 200 pounds.

The factors specified in Table VII for this condition shall be used, with the following exception:

(*a*) If the propeller axes are not in the plane of symmetry, the design speed V_L specified in Table VII may be reduced to the maximum speed in level flight with any engine inoperative.

04.2224 Special cases (vertical surfaces). A special ruling shall be obtained from the Administrator when an automatic pilot is used on airplanes with propeller axes not in the plane of symmetry.

04.223 Ailerons.

04.2230 Maneuvering. The factors given in Table VIII and figure 22 for this condition shall be used, with the following provisions:

(*a*) If the propeller axes are not in the plane of symmetry, the design speed shall not be less than the maximum speed in level flight with any engine inoperative.

(*b*) The *limit* unit loading in either direction need not exceed that corresponding to the maximum control force (Table IX) resisted by only one aileron, except as modified by paragraph (*c*) following.

(*c*) In any case the average *limit* unit loading shall not be less than the minimum pressure specified in Table VIII for this condition.

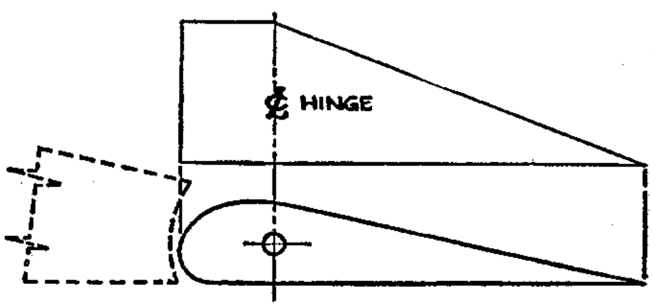


Figure 22.—Aileron load distribution.

TABLE VIII.—Loading Conditions for Ailerons

	Condition	Maneuvering (page 44)	Tab effects (page 45)
1	Design speed (see *Airspeeds*, p. 25)	[1] V_p	[3] V'_L
2	C_N or gust	C_N=0.45	
3	Average *limit* pressure, *p. s. f*	[2] $C_N q_p$	
4	Chord distribution	Fig. 22	[4] Fig. 22
5	Span distribution	Constant C_N	[4] Constant C_N
6	Minimum average *limit* pressure, *p. s. f*	12	
7	Special requirements	(*b*), page 44	
8	Minimum yield factor of safety, j_y	1.0	1.0
9	Minimum ultimate factor of safety, j_u	1.5	1.5

[1] See (*a*), 04.2230 for exception.
[2] q_p is the dynamic pressure corresponding to V_p, see *dynamic pressure*, page 16 (04.118.)
[3] V'_L is the maximum level flight air speed with any engine inoperative.
[4] Refers to main surface, disregarding tab; uniform pressure distribution may be assumed over tab.

04.2231 Tab effects (ailerons). (Applies only to airplanes with propeller axes not in the plane of symmetry.) When a tab is installed on one or both ailerons so that it can be used by the pilot to assist in moving the ailerons the *limit* unit loading over both ailerons shall be of sufficient magnitude and in such direction as to hold the ailerons in equilibrium with the tab or tabs deflected to the maximum position. The factors specified in Table VIII for this condition shall be used.

04.2232 Flying conditions (ailerons). The ailerons and their control system shall be capable of meeting all requirements specified in the basic symmetrical flying conditions so far as the latter produce symmetrical loads on the ailerons.

04.224 Wing flaps. Wing flaps shall be loaded in accordance with Conditions *VII* and *VIII* (§ 04.2141 and § 04.2142) and in addition shall be capable of developing an *ultimate* factor of safety of at least 1.5 with respect to any intermediate conditions which are more severe for any part of the flap or its operating mechanism.

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

04.225 Tabs. The limit forces acting on control-surface tabs shall be determined from the most severe combination of airplane speed and tab normal force coefficient likely to be obtained for any usable loading condition of the airplane and at speeds up to the design gliding speeds, V_g. An *ultimate* factor of safety of at least 1.5 shall be maintained.

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

04.23 CONTROL SYSTEM LOADS.

04.230 General. All control systems shall 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 surface to be in that position which produces the greatest load in the control system, except that the maximum and minimum control force limits in Table IX shall apply as hereinafter specified. The factors of safety specified in Table IX shall be used. See also § 04.27 for multiplying factors of safety required in certain cases. See also § 04.331 for operation requirements for control systems.

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

a. Differences between the actual and the assumed control surface load distribution.
b. Desirability of extra strength in the control system to reduce deflections.
c. Reduction in strength due to wear, play in joints, etc.

The maximum control force limits are based on the greatest probable forces which will be exerted by the pilot. These forces can be exceeded under severe conditions, but the probability of this occurrence is very low. The ultimate factor of safety of 1.5, which is required in any case, will permit the maximum limit load to be exceeded for a relatively short time without serious consequences.

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

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

04.2300 The forces in the control wires or push rods operating the movable surfaces shall be computed and their effect on the rest of the structure shall be investigated and allowed for in the design of such structure.

04.231 Elevator systems. In applying § 04.230 the control force specified in Table IX and figure 23 shall be assumed to act in a fore-and-aft direction and shall be applied at the grip of a control stick, or shall be equally divided between two diametrically opposite points on the rim of a control wheel.

TABLE IX.—Loading Conditions for Control Systems

(See General, page 45)

Condition	Elevator (page 46)	Rudder: Symmetrical thrust [1] (page 46)	Rudder: Unsymmetrical thrust [2] (page 46)	Aileron (page 46)	Flaps, tabs, etc. (page 46)
1 Maximum *limit* control force, pounds	200	200	200	80	None
2 Minimum *limit* control force, pounds	Fig. 23	130	200	Fig. 24	See *General.* p. 45.
3 Minimum yield factor of safety, j_y	1.0	1.0	1.0	1.0	1.0
4 Minimum ultimate factor of safety, j_u	1.5	1.5	1.5	1.5	1.5

[1] Propeller axes all in plane of symmetry.
[2] Propeller axes not all in plane of symmetry.

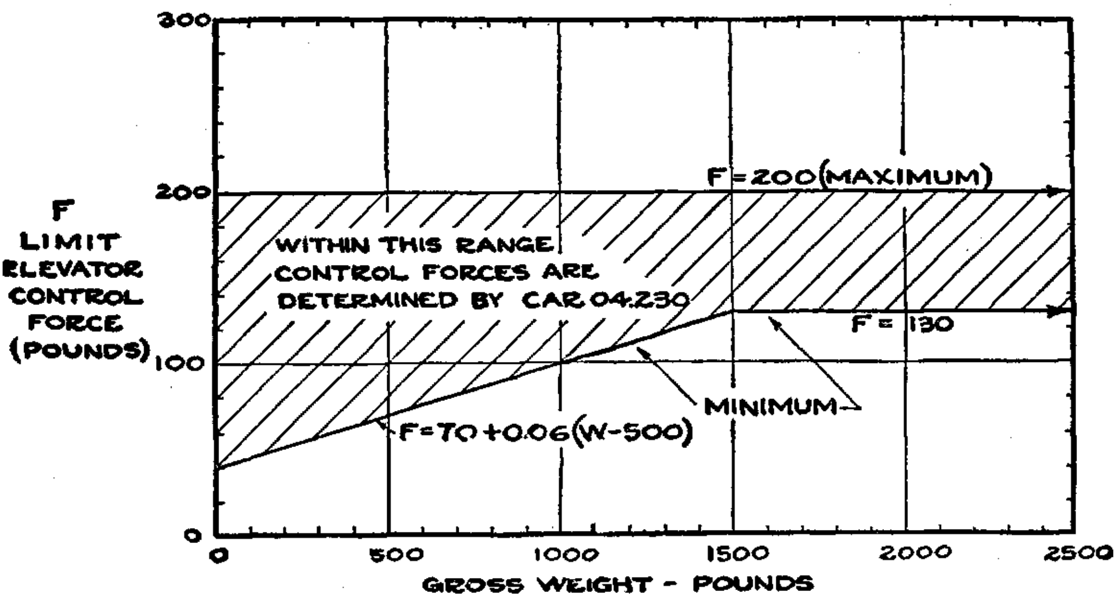


Figure 23.—Elevator control force limits.

04.232 Rudder systems. In applying § 04.230 the control force specified in Table IX shall be assumed to act in a direction which will produce the greatest load in the control system and shall be applied at the point of contact of the pilot's foot.

04.233 Aileron systems. In applying § 04.230 it shall be assumed that the ailerons are loaded in opposite directions. The control force specified in Table IX and figure 24 shall be assumed to act in a lateral direction at the grip of a control stick, or shall be assumed to act as part of couple equal to the specified force multiplied by the diameter of a control wheel. The following assumptions shall be made:

(*a*) For nondifferential ailerons, 75 percent of the stick force or couple shall be assumed to be resisted by a down aileron, the remainder by the other aileron; also, as a separate condition, 50 percent shall be assumed to be resisted by an up aileron, the remainder by the other aileron.

(*b*) For differential ailerons, 75 percent of the stick force or couple shall 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 shall be made.

04.234 Flap and tab control systems. In applying § 04.230 suitable minimum manual forces shall be assumed to act on flap and tab control systems and other similar controls.

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

The following design conditions apply to crank and twist type controls for airplanes certificated in the transport category:

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

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

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

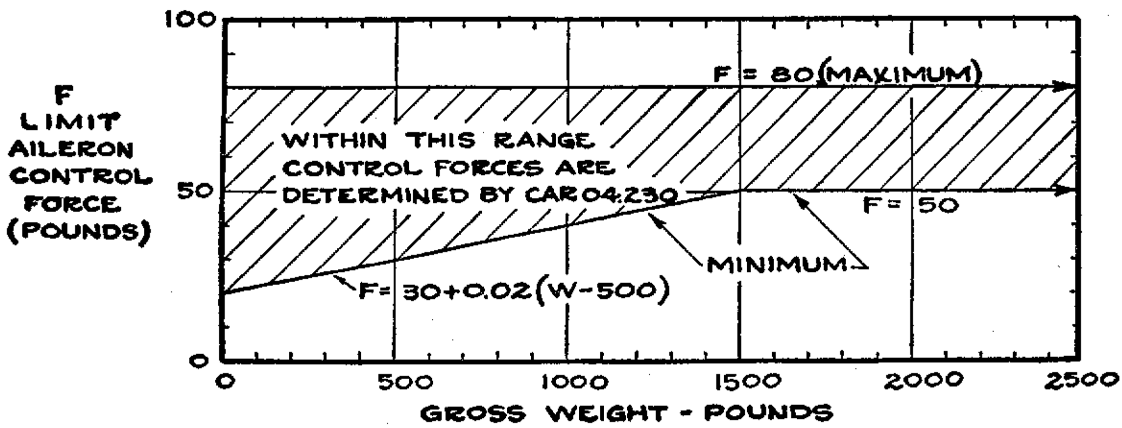


Figure 24.—Aileron control force limits.

4.24 GROUND LOADS.

04.240 General. The following conditions represent the minimum amount of investigation required for conventional (tail down type) landing gear. For unconventional types it may be necessary to investigate other landing attitudes, depending on the arrangement and design of the landing gear members. Consideration will be given to a reduction of the specified *limit* load factors when it can be proved that the shock absorbing system will positively limit the acceleration factor to a definite lower value in the drop test specified in § 04.2411. The minimum factors of safety are specified for each loading condition. See also § 04.27 for multiplying factors of safety required in certain cases.

1. **Tail wheel type gear.** The basic landing conditions outlined in 04.24 for conventional land type gear are tabulated in Table X. This chart can be used as a summary of the load factors for the landing conditions by inserting the actual values used.

2. **Nose wheel type gear.** The following design conditions have been found acceptable in certain cases of nose wheel gear. It is emphasized, however, that all unusual features of a particular design should be investigated to insure that all possible critical loadings have been considered. See also page 77 for a discussion of energy absorption tests.

a. Three-wheel landing with vertical reactions.—The minimum limit load is specified in figure 25. The value of the sum of the static ground reactions shall be the weight of the airplane less landing gear. The total load shall be divided between the front and rear gear in inverse proportion to the distances, measured parallel to the ground line, from the *CG* of the airplane less landing gear to the points of contact with the ground. The load on the rear gear shall be divided equally between wheels. Loads shall be assumed to be perpendicular to the ground line in the three-wheel landing attitude, with all shock absorbing units and tires deflected to one-half their total travel unless it is apparent that a more critical arrangement could exist. The critical positions of the *CG* shall be investigated. The minimum ultimate factor of safety shall be 1.5.

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

c. Two-wheel landing with vertical reactions—nose up.—The minimum limit load factor is specified in figure 25. The airplane shall be assumed to be in an extreme nose-up attitude. The gross weight of the airplane less the rear gear shall be assumed to act at the rear wheels in a direction perpendicular to the ground line. The total load shall be divided equally between the two rear wheels. The resultant moment on the airplane shall be balanced by inertia forces. The shock absorbers and tires shall be deflected to the same degree as in condition *a* above. The minimum ultimate factor of safety shall be 1.5.

TABLE X.—Landplane Landing Conditions for Tail Wheel Type Gear

Condition	Level	3-point	Side [1]	One-wheel landing [1]	Braked
04 Regulation	.241	.242	.243	.244	.245
Load factor n [2] (*limit*)	$\begin{cases} 2.80+\frac{9000}{W+4000} \, ^{3} \\ 3.00+0.133\,(W/S) \end{cases}$	Same as level	0.667	0.5 level	1.33
Attitude	Propeller axis horizontal	3-point	3-point [4]	Propeller axis horizontal	3-point [4]
Vertical component	nW [5]	nW [6 7]	nW [6]	nW [5]	nW [5]
Rearward component	Resultant through *CG* [8]	Zero	0.55 nW	Resultant [7 8] through *CG* (side view)	0.55 vertical
Side component	Zero	Zero	nW (inward)	Zero	Zero
Shock strut deflection	50% travel [9]	50% travel [9]	Static position	50% travel	50% travel
Tire deflection	50%	50%	25%	50%	25%

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

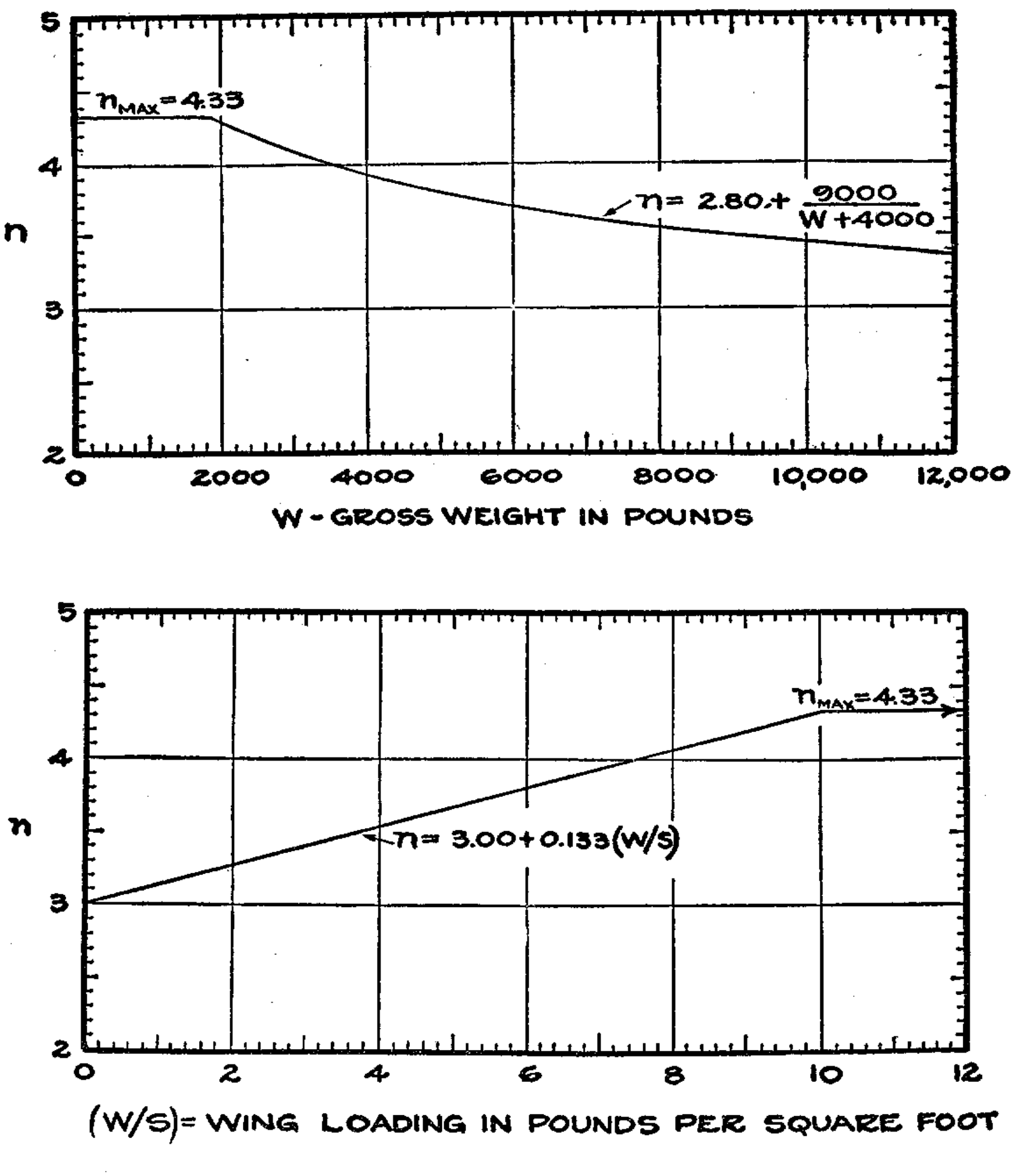
d. Two-wheel landing with inclined reactions—nose up.—The minimum limit load factor is specified in figure 25. The airplane shall be assumed to be in an extreme nose-up attitude. The resultant force shall be determined in the same manner as in condition *b* above except that the gross weight of the airplane less the rear gear shall be used. The total load shall be divided equally between the two rear wheels. The resultant moment on the airplane shall be balanced by inertia forces. The shock absorbers and tires shall be deflected to the same degree as in condition *a* above. The minimum ultimate factor of safety shall be 1.5.

e. Two-wheel landing with inclined reactions—nose down.—The minimum limit load factor is specified in figure 25. The airplane shall be assumed to be in a nose-down attitude with the front wheel just off the ground. The resultant force shall be determined in the same manner as in condition *b* above except that the weight of the airplane less the rear gear shall be used. The total load shall be divided equally between the two rear wheels. The resultant moment on the airplane shall be balanced by inertia forces. The shock absorbers and tires shall be deflected to the same degree as in condition *a* above. The critical position of the *CG* shall be investigated. The mimimum ultimate factor of safety shall be 1.5.

f. Two-wheel landing with brakes—nose down.—The minimum limit load factor shall be 1.33. The airplane shall be assumed to be in a nose-down attitude with the front wheel just off the ground. The gross weight of the airplane less the rear gear shall be assumed to act at the rear wheels in a direction perpendicular to the ground line. In addition, a horizontal aft component equal to 0.55 times the vertical shall be applied at each wheel at the points of contact with the ground. The total load shall be divided equally between the two rear wheels. The resultant moment on the airplane shall be balanced by inertia forces. The tires shall be assumed to have deflected not more than one-quarter the nominal diameter of their cross-section, and the deflection of the shock absorbers shall be the same as in condition *a* above. The minimum ultimate factor of safety shall be 1.5.

g. Side drift landing.—The minimum limit load factor is specified in figure 25. The attitude of the airplane, the vertical components of the landing gear reactions, and the deflections of the shock absorbers and tires shall be the same as in condition *a* above. In addition, a horizontal aft component and a side component, each equal to 0.25 times the vertical component, shall be applied at each wheel at the points of contact with the ground. The resultant moment on the airplane shall be balanced by inertia forces. The minimum ultimate factor of safety shall be 1.5.

h. Side drift landing with brakes.—The minimum limit load factor shall be 1.0. The attitude of the airplane, the static ground reactions on the front and rear gear, and the deflections of the shock absorbers and tires shall be the same as in condition *a* above. The total load on the rear gear shall, however, be applied entirely on one wheel. In addition, a side component equal to 0.75 times the vertical component shall be applied at each wheel at the points of contact with the ground. The side load at the rear wheel shall be assumed to act inward and the side load



NOTE: USE THE CHART INDICATING THE LOWER VALUE

Figure 25.—Limit load factors for level and 3-point landing conditions.

at the nose wheel shall be assumed to act in the same direction. A horizontal aft component equal to 0.55 times the vertical component shall be applied at the point of contact with the ground of each wheel equipped with brakes. (It should be noted that one rear wheel is not loaded.) The resultant moment on the airplane shall be balanced by inertia forces. The minimum ultimate factor of safety shall be 1.5.

i. One-wheel landing.—An investigation of the fuselage structure is required for a one-wheel landing in which only those loads obtained on one side of the fuselage in condition *e* above are applied. The resulting limit load factor is therefore one-half of the minimum limit load factor specified in figure 25. (This condition is identical with condition *e* above insofar as the landing gear structure is concerned.) The minimum ultimate factor of safety shall be 1.5.

3. **Ski gear.** As noted in 04.2410, the ground loads for ski gear are the same as for wheel gear. However, the strength of skis and ski pedestals must be substantiated in accordance with the requirements of CAR 15.12. See also Manual 15.12. Approval of ski installations is covered on page 13. The Canadian ski gear requirements, which are of interest to manufacturers contemplating export to Canada, are listed in Inspection Handbook, Chapter XII.

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

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

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

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

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

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

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

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

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

n_o= ultimate load factor (value from fig. 25 times 1.5),

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

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

04.241 Level landing. The minimum *limit* load factor is specified in figure 25. The resultant of the ground reaction shall be assumed to be a force lying at the intersection of the plane of symmetry and a plane in which are located the axles and the center of gravity of the airplane less chassis. The propeller axis (or equivalent reference line) shall be assumed horizontal and the basic value of the vertical component of the resultant of the ground reaction shall be equal to the gross weight of the airplane minus chassis and wheels. The horizontal component shall be of the magnitude required to give the resultant force the specified direction except that it need not be greater than 25 percent of the vertical component. The resultant of the ground reaction shall be assumed to be divided equally between wheels and to be applied at the axle at the center of the wheel. The shock-absorber unit and tires shall be assumed to be deflected to half their total travel, unless it is apparent that a more critical arrangement could exist. The minimum *ultimate* factor of safety shall be 1.5.

04.2410 If a sliding element instead of a rolling element is used for the landing gear, a horizontal component of one-half of the vertical component shall be used to represent the effect of ground friction, except that ski gear which is designed and used only for landing on snow and ice may be designed for the same horizontal component as wheel gear.

04.2411 Energy absorption. The level landing condition specified in § 04.241 shall be assumed to be produced by a free drop, in inches, equal to 0.36 times the calculated stalling speed (V_S) in miles per hour, except that the heights of free drop shall not be less than 18 inches for airplanes employing devices which increase the normal sinking speed, but need not exceed 18 inches when such devices are not employed. The height of free drop is measured from the bottom of the tire to the ground, with the landing gear extended to its extreme unloaded position. (See §§ 04.340 and 04.440.)

The definition of stalling speed V_s used in drop height calculations is given in 04.113. If accurate flight test data for the airplane in question, or for a very similar airplane, are available, such data may be used as a basis for calculating the power-off stalling speed. However, the determination of speeds in the flight tests used in this connection should not involve an extensive extrapolation of the airspeed calibration. See also discussion on "Energy Absorption Tests," page 77.

04.242 Three-point landing. The minimum *limit* load factor is specified in figure 25. The value of the sum of the static ground reactions shall be the gross weight of the airplane less chassis. The total load shall be divided between the chassis and tail skid or wheel in inverse proportion to the distances, measured parallel to the ground line, from the center of gravity of the airplane less chassis to the points of contact with the ground. The load on the chassis shall be divided equally between wheels. Loads shall be assumed to be perpendicular to the ground line in the tree-point landing attitude, with all shock absorbers and tires deflected to the same degree as in level landing. The tail wheel or skid installation shall also be investigated for this condition. The minimum *ultimate* factor of safety shall be 1.5.

04.2420 Energy absorption. The three-point landing condition specified in § 04.242 shall be assumed to be produced by a free drop as specified under § 04.2411. This requires shock absorption by both main wheels and tail wheel (or skid). (See §§ 04.340 and following discussion, page 77, and .04.440, page 112.)

04.243 Side load. The minimum *limit* load factor shall be 0.667. The weight of the airplane shall be assumed to act on one wheel in a direction perpendicular to the ground. In addition, a side component of equal magnitude shall be assumed to act inward and normal to the plane of symmetry at the point of contact of the wheel, and an aft component equal to 0.55 times the vertical component shall be assumed to act parallel to the ground at such point. The airplane shall be assumed to be in a three-point attitude with the shock absorbers deflected to their static position and the tires deflected one-quarter the nominal diameter of their cross section. The minimum *ultimate* factor of safety shall be 1.5.

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

04.244 One-wheel landing. An investigation of the fuselage structure is required for a one-wheel landing, in which only those loads obtained on one side of the fuselage in the level landing condition are applied. The resulting load factor is therefore one-half of the level landing load factor. (This condition is identical with the level landing condition insofar as the landing gear structure is concerned.) The minimum *ultimate* factor of safety shall be 1.5.

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

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

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

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

04.245 Braked landing. The minimum *limit* load factor shall be 1.33. Airplanes equipped with brakes shall be investigated for the loads incurred when a landing is made with the wheels locked and the airplane is in an attitude such that the tail skid or wheel just clears the ground. The weight of the airplane less chassis shall be assumed to act on the wheels in a direction perpendicular to the ground line in this attitude. In addition, a component parallel to the ground line shall be assumed to act at the point of contact of the wheels and the ground, the magnitude of this component being equal to the weight of the airplane less chassis times a coefficient of friction of 0.55. The tire in all cases shall be assumed to have deflected not more than one-quarter the nominal diameter of its cross section, and the deflection of the shock absorbers shall be the same as in level landing. The minimum *ultimate* factor of safety shall be 1.5.

04.246 Side loads on tail wheel or skid. Suitable assumptions shall be made to cover side loads acting on tail skids or tail wheels which are not free to swivel or which can be locked or steered by the pilot.

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

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

04.25 WATER LOADS.

04.250 General. The following requirements shall apply to the entire airplane, but have particular reference to hull structures, wings, nacelles, and float supporting structure. The requirements for certification of floats as individual items of equipment are specified in Part 15. The minimum factors of safety are specified for each loading condition. See also § 04.27 for multiplying factors of safety required in certain cases. Detail design requirements for hulls and floats are specified in § 04.45.

The basic water landing conditions are given in Table XI. This chart can be used as a summary of the load factors for the landing conditions by inserting the actual values used.

The landing conditions outlined for float seaplanes correspond, in general, to the conditions used for landplanes. These conditions apply to conventional float installations and in such cases will provide a sufficient range of loadings. When unconventional types of float bracing are employed it may be advisable to investigate other landing attitudes, depending on the type of loading which appears to be most critical for the structure.

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

TABLE XI.—Seaplane Landing Conditions

COMPONENT	FLOAT [1]		
Condition	Inclined reaction	Vertical reaction	Side landing
04 Regulation	.251	.252	.253
N (limit)	[2] 4.20	[2] 4.33	4.0
Vertical reaction	[3] nW	[3] nW	[3] nW
Rearward reaction	¼ vertical	0	0
Side reaction	0	0	¼ vertical
Resultant	Through *CG* less floats and bracing		In plane through *CG* and perpendicular to propeller axis
Factor of safety	[4] 1.85 [5] 1.50		1.50
Attitude	Propeller axis or reference line horizontal		

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

04.251 Landing with inclined reactions (float seaplanes). The vertical component of the *limit* load factor shall be 4.20 except that it need not exceed a value given by the following formula:

$$n=3.0+0.133\ (W/S).$$

The propeller axis (or equivalent reference line) shall be assumed to be horizontal and the resultant water reaction to be acting in the plane of symmetry and passing through the center of gravity of the airplane less floats and float bracing but inclined so that its horizontal component is equal to one-quarter of its vertical component. The forces representing the weights of and in the airplane shall be assumed to act in a direction parallel to the water reaction. The weight of the floats and float bracing may be deducted from the gross weight of the airplane.

04.2510 For the design of float attachment members, including the members necessary to complete a rigid brace truss through the fuselage, the minimum *ultimate* factor of safety shall be 1.85. For the remaining structural members the minimum *ultimate* factor of safety shall be 1.50.

04.252 Landing with vertical reactions (float seaplanes). The *limit* load factor shall be 4.33, acting vertically, except that it need not exceed a value given by the following formula:

$$n=3.0+0.133\ (W/S).$$

The propeller axis (or equivalent reference line) shall be assumed to be horizontal, and the resultant water reaction to be vertical and passing through the center of gravity of the airplane less floats and float bracing. The weight of the floats and float bracing may be deducted from the gross weight of the airplane.

04.2520 The minimum factors of safety shall be the same as those specified in § 04.2510.

04.253 Landing with side load (float seaplanes). The vertical component of the *limit* load factor shall be 4.0, to be applied to the gross weight of the airplane less floats and float bracing. The propeller axis (or equivalent reference line) shall be assumed to be horizontal and the resultant water reaction shall be assumed to be in the vertical plane which passes through the center of gravity of the airplane less floats and float bracing and is perpendicular to the propeller axis. The vertical load shall be applied through the keel or keels of the float or floats, and evenly divided between the floats when twin floats are used. A side load equal to one-fourth of the vertical load shall be applied along a line approximately half-way between the bottom of the keel and the level of the water line at rest. When built-in struts are used, check calculations shall be made for the built-in struts with the side load at the level of the water line at rest. When twin floats are used, the entire side load specified shall be applied to the float on the side from which the water reaction originates. The minimum *ultimate* factor of safety shall be 1.50.

04.254 Boat seaplanes.

04.2540 Local bottom pressures. (*a*) Maximum local pressure. The maximum value of the *limit* local pressure shall be determined from the following equation:

$$p_{max}=0.055\ V_s^{1.4}\left(1+\frac{W}{50{,}000}\right)^{1/4},\ \text{where}$$

p=pressure, pounds per square inch
V_s=stalling speed, flaps down, power on, in miles per hour. (To be calculated on the basis of wind tunnel data or flight tests on previous airplanes.)
W=design weight.

The minimum *ultimate* factor of safety shall be 1.5. (*b*) Variation in local pressure. The local pressures to be applied to the hull bottom shall vary in accordance with figure 26. No variation from keel to chine (beamwise) shall be assumed, except when the chine flare indicates the advisability of higher pressures of the chine. (*c*) Application of local pressure. The local pressures determined from § 04.2540 (*a*) and figure 26 shall be applied over a local area in such a manner as to cause the maximum local loads in the hull bottom structure.

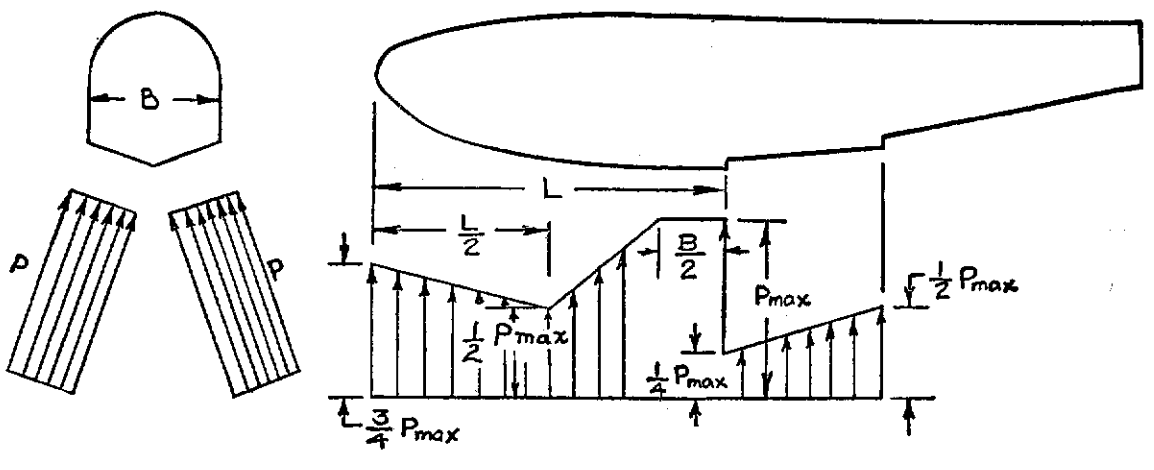


Figure 26.—Distribution of local pressures—boat seaplanes.

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

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

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

04.2541 Distributed bottom pressures. (*a*) For the purpose of designing frames, keels, and chine structure, the *limit* pressures obtained from § 04.2540 (*a*) and figure 26 shall he reduced to *one-half* the "local" values and simultaneously applied over the entire hull bottom. The loads so obtained shall be carried into the side-wall structure of the hull proper, but need not be transmitted in a fore-and-aft direction as shear and bending loads. The minimum *ultimate* factor of safety shall be 1.5. (*b*) Unsymmetrical loading. Each floor member or frame shall be designed for a load on one side of the hull centerline equal to the most critical symmetrical loading, combined with a load on the other side of the hull centerline equal to one-half of the most critical symmetrical loading.

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

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

04.2542 Step loading condition. (*a*) Application of load. The resultant water load shall be applied vertically in the plane of symmetry so as to pass through the center of gravity of the airplane (in full load condition). (*b*) Acceleration. The *limit* acceleration shall be 4.33. (*c*) Hull shear and bending loads. The hull shear and bending loads shall be computed from the inertia loads produced by the vertical water load. To avoid excessive local shear loads and bending moments near the point of water load application, the water load may be distributed over the hull bottom, using pressures not less than those specified in § 04.2541 (*a*). The minimum *ultimate* factor of safety shall be 1.5.

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

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

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

04.2543 Bow loading condition. (*a*) Application of load. The resultant water load shall be applied in the plane of symmetry at a point one-tenth of the distance from the bow to the step and shall be directed upward and rearward at an angle of 30 degrees from the vertical. (*b*) Magnitude of load. The magnitude of the *limit* resultant water load shall be determined from the following equation:

$P_b = \frac{1}{2} n_s W_e$, where

P_b is the load in pounds,

n_s is the step landing load factor,

W_e is an effective weight which is assumed equal to one-half the design weight of the airplane.

(*c*) Hull shear and bending loads. The hull shear and bending loads shall be determined by proper consideration of the inertia loads which resist the linear and angular accelerations involved. To avoid excessive local shear loads, the water reaction may be distributed over the hull bottom, using pressure not less than those specified in § 04.2541 (*a*). The minimum *ultimate* factor of safety shall be 1.5.

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

Considering the arbitrary nature of the hull loading conditions, it seems reasonable to dispense with numerous refinements in specifying the loading condition and to apply a concentrated load at some specified point in an arbitrarily chosen direction. Therefore, the bow impact load is applied to the keel at a point one-tenth of the distance from the bow to the step, and in an upward and rearward direction at an angle of 30° from the vertical. See figure 27a. As this loading condition will produce a combination of vertical horizontal, and angular accelerations, all items of weight will produce inertia loads accordingly.

In those cases when it would be otherwise unnecessary to calculate the moment of inertia of the airplane about its lateral axis, the specified condition may be replaced by a simplified loading by making some conservative assumptions. To simplify the computation of inertia loads, an "average" linear acceleration factor can be used. The approximate method then consists of applying to the keel at a point one-tenth of the distance from the bow to the step a load equal to $n_bW_e=\frac{1}{2}n_sW_e$ and computing the inertia loads over the forward portion of the hull by using a load factor of $0.65n_b$ applied vertically, together with a horizontal component of $0.50n_b$. This would result in an unbalanced bending moment and shear by the rearward portion of the hull. Since this condition is not likely to be critical for the rearward portion of the structure, these unbalanced forces and moments need not be applied to the rear portion for design purposes. The "simplified" system for the bow loading condition is illustrated by figure 27a.

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

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

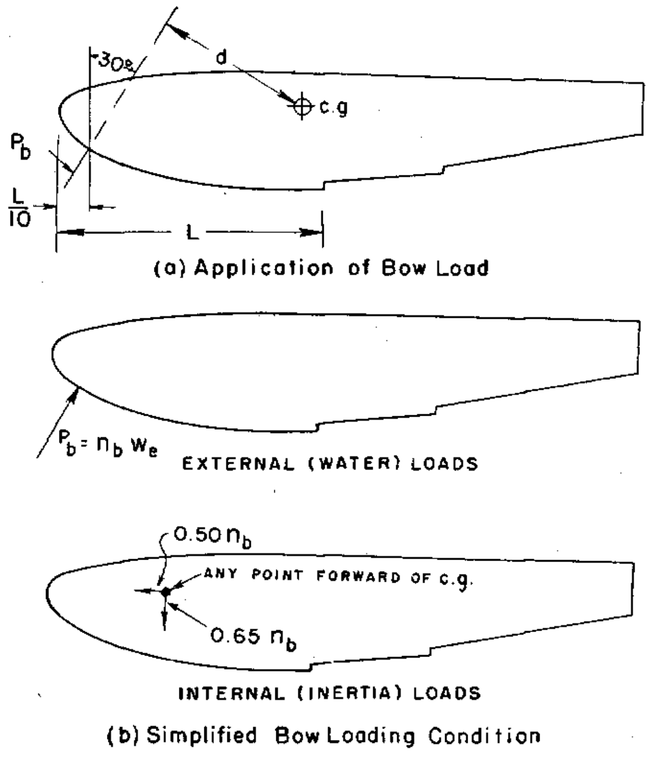


Figure 27.

04.2544 Stern loading condition. (*a*) Application of load. The resultant water load shall be applied vertically in the plane of symmetry and shall be distributed over the hull bottom from the second step forward with an intensity equal to the pressures specified in § 04.2541 (*a*). (*b*) Magnitude of load. The *limit* resultant load shall equal three-quarters of the design weight of the airplane. (*c*) Hull shear and bending loads. The hull shear and bending loads shall be determined by assuming the hull structure to be supported at the wing attachment fittings and neglecting internal inertia loads. This condition need not be applied to the fittings or to the portion of the hull ahead of the rear attachment fittings. The minimum *ultimate* factor of safety shall be 1.5.

To simplify the computations and to decrease the amount of work required, the area to be used with the pressures specified in 04.2541(*a*) may be taken as the projected area of the hull bottom on the plane defined in the last paragraph under 04.2542, page 54.

04.2545 Side loading condition. (*a*) Application of load. The resultant water load shall be applied in a vertical plane through the center of gravity. The vertical component shall be assumed to act in the plane of symmetry and the horizontal component at a point half-way between the bottom of the keel and the load waterline at design weight (at rest). (*b*) Magnitude of load. The *limit* vertical component of acceleration shall be 3.25 and the side component shall be equal to fifteen percent of the vertical component. (*c*) Hull shear and bending loads. The hull shear and bending loads shall be determined by proper consideration of the inertia loads or by introducing couples at the wing attachment points. To avoid excessive local shear loads, the water reaction may be distributed over the hull bottom, using pressures not less than those specified by § 04.2541 (*a*). The minimum *ultimate* factor of safety shall be 1.5.

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

To avoid excessive local shear loads, the water reaction may be distributed over the hull bottom. The area to be used in determining pressures (for comparison with those specified in 04.2541 (*a*)) should be the projected area of the hull bottom on the plane defined in the last paragraph under 04.2542, page 54. The load to be used in determining pressures should be the vertical component of the resultant load.

04.257 Seaplane float loads. Each main float of a float seaplane shall be capable of carrying the following loads when supported at the attachment fittings as installed on the airplane. The minimum *ultimate* factor of safety shall be 1.5. (*a*) A *limit* load, acting upward, applied at the bow end of the float and of magnitude equal to one-half of that portion of the airplane gross weight normally supported by the particular float. (*b*) The *limit* load specified in paragraph (*a*), above, acting upward at the stern. (*c*) A *limit* load, acting upward, applied at the step and of magnitude equal to 1.33 times that portion of the airplane gross weight normally supported by the particular float.

04.2570 Seaplane float bottom loads. Main seaplane float bottoms shall be designed to withstand the following loads. The minimum *ultimate* factor of safety shall be 1.5. (*a*) A *limit* load of at least 5.33 pounds per square inch over that portion of the bottom lying between the first step and a section at 25 percent of the distance from the step to the bow. (*b*) A *limit* load of at least 2.67 pounds per square inch over that portion of the bottom lying between the section at 25 percent of the distance from the step to the bow and a section at 75 percent of the distance from the step to the bow. (*c*) A *limit* load of at least 2.67 pounds per square inch over that portion of the bottom lying between the first and second steps. If only one step is used, this load shall extend over that portion of the bottom lying between the step and a section at 50 percent of the distance from the step to the stern.

04.258 Wing-tip float loads. Wing-tip floats and their attachment, including the wing structure, shall be analyzed for each of the following conditions, using a minimum *ultimate* factor of safety of 1.5. (*a*) A *limit* load acting vertically up at the completely submerged center of buoyancy and equal to three times the completely submerged displacement. (*b*) A *limit* load inclined upward at 45 degrees to the rear and acting through the completely submerged center of buoyancy and equal to three times the completely submerged displacement. (*c*) A *limit* load acting parallel to the water surface (laterally) applied at the center of area of the side view and equal to one and one-half times the completely submerged displacement.

04.2580 The primary wing structure shall incorporate sufficient extra strength to insure that failure of wing-tip float attachment members occurs before the wing structure is damaged.

04.259 Miscellaneous water loads.

04.2590 Seawing loads. Special rulings shall be obtained from the Administrator for the strength requirements for seawings.

04.26 SPECIAL LOADING CONDITIONS.

04.260 Engine torque. In the case of engines having five or more cylinders the stresses due to the torque load shall be multiplied by a *limit* load factor of 1.5. For 4, 3, and 2 cylinder engines the *limit* load factors shall be 2, 3, and 4, respectively. The torque acting on the airplane structure shall be computed for the take-off power desired and the propeller speed corresponding thereto. (See § 04.744.) The engine mount and forward portion of the fuselage and nacelles shall be designed for this condition. The minimum *ultimate* factor of safety shall be 1.5, unless higher factors are deemed necessary by the CAA in order to make special provision for conditions such as vibration, stress concentration, and fatigue.

04.261 High angle of attack and torque. The limit loads determined from § 04.260 shall be considered as acting simultaneously with 75 percent of the *limit* loads determined from Condition *I* (§ 04.2131). The engine mount, nacelles and forward portion of the fuselage (when a nose engine is installed) shall be designed for this condition. The minimum *ultimate* factor of safety shall be 1.5.

04.2610 The engine mounts nacelles, and forward portion of the fuselage (when a nose engine is installed) shall be investigated for the *limit* loads determined from Condition *I* (see §§ 04.2131 and 04.2160) acting simultaneously with the *limit* loads due to the engine torque determined in accordance with § 04.260, except that the engine power and the propeller speed shall correspond to the design power (§ 04.105) or the output specified for climbing flight (see § 04.744), whichever is higher. The minimum *ultimate* factor of safety shall be 1.5.

04.262 Side load on engine mount. The *limit* load factor for this condition shall be equal to one-third of the *limit* load factor for Flight Condition *I* (§ 04.2131) but shall in no case be less than 1.33. The engine mount and forward section of the fuselage and nacelles shall be analyzed for this condition, considering the *limit* load to be produced by inertia forces. The minimum *ultimate* factor of safety shall be 1.5.

04.263 Up load on engine mount. For engine mounts the *limit* load in each member shall be arbitrarily assumed as 50 percent of that in the level landing condition but of opposite sign. The minimum *ultimate* factor of safety shall be 1.5.

04.264 Passenger loads. Passenger loads in the accelerated flight conditions shall be computed for a standard passenger weight of 170 pounds, and a minimum *ultimate* factor of safety of 1.50 shall be used, except that seats and berths need not be designed for the reduced weight gust conditions specified in § 04.2160. This shall not exempt the primary structure from such gust conditions.

04.2640 Structures to which safety belts are attached shall be capable of withstanding an *ultimate* load of 1,000 pounds per person applied through the safety belt and directed upward and forward at an angle of 45 degrees with the floor line.

04.265 Local loads. The primary structure shall be designed to withstand local loads caused by dead weights and control loads. Baggage compartments shall be designed to withstand loads corresponding to the maximum authorized capacity. The investigation of deadweight loads shall include a sufficient number of reduced weight gust conditions to insure that the most severe combinations have been investigated. See § 04.90 for standard weights.

04.266 Rigging loads. Structures braced by wires (or tie-rods) shall be capable of developing an ultimate factor of safety of 1.5 with respect to the *limit* loads due to rigging the wires to 20 percent of their rated strength (strength of wire, not terminal). When the structure is such that all wires cannot be simultaneously rigged to 20 percent of their rated loads, a rigging condition shall be assumed in which the average of the rigging loads, expressed in percent, equals 20. (See also § 04.274.) The above condition need not be superimposed on other loading conditions, but the Administrator may require additional investigation for residual rigging loads when such investigation appears necessary. (See also § 04.315.)

The requirements are based on the necessity for proportioning wire sizes so as to prevent an excessive load being produced in any wire while rigging any other wire. They provide for an average rigging load of 20 percent. This means that when the maximum allowable ratio of rigging loads (two to one) exists between two wires, one will be assumed to be rigged to 13.3 percent, the other to 26.7 percent. If a larger ratio were permitted, such as three to one for instance, there would be a possibility of obtaining an excessively high rigging load in one wire while rigging the other to a relatively low percentage of its rated load.

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

04.267 Air loads on struts. External wing-brace struts which are at an angle of more than 45 degrees with the plane of symmetry and which have a cross-sectional fineness ratio of more than 3 shall be assumed to act as lifting air foils and shall be designed to carry the resultant transverse loads in combination with the specified axial loads. In computing the *limit* loads the strut sections shall be assumed to have a normal force coefficient equal to 1.0 and the total air load shall be based on the exposed area of the strut. The chord components and vertical reactions of such air load and the lift contributed by the strut shall not be considered in the analysis of the wing.

04.27 MULTIPLYING FACTORS OF SAFETY.

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

TABLE XII.—Additional (Multiplying) Factors of Safety

	Item	Reference Part 04	Additional yield factor of safety, j_y	Additional ultimate factor of safety, j_u	May be covered by item No.
1	Fittings (except control system fittings)	.271	None	1.20	2, 4, 5, 6, 7, 8, 9
2	Castings	.272	None	2.00	7, 8
3	Parallel double wires in wing lift truss	.273	None	1.05	4
4	Wires at small angles	.274	None	See 04.27	
5	Double drag truss wires	.275	None	See 04.27	
6	Torque tubes used as hinges	.276	None	1.5	
7	Control surface hinges [1]	.277	None	6.67	
8	Control system joints [1]	.277	None	3.33	
9	Wire sizes	.278	None	See 04.27	
10	Wing lift truss (landing conditions only)	.279	None	1.10	

[1] For bearing stresses only.

04.271 Fittings. All fittings in the primary structure shall incorporate the multiplying factor of safety specified in Table XII. For this purpose fittings are defined as parts used to connect one primary member to another and shall include the bearing of those parts on the members thus connected. Continuous joints in metal plating and welded joints between primary structural members are not classified as fittings. (See also §§ 04.4030 and 04.4031.)

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

04.272 Castings. All castings used in the primary structure shall incorporate a multiplying factor of safety not less than that specified in Table XII.

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

04.273 Parallel double wires. When parallel double wires are used in wing lift trusses each wire shall incorporate a multiplying factor of safety not less than that specified in Table XII.

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

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

The requirement is based on the fact that a decrease in the angle, between a lift wire and a spar, will greatly increase the deflection for a given loading. The formula used is so adjusted as to maintain, approximately, the deflection which would be obtained for a 30° angle between the wire and the spar. It will be noted that the value of K becomes 1.0 when the angle is 30°. Since K approaches infinity as the angle approaches zero, it will be found impractical to design wire-braced structures for small angles between the wires and the members which they support.

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

04.276 Torque tubes used as hinges. When steel torque tubes are employed in direct bearing against strap-type hinges they shall incorporate a multiplying factor of safety at the hinge point not less than that specified in Table XII. (See also § 04.422.)

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

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

04.278 Wire sizes. (See §§ 04.403, 04.4032, and 04.412.)

04.279 All structural members in the wing lift truss system which transmit direct loads from the landing gear shall, in the landing conditions, incorporate a multiplying factor of safety not less than that specified in Table XII.

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

04.3 PROOF OF STRUCTURE

04.30 GENERAL.

Proof of compliance with the loading requirements outlined in § 04.2 shall be made in a manner satisfactory to the Administrator and may consist of structural analyses, load tests, flight tests, references to previously approved structures, or combinations of the above. Any condition which can be shown to be noncritical need not be further investigated.

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

04.3000 The structural analysis shall be based on guaranteed minimum mechanical properties of the materials specified on the drawings, except in cases where exact mechanical properties of the materials used are determined.

Acceptable methods for computing the allowable loads and stresses corresponding to the minimum mechanical properties of various materials are given in the Army-Navy-Civil Publication ANC-5, "Strength of Aircraft Elements", obtainable from the Superintendent of Documents, Government Printing Office, Washington 25, D. C.

04.3001 The effects of welding, form factors, stress concentrations, discontinuities, cut-outs, instability, end fixity of columns and vibration shall be accounted for when such factors are present to such an extent as to influence the strength of the structure.

04.301 Combined structural analysis and tests. In certain cases it will be satisfactory to combine structural analysis procedure with the results of load tests of portions of the structure not subject to accurate analysis. In such cases test results shall be reduced to correspond to the mechanical properties of the materials actually used in the airplane. When a unit other than the specific one tested is incorporated in the airplane presented for certification, test results shall be reduced to correspond to the minimum guaranteed mechanical properties of the materials specified on the drawings.

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

04.302 Load tests. Proof of compliance with structural loading requirements by means of load tests only is acceptable provided that strength and proof tests (see §§ 04.126 and 04.127) are conducted to demonstrate compliance with §§ 04.200 and 04.201, respectively, and further provided that the following sub-paragraphs are complied with.

04.3020 The tests shall be supplemented by special tests or analyses to prove compliance with multiplying factor of safety requirements. (See § 04.27.)

04.3021 When a unit other than the specific one tested is incorporated in the airplane presented for certification, the results of *strength* tests shall be reduced to correspond to the minimum guaranteed mechanical properties of the materials specified on the drawings, unless test loads are carried at least 15 percent beyond the required values.

04.3022 The determination of test loads, the apparatus used, and the methods of conducting the tests shall be satisfactory to the Administrator.

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

04.3023 The tests shall be conducted in the presence of a representative of the Administrator unless otherwise directed by the Administrator.

04.303 Flight load tests. Proof of strength by means of flight load tests will not be accepted unless the necessity therefor is established and the test methods are proved suitable to the satisfaction of the Administrator.

04.304 Load tests required. The following load tests are required in all cases and shall be made in the presence of a representative of the Administrator unless otherwise directed by the Administrator:

(*a*) Strength tests of wing ribs. (See § 04.313.)
(*b*) Pressure tests of fuel and oil tanks. (See § 04.623.)
(*c*) Proof tests of tail and control surfaces. (See § 04.32.)
(*d*) Proof and operating tests of control systems. (See §§ 04.33 and 04.331.)

04.31 PROOF OF WINGS.

The strength of stressed-skin wings shall be substantiated by load tests (§ 04.302) or by combined structural analysis and tests (§ 04.301). The torsional rigidity of the wings shall be within a range of values satisfactory for the prevention of flutter. Compliance with such torsional rigidity requirement shall be demonstrated by static tests or other methods acceptable to the Administrator.

A. Determination of Spar Loading

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

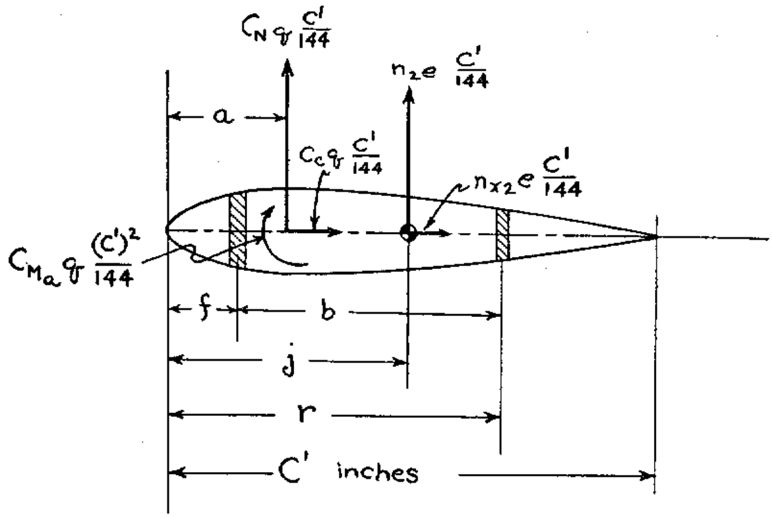


Figure 28.—Unit section of a conventional 2-spar wing.

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

$$y_f = [\{C_N(r-a)+C_{M_a}\}q+n_2e(r-j)]\frac{C'}{144b}$$

$$y_r = [\{C_N(a-f)-C_{M_a}\}q+n_2e(j-f)]\frac{C'}{144b}$$

Where y_f=net running load on front spar, lbs/inch.
y_r=net running load on rear spar, lbs/inch.
a, b, f, j, and r are shown on figure 28 and are *all expressed as fractions of the chord* at the station in question.

NOTE: the value of a must agree with the value on which C_{M_a} is based

q=dynamic pressure for the condition being investigated.
C_N and C_{M_a} are the airfoil coefficients at the section in question.
C' is the wing chord, in inches.
e is the average unit weight of the wing, in pounds per square foot, over the chord at the station in question. It should be computed or estimated for each area included between the wing stations investigated, unless the unit wing weight is substantially constant, in which case a constant value may be assumed. By properly correlating the values of e and j, the effects of local weights, such as fuel tanks and nacelles, can be directly accounted for.
n_2 is the *net* limit load factor representing the inertia effect of the whole airplane acting at the *CG*. The inertia load always acts in a direction opposite to the net air load. *For positively accelerated conditions n_2 will always be negative,* and vice versa. Its value and sign are obtained in the balancing process outlined under "Balancing Loads," page 37.

3. The computations required in using the above method are outlined in Tables XIII and XIV, in a form which is convenient for making calculations and for checking. The following modifications and notes apply to these tables:

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

b. The local values of C_N, item 14, are determined from the design value of C_N in accordance with the proper span distribution curve.

c. Item 15 provides for a variation in the local value of C_M. For Condition *I*, the value of C_M should be determined from the design value of CP by the following equation, using item numbers from Tables XIII and XIV:

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

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

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

TABLE XIII.—Computation of Net Unit Loadings (Constants)

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

[1] These values will depend on the amount of disposable load carried in the wing.

TABLE XIV.—Computation of Net Unit Loadings (Variables)

Condition ....

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

		Distance b from root					
14	$C_{Nb}=C_{NI}(\text{etc.})\times R_b/+K_b$						
15	C_{N_a} (variation with span)						
	Front Spar						
16	$(14)\times(9)$						
17	$(16)+(15)$						
18	$(17)\times q$						
19	$n_2\times(8)\times(11)$						
20	$(18)+(19)$						
21	$y_f=(20)\times(13)$, lbs/inch						
	Rear Spar						
22	$(14)\times(10)$						
23	$(22)-(15)$						
24	$(23)\times q$						
25	$nn_2\times(8)\times(12)$						
26	$(24)+(25)$						
27	$y_r=(26)\times(13)$, lbs/inch						
	Chord Spar						
28	C_C (variation with span)						
29	$(28)\times q$						
30	$n_{x_2}\times(8)$						
31	$(29)+(30)$						
32	$y_c=(31)\times(2)$, lbs/inch						

B. Determination of Running Chord Load

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

$$y_c=[C_c q+n_{x_2}e]C'/144$$

Where y_c=running chord load, lbs/inch

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

q=dynamic pressure for the condition being investigated.

n_{x_2}=net limit chord load factor approximately representing the inertia effect of the whole airplane in the chord direction. The value and sign are obtained in the balancing process outlined under "Balancing Loads," page 37. *Note that when C_c is negative, n_{x_2} will be positive.*

e and C' are the same as in paragraph A2, page 60.

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

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

b. The relative location of the wing spars and drag truss will affect the drag truss loading produced by the chord and normal air forces. This can easily be accounted for by correcting the value of C_c as indicated in paragraph A2, page 19 and figure 4a.

3. It is often necessary to consider the local loads produced by the propeller thrust and by the drag of items attached to the wing. The general rules concerning these items are outlined in "C", page 33. The drag of nacelles built into the wing is usually so small that it can be safely neglected. The drag of independent nacelles and that of wing-tip floats can be computed by using a rational drag coefficient or drag area in conjunction with the design speed. The beam and torsional loads applied to the wing through the attachment members should also be considered in the analysis. In general, the effects of nacelles or floats can be separately computed and added to the loads obtained in the design conditions.

C. Determination of Running Load and Torsion at Elastic Axis

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

2. As shown in figure 29, x denotes the location of the reference axis, expressed as a fraction of the chord. The net running load along the locus of the points x and the net running torsion about these points are found from the following equations:

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

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

Where y_x is in pounds per inch run.

m_x is in inch pounds per inch run.

x is expressed as a fraction of the chord.

C' is the wing chord, in *inches*.

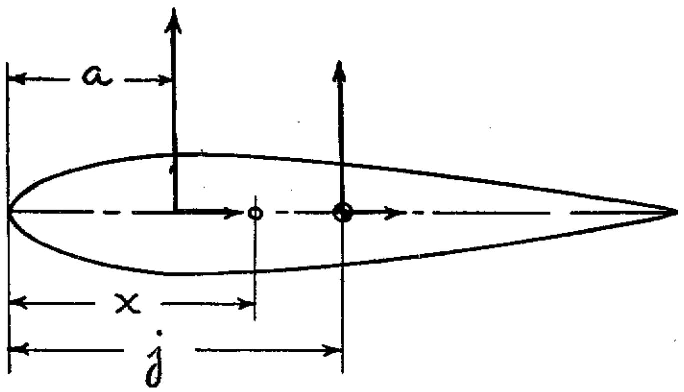


Figure 29.—Section showing location of elastic axis.

The remaining symbols are explained under A, page 60. (As noted previously, n_2 will always be negative in positively accelerated conditions.)

3. The computations required for this form of analysis can be conveniently carried out through the use of tables similar to XIII and XIV. The items appearing in each table would be changed to correspond to the equations given in 2 above. The computation of the running chord load can be made in the manner outlined in "B", page 62.

D. Lift-truss Analysis

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

2. *Redundant wire bracing.*—When two or more wires are attached to a common point on the wing but are not parallel, the following approximate equations may be used for determining the load distribution between wires, provided that the loads so obtained are increased 25 percent.

$$P_1=\left[\frac{V_1A_1L_1L_2^3}{V_1^2A_1L_2^3+V_2^2A_2L_1^3}\right]B$$

$$P_2=\left[\frac{V_2A_2L_1^3L_2}{V_1^2A_1L_2^3+V_2^2A_2L_1^3}\right]B$$

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

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

3. *Indeterminate wing cellules.*—In biplanes which have two complete lift truss and drag truss systems interconnected by an N strut, a twisting moment applied to the wing cellule will be resisted in an indeterminate manner, as each pair of trusses can supply a resisting couple. An exact solution involving the method of least work, or a similar method, can be used to determine the load distribution. For simplicity, however, it is usually assumed that the drag trusses resist only the direct chord loads and that all the normal loads and torsional forces are resisted by the lift trusses. This assumption is usually conservative for the lift trusses, but does not adequately cover the possible loading conditions for the drag trusses. In the usual biplane arrangement the lower drag truss will tend to be loaded in a rearward direction by the wing moment. Design Condition *VI* (04.2136) therefore represents the most critical condition for the lower drag truss. This condition should be investigated by assuming that a relatively large portion (approximately 75 percent) of the torsional forces about the aerodynamic center are resisted by the drag trusses. In the case of a single-lift-truss biplane, the drag trusses must, of course, resist the entire moment of the air forces with respect to the axis of the lift truss.

E. Wing Torsion Tests and Determination of Coefficient of Torsional Rigidity C_{TR}

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

2. *Set-up.*—The wing should be mounted on the fuselage or a suitable jig, either of which should be anchored solidly to the floor or wall to prevent movement or displacement of the wing. The landing gear should be blocked on the airplane. The torque load may be applied to the wing tip through several beams clamped to the wing as near to the tip as is practical, such as the outermost drag truss compression rib location. The platform cables should be attached to the torque beams an equal distance forward and aft of the elastic axis of the wing. This axis may be located experimentally by rocking the torque beam and noting the nodal point on the wing chord as viewed from the tip. Typical set-ups are shown in figure 30. Care should be taken to see that the strength of the local wing structure at the points of application of the torque loads from the beams is adequate. For conventional two spar wood wings, it is advisable to apply the load directly to the

spars through wood blocks rather than attempt to carry the load through a rib to the spars. Wings which are to be fabric covered should be tested uncovered, unless a certain amount of conservatism is considered in comparing the results with figure 36, in order to simulate the conditions found in service.

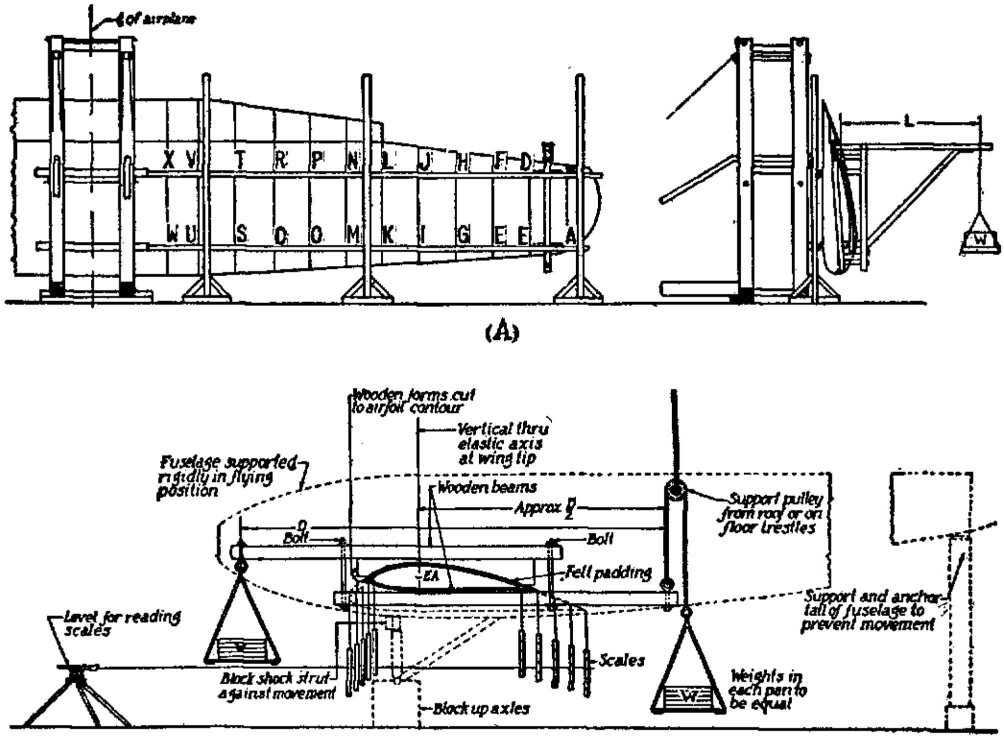


Figure 30.—Set-up for torsional test of wing.

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

3. *Loading.*—The following procedure may be used:

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

b. Add a sufficient amount of weight to each platform until readable deflections are obtained. In general, for most aircraft from 1,500 lbs. to 25,000 lbs. gross weight, it will be found desirable to make this first torque moment (in.-lbs.) equal numerically to twice the gross weight of the airplane. For aircraft below 1,500 lbs. gross weight and biplanes, 70 percent of the above values may be used as a first trial. Care should be taken to include the tare weights of the platforms in the torque computations.

c. Take readings of all scales.

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

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

f. The data to be recorded are the load applied; its lever arm; the deflection readings at selected points; and the exact location of these points both along the span and along the chord of the wing. It would be desirable to use a table such as shown on page 65 which would include all computations necessary for determining C_{TR}.

TABLE XV

WING TORSION TEST OF.................... MODEL.................... SERIAL NO....................

DATE.............................. TORQUE ARM..............................inches

BY........................ LOCATED........................inches from wing tip

MOMENTS (M) 1. $W_1 \times$ ARM =
2. $W_2 \times$ ARM =
3. $W_3 \times$ ARM =

Platform load (incl. platform wt.) (lbs)	Section A–B		C–D		E–F		G–H		I–J		K–L		Etc
	Front	Rear	F	R	F	R	F	R	F	R	F	R	—
DEFLECTION READINGS OF....................WING (in.)													
0	F_o	R_o											
1 W_1	F_1	R_1											
2 W_2	F_2	R_2											
3 W_3													
0													
DEFLECTIONS OF....................WING (in.)													
1 W_1	F_1-F_o	R_1-R_o											
2 W_2													
3 W_3													
TOTAL DEFLECTION OF....................WING (in.) = F+R													
1 W_1													
2 W_2													
3 W_3													
(C) CHORD DISTANCE BETWEEN DEFLECTION POINTS, F and R (in.)													
ANGLE OF TWIST OF....................WING (degrees) $= 57.3 \frac{\text{(total defl)}}{C}$													
1 W_1													
2 W_2													
3 W_3													
(L) SEMI-SPAN DISTANCE FROM WING TIP (in.)													
$dL/d\Theta = 1/$TANGENT TO "Θ" VS "L" CURVE AT SECTIONS													
1 W_1													
2 W_2													
3 W_3													
$C_{TB} \times 10^{-6} = M \frac{dL}{d\Theta} \times 10^{-6}$													
1 W_1													
2 W_2													
3 W_3													

Remarks:

4. *Interpretation of results.*—Having obtained the leaning and tailing edge deflections (F and R in Table XV) or a corresponding set of data, the angle of twist at each section of the wing for a given

torque, or platform load, is calculated and plotted against the wing semi-span measured from the wing tip.

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

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

or

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

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

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

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

where

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

$d\Theta$=Angle of twist in degrees, in length dL (in inches), caused by a torque of M inch pounds. Referring to the curve of angle of twist (Θ) vs. semi-span (L) shown in figure 31, it will be seen that $\frac{d\Theta}{dL}$=slope of the tangent drawn to the curve at any given point. Hence, it is only necessary to draw the required tangent to the curve at the value of L at which the C_{TR} is desired and obtain $\frac{d\Theta}{dL}$ to use in the above formula for C_{TR}.

It is very important that the tangent line be drawn accurately. This can best be done by

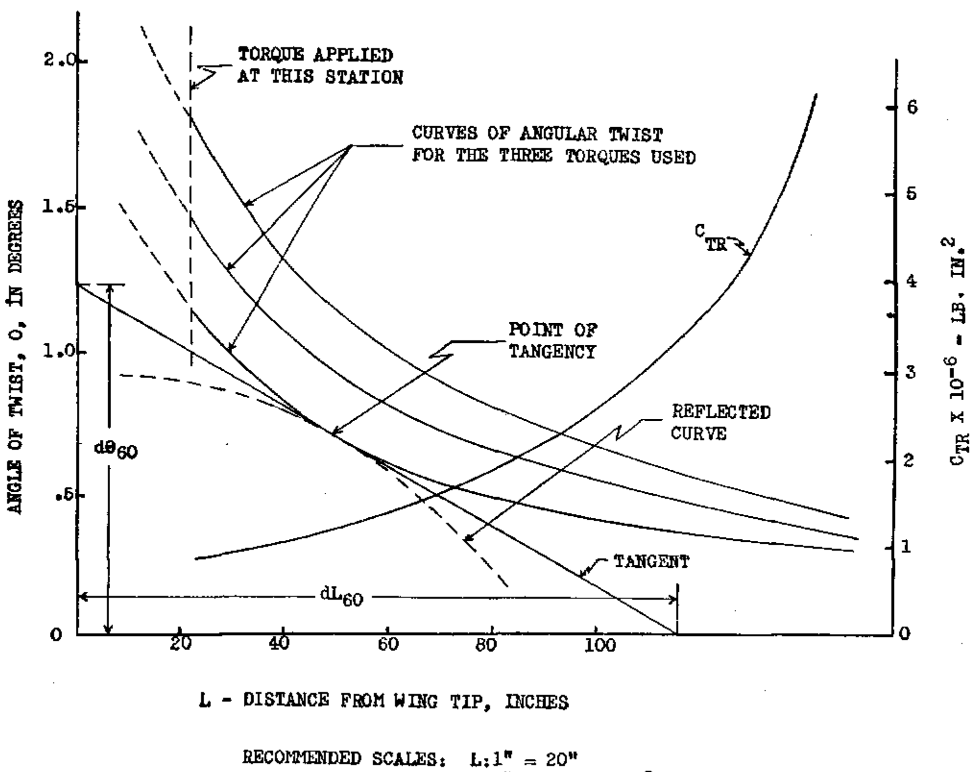


Figure 31.—Plot of results of torsion tests.

first drawing the reflected curve to the point of tangency (original curve may be drawn on transparent paper and used reversed or the tangent spotted in directly by use of a small mirror), and then by bisecting the resulting angle, as shown in figure 31 for a wing section 60 inches from the wing tip. The tangent line should be extended to both axes so that the slope of the line may be computed accurately which in this example is equal to $d\Theta_{60}/d\Theta_{60}$.

C_{TR} should be computed for each of the three torque conditions used at a number of points along the wing semi-span and plotted against the distance from the wing tip (L). This curve will show the variation of torsional rigidity throughout the semi-span and may be used for purposes of comparison with wings similarly tested. (See paragraph 2, page 90, and figure 36.)

04.310 Redundancies. Wing cellules in which the division of loading between lift trusses and drag trusses is indeterminate shall be analyzed either by an acceptable method for indeterminate structures or by making assumptions which result in conservative design loads for all members.

04.311 Beams. The following points shall be covered in the proof of strength of wing beams, in addition to any special types of possible failure peculiar to the structure.

A. Wood Spars

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

B. Metal Spars—General

1. The bending moments and shears should be computed by precise formulas which allow for the effects of the axial loads. Formulas for shear can be developed by differentiating the formulas for bending moments. The values of EI used in the computations should preferably be determined from a test on a section of beam subjected to loads in the plane of the beam and normal to its axis. In such tests it is recommended that the beam be simply supported at the lift truss fittings and subjected to equal concentrated loads, at or near the third points of the span, of such magnitude that the maximum shear and bending moment on the test specimen are in the same ratio as are the maximum primary shears and bending moments on the corresponding spans of the beam in the airplane. If this is not practicable, the shear on the test beam should be relatively larger than in the airplane. The deflections in the test should be read to the degree of precision necessary to obtain computed values of EI which are accurate within ± 5 percent.

2. When such a test cannot be made, the value of EI may be computed from the geometrical properties of the section and the elastic properties of the material used, but before being used in the formulas for computing deflections, shears, or secondary bending moments, this value should be multiplied by a correction factor to allow for shear deformation, play in joints, and lack of precision in computing the geometric properties of irregular sections. The correction factors recommended are 0.95 for beams having continuous webs that are integral with the chords, extruded I, and similar beams; 0.85 for built-up plate girders having continuous webs connected to the chord by riveting; 0.75 for beams with webs having lightening holes of such shape that the beam cannot be analyzed as a truss.

C. Truss-Type Metal Spars

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

2. In metal truss spars, for which L/j is greater than unity, the bending moments and shears on the spar should be obtained by the use of the precise formulas. The values of EI to be used in these formulas should be obtained whenever possible from deflection tests of the type described above under "B". When tests are not practicable the deflections used for determining EI

may be obtained by the use of any of the standard methods of computing the deflections of a truss, the assumed loading being that which would be used in a test. In computing these deflections it should be assumed that there is from 0.005- to 0.010-inch slip in the joint at each end of each web member of a riveted or bolted truss. No slip need be assumed in welded joints. Whether the deflections are obtained by test or are computed, EI values should be obtained for at least three points in each span of the truss and the average used in the precise formulas. When an external load parallel to the axis of the spar is applied at any section at a point other than the centroid of the chords at that section considered as a unit, it should be treated in the precise formulas as an equivalent combination of an axial load at that centroid and a bending moment.

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

4. *Design of chord members.*—The column length should be assumed as the centerline distance between truss joints for bending in the plane of the truss, using a restraint coefficient of not more than 2.0. For bending laterally it should be assumed as the distance between drag struts except that:

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

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

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

D. Thin-Web Metal Spars

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

E. Stressed-Skin Wings

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

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

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

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

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

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

04.3110 Secondary bending. When axial loads are present the required minimum *ultimate* factor of safety shall be introduced before the computation of the bending moments in order to insure that the required *ultimate* loads can be supported by the structure.

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

04.3111 Lateral buckling. The ability of beams to resist lateral buckling shall be proved.

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

04.3112 Webs. The strength of shear webs shall be proved.

04.3113. When axial load is present tests are required to determine the effective *EI* in the case of truss-type beams and beams having unconventional web construction.

04.3114 Joint slippage in wood beams. When a joint in a wood beam is designed to transmit bending from one section of the beam to another or to the fuselage, the stresses in each part of the structure shall be calculated on the assumption that the joint is 100 percent efficient (except in mid-bay for which see § 04.4110) 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 shall be designed to carry the most severe loads determined from the above assumptions.

04.3115 Bolt holes. In computing the area, moment of inertia, etc., of wood beams pierced by bolts, the diameter of the bolt hole shall be assumed to be one-sixteenth inch greater than the diameter of the bolt.

04.3116 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 degrees to the beam shall be considered. The more conservative method of neglecting the web entirely may be employed.

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

04.313 Ribs. The strength of ribs shall be proved by test to at least 125 percent of the *ultimate* loads for the most severe loading conditions, except that consideration will be given to structural analyses in conjunction with suitable specimen test data when it can be demonstrated to the satisfaction of the Administrator that it is impractical to simulate the actual loading conditions in a static test. Such analyses shall, on the basis of guaranteed minimum material properties, show proof of strength at 125 percent of the required *ultimate* loads. The following points shall also apply in proving the strength of ribs.

A. Test Requirements

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

The leading edge portion of the rib may be very severly loaded in Conditions *II* and *IV*. An investigation of the maximum down loads on this portion should be made when V_s exceeds 200 mph. (See "Leading Edge Loads," page 33.) When this requirement does not apply, it should be demonstrated that the rib structure ahead of the front spar is strong enough to withstand its portion of the test load acting in the reverse direction. A test for this condition will be required in the case of a rib which appears to be weak.

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

B. Test loadings

The following loadings are acceptable for two-spar construction when the rib forms a complete truss between the leading and trailing edges. (See "Chord Distribution," page 33 for other cases.)

TABLE XVI.—*Rib Load Points for High Angle of Attack*

A

PD Classification	Camber[2]	$C_{m_{a.c.}}$	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 −0.019[3]	0.5	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
		−.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	40.7	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	6.5	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 NACA Reports 610 and 628.
[2] Expressed as percent chord.
[3] Airfoils with + values of $C_{m_{a.c.}}$ are classified with those having a $C_{m_{a.c.}}=0$.

B

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 −0.0199	0.7	2.3	3.9	6.8	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	17.6	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	2.8	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

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 XVI A or XVI B. In order to determine which set of load points is applicable to the particular airfoil used, it is first necessary to determine the following airfoil characteristics:

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

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

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

These characteristics are readily obtainable for most airfoils from NACA Technical Reports Nos. 610 and 628. For airfoils in the 10 or 11 classification, the load points should be taken from Table XVI A, using the line corresponding to the $C_{m_{a.c.}}$ value of the airfoil. (Table XVI B should also be used for rib loading points in cases where the *PD* 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 XVI B, using the line corresponding to the $C_{m_{a.c.}}$ and the camber of the airfoil. In cases where the actual position of load number 1 is less than 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 greater than 60 inches, 8 equal loads for chords less than 60 inches. In either case the total load shall be computed as specified in "Test Requirements," page 69. When 16 loads are used, they shall be applied at 8.34, 15.22, 19.74, 23.36, 26.60, 29.86, 33.28, 36.90, 40.72, 44.76, 49.22, 54.08, 59.50, 65.80, 73.54 and 85.70 percent of the chord. When 8 loads are used they shall be so arranged as to give comparable results.

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

When the covering is to be attached separately to the two chords of the rib, the loading specified in paragraph 1 of this section should be modified so that approximately 75 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.

04.3130 The load shall be suitably distributed between upper and lower wing surfaces unless a more severe distribution is used.

04.3131 The effects of ailerons and high-lift devices shall be properly accounted for.

04.3132 Rib tests shall simulate conditions in the airplane with respect to torsional rigidity of spars, fixity conditions, lateral support and attachment to spars.

04.314 Covering. Proof of strength of fabric covering is not required when standard grades of cloth and methods of attaching and doping are employed *provided, however,* that the Administrator may require special tests when it appears necessary to account for the effects of unusually high design airspeeds or slipstream velocities, or similar factors. When metal covering is employed its ability to perform its structural function shall be demonstrated by tests of typical panels or by other means acceptable to the Administrator. In particular, compliance with § 04.201 requires demonstration of the behavior of the covering under load in order to determine the effects of temporary deformations (wrinkles).

04.315 Nonparallel wires. When two or more wires are attached to a common point on the wing, but are not parallel, proper allowance for redundancies and the effects of rigging shall be made.

04.32 PROOF OF TAIL AND CONTROL SURFACES.

Structural analyses of tail and control surfaces will be accepted as complete proof of compliance with *ultimate* load requirements only when the structure conforms with conventional types for which reliable analytical methods are available. Proof tests as defined in § 04.127 are required to prove compliance with *yield* load requirements.

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

04.320 Control surface tests shall include the horn or fitting to which the control system is attached.

04.321 In the analysis of control surfaces proper allowance shall be made for rigging loads in brace wires in cases where the counter wires do not go slack before the *ultimate* load is reached.

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

04.322 Analyses or individual load tests shall be conducted to demonstrate compliance with the multiplying factor of safety requirements outlined in § 04.27 for control surface hinges and brace wires.

04.323 Vibration tests. The natural frequencies of vibration of the wings, fuselage, and control surfaces shall be within such ranges of values as are satisfactory for the prevention of flutter. Compliance with this requirement shall be demonstrated by vibration tests or other methods acceptable to the Administrator.

The required vibration tests may be made by shaking the various units of the airplane by means of an unbalanced rotating weight driven through a flexible shaft at speeds which can be controlled and measured, or by other acceptable methods. These tests should be made on a complete airplane. The frequencies obtained for the various units should be entered in Form

Form ACA-719

TABLE XVII

DATE OF TEST: ________________ EQUIPMENT: ___________

USE INK OR HEAVY PENCIL LINE AND ENTER DATA IN LEGIBLE FORM

FLUTTER CONTROL DATA (k)

INDICATED DIVE SPEEDS
DESIGN: ..... mph
TESTED: ..... mph

MODEL: _________________

TEST ENGINEER: _______________

SERIAL NO.: ____________

Surface			Actual wt. of surface and balance lbs.	Semi-span, length or height ft.	Chord at ¾ semi-span ft.	Area sq. ft. (a)	Balance area (b)	Cable tension & temp. lbs. & degrees F	Cable size or type of control system	Elastic axis percent chord	C G Surface & balance [1]	Static unbalance [2]	K State axis (h) ⊥	K State axis (h)	K/I State axis ⊥	K/I			Vibration test data: Frequencies: Structure: Bending C. P. M. 1	Torsion C. P. M. 2	Control system: Symmetrical C. P. M. (j) 3	Unsym. C.P.M. 4		Size & weight of exciter	Sketch no. (g)	Remarks	Frequency ratios (j)
A Aileron	Left				X					X									X								
	Right				X					X									X								
B Wing	Left		(c)				X	X	X		(i)	X	X	X	X	X			(e)	(f)	X	X					
	Right						X	X	X			X	X	X	X	X					X	X					
C Rudder					X					X									X								
D Fin			X		X		X	X	X	X	X	X	X	X	X	X					X	X					
E Fuselage			X	X	X	X	X	X	X	X	X	X	X	X	X	X			V / S		X	X					
F Elevator	Left				X					X									X								
	Right				X					X									X								
G Stabilizer			X		X		X	X	X	X	X	X	X	X	X	X					R	X					
H Tab-aileron	Trim servo				X					X			(d)						X	X							
I Tab-rudder	Trim servo				X					X			(d)						X	X							
J Tab-elevator	Trim servo				X					X									X	X							
K Flap					X					X									X	X							

Frequency ratios (j):

- $\frac{\text{Rudder}}{\text{Fuselage Torsion}} = \frac{C3}{E2} = \{\quad\} =$
- $\frac{\text{Rudder}}{\text{Fuselage Side Bending}} = \frac{C3}{E1_S} = \{\quad\} =$
- $\frac{\text{Rudder}}{\text{Fin Bending}} = \frac{C3}{D1} = \{\quad\} =$
- $\frac{\text{*Rudders}}{\text{Stab. Bending}} = \frac{C3}{G1} = \{\quad\} =$
- $\frac{\text{*Rudders}}{\text{Stab. Rock. Attach.}} = \frac{C4}{G3_R} = \{\quad\}$
- $\frac{\text{*Rudders}}{\text{Fin Bending}} = \frac{C4}{D1} = \{\quad\} =$
- $\frac{\text{Elevator Sym.}}{\text{Fuselage Vertical Bend.}} = \frac{F3}{E1_V} = \{\quad\} =$
- $\frac{\text{Elevator Sym.}}{\text{Stabilizer Bending}} = \frac{F3}{G1} = \{\quad\} =$
- $\frac{\text{Elevator Unsym.}}{\text{Fuselage Torsion}} = \frac{F4}{E2} = \{\quad\} =$
- $\frac{\text{Ailerons}}{\text{Wing Bending}} = \frac{A3}{B1} = \{\quad\} =$
- $\frac{\text{Ailerons}}{\text{Wing Torsion}} = \frac{A3}{B2} = \{\quad\} =$
- $\frac{\text{Wing Bending}}{\text{Wing Torsion}} = \frac{B1}{B2} = \{\quad\} =$
- $\frac{\text{Stabilizer Bending}}{\text{Stabilizer Torsion}} = \frac{G1}{G2} = \{\quad\} =$

NOTES:

(a) Area of: (1) Aileron, (1) Rudder, (1) Flap, (½) Total Wing (Including Aileron and Flaps), (½) Total Stabilizer, (½) Total Elevator.
(b) All area forward of hinge line.
(c) Estimated unit weight at ¾ semi-span location.
(d) Not required if irreversible mechanism is used and tab frequency is greater than 1500 c. p. m.
(e) Include secondary mode for large aircraft, also unsymmetrical modes.
(f) For bi-planes include cellule torsion frequency.
(g) Show nodal lines, amplitudes, and vibration exciter location.
(h) Use axes given in CAM 04, or base upon vibration tests.
(i) *CG* location at ¾ semi-span location. In percent of chord.
(j) Symmetrical—Both Ailerons down (or up).
—Rudder—Twin
—Rudder Trailing Edges both in (or both out)
—Elevators—Both halves down (or up).
(k) See CAM 04 for details.
(j) In coded fractions shown in Frequency Ratio column, letters refer to SURFACE line, numbers refer to Frequencies column, thus $\frac{C3}{E2}$ is $\frac{\text{rudder symmetrical (control system).}}{\text{Fuselage torsion.}}$

V=Vertical Bending.
S=Side Bending.
R=Stab. Rocking About Attachment.
[1] Fwd. (−) or Aft (+) from centerline hinge, inches.
[2] (−) Overbalance (+) Underbalance with respect to centerline hinge in.-lbs.
*Only for rudders not in the plane of symmetry.

ACA-719, "Flutter Control Data." (A reproduction of this form to approximately one-half scale is shown as Table XVII.) Copies of this form may be obtained from the offices mentioned below.

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

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

Location of Vibrator on the Structure.—*a.* The proper location of the vibrator on the structure is important in obtaining satisfactory results. Suggested vibrator locations for exciting various modes of vibration are given in Table XVIII.

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

c. In general, the following points should be considered in the attachment of any type of vibrator to a structure. (1) The location is of primary importance and should be at a point of large deflection. See Table XVIII. (2) The vibrator should be so mounted that its line of force will be in the most advantageous direction to excite the vibration mode desired. (3) It is desirable to attach the vibrator to a part of the structure that is fairly rigid such as the wing spar, control surface ribs, etc. (4) The local structure to which the vibrator is attached should have adequate strength for the loads imposed by the vibrator.

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

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

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

TABLE XVIII.—Vibration Modes and Vibrator Location

(In general only a fraction of the modes listed will be applicable to any one airplane)

Item	Surface	Mode	Description of mode and suggested vibrator location
1	Rudder (single)	(As a unit)	Rudder swinging back and forth, under the spring action of the control cables. (Vibrator aft of horn on rudder T. E.)
2	Rudder (single)	Torsion	Torsional vibration of the rudder, under the spring action of the torque tube (Nodal line extends from trailing edge to torque tube.) (Vibrator near upper end on rudder T. E.)
3	Rudders (twin tail)	Sym	Same as item 1 above, except both rudders swinging inward (or outward) at same time. (Vibrator aft of horn on rudder T. E.)
4	Rudders (twin tail)	Unsym	Same as item 1 above, except both rudders moving to right (or left) together. (Vibrator aft of horn on rudder T. E.)
5	Rudders (twin tail)	Torsion	Same as item 2 above.
6	Elevator	Sym	Both elevators swinging up or down together under the spring action of the control cables (or push pull tubes). (Vibrator at inner end of one elevator T. E. or aft of horn on single elevator.)
7	Elevator	Unsym	One elevator (or ½ of single elevator) moving up, other down at same time under spring action of torque tube. (Nodal line in plane of surface, extending aft from point near center of elevator spar.) (Vibrator at center of semi-span near T. E.)
8	Aileron	Sym	Each aileron as a unit swinging up and down together, under the spring action of the control cables (or push pull tubes). (Vibrator between center and inner end at aileron T. E.)
9	Aileron	Unsym	One aileron as a unit moving up, other down at same time, under spring action of control cables. (Vibrator aft of horn at aileron T. E.)
10	Aileron	Torsion	Torsional vibration of aileron, under the spring action of the torque tube. (Nodal line extends from trailing edge to torque tube.) (Vibrator near outer end at aileron T. E.)
11	Wing	Sym. bending	Wing bending as a beam supported at its midpoint (fuselage), or as a braced beam supported at the brace points. (Vibrator near wing tip approximately on elastic axis of wing.)
12	Wing	Unsym. bending	Same as item 11 above, except that right and left halves of the wing move in opposite directions at the same time. Usually only important in large multi-engine aircraft. (Vibrator near wing tip approximately on elastic axis of wing.)
13	Wing	Sym. torsion	Torsional vibration of the wing about a spanwise axis. Right and left halves of the wing move in same direction about this axis at the same time. (Vibrator near wing tip, forward or aft of elastic axis of wing.)
14	Wing	Unsym. torsion	Same as 13 above, except right and left halves of the wing move in opposite directions about a spanwise axis at the same time. (Vibrator near wing tip, forward or aft of elastic axis of wing.)
15	Wing (biplane only).	Cellule sym. torsion.	Same as item 13 above, except center of rotation may be located between upper and lower wings. (Vibrator acting in chordwise direction at upper wing outer rear interplane strut attachment.)
16	Wing (biplane only).	Cellule unsym. torsion.	Same as item 14 above, except center of rotation may be located between upper and lower wings. (Vibrator acting in chordwise direction at upper wing, outer rear interplane strut attachment.)
17	Fuselage	Torsion	Whole tail unit and fuselage vibrating torsionally about longitudinal axis. (See item 23 also.) (Vibrator on fin or stabilizer—on hinge line, at tip of surface.)
18	Fuselage	Side bending	Whole tail unit vibrating in side direction about vertical axis through fuselage forward of tail unit. Usually important only on large airplanes. (Vibrator at tail end of fuselage.)
19	Fuselage	Vertical bending	Same as item 18 above, except vibrating in vertical direction about horizontal axis through fuselage. Only important in very large aircraft having fuselage cutouts in tops and bottom sides. (Vibrator at tail end of fuselage.)
20	Stabilizer	Sym. bending	Stabilizer bending as a beam supported at its midpoint (fuselage). Usually only important in cantilever tail surface designs. If wire or strut braced with small tip overhang, the mode of vibration may not be such as to permit interaction with the elevator. (Vibrator near outboard end of stabilizer.)
21	Stabilizer	Sym. torsion	Torsional vibration of the stabilizer. Similar to item 13, for the wing. Usually only important for cantilever and twin tail aircraft. (Vibrator near outer end of stabilizer at L. E.)
22	Stabilizer	Unsym. torsion	Torsional vibration of stabilizer. Similar to item 14 for the wing. Nodal line at ℄ of stabilizer (fuselage). (Vibrator at leading edge of stabilizer or at trailing edge of elevator. Elevator clamped to stabilizer.)
23	Stabilizer	Rocking about its fuselage attachments.	See item 17 also. Stabilizer as a unit rocking about its fuselage attachments. Usually only important for twin tail aircraft. (Vibrator on fin or outboard end of stabilizer on twin tail aircraft, or near outboard end of stabilizer—elevator hinge line on single tail aircraft.)
24	Fin (single tail)	Bending	Fin bending as a beam fixed at one end. Usually only important in cantilever tail surface designs. (Vibrator near upper end of fin near rudder hinge line.)
25	Fin (single tail)	Torsion	Torsional vibration of the fin. Similar to item 21 for the stabilizer. (Vibrator on fin L. E. near upper end.)
26	Fin (twin tail only).	Bending (sym. with respect to attaching fin to stabilizer).	Fin bending as a beam fixed at its attachment to stabilizer. (Vibrator near upper, or lower, end of fin near elastic axis of fin). (Upper and lower portions may have different frequencies.)
27	Fin (twin tail only).	Bending (unsym. with respect to attachment of fin to stabilizer).	Fin bending as a beam fixed at its attachment to stabilizer.
28	Flap	Torsion	Torsional vibration of outboard end of flap. Similar to item 10 above. Assuming irreversible control arm used. (Vibrator near outboard end of flap on T. E.) Only important when flap extends outboard on wing beyond 50% semi-span location.
29	Tabs (rudder aileron elevator).		Note: Effect of vibrator weight on tab frequency should be investigated when vibrator is attached directly to tab.
30	Trim or balance tabs.	Torsion	Torsional vibration of the tab under the spring action of the torque tube. Node on T. E. (Vibrator on tab near one end.)
31	Servo tabs	Sym	Tab as a unit swinging back and forth under the spring action of the control cables. (Vibrator on adjacent tab supporting structure, or on tab itself.)
32	Servo tabs	Torsion	Same as item 10 above. (Vibrator on tab near one end.)
33	Balance weights for movable surfaces—mounted on long supports.	Bending of support.	Support bending as a beam (fixed at one end) in various planes depending on the rigidity. Most important directions are vertical and sideways with relation to the airplane centerline. (Vibrator near balance weight support.)

The phase relationship of vibrating parts may be determined by the method shown in figure 32 as applied to the particular case of the elevators. The metal plates A and B, attached to the trailing edges of the elevators and interconnected with a wire, are necessary only in the case of fabric covered surfaces or surfaces which have a poor electrical interconnection. When the parts are vibrating the phase relationship may be determined by manually holding the leads C and D close to the surfaces so that intermittent contact is made during each cycle. If the light flashes or clicks are heard in the headphones at regular intervals (with the contacts in the same side; i. e., upper or lower), the surfaces are vibrating in phase, whereas, if the light does not flash, or no click is heard in the headphones, the surfaces are out of phase. This should be verified by reversing one contact, for example, putting contact D on the upper side.

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

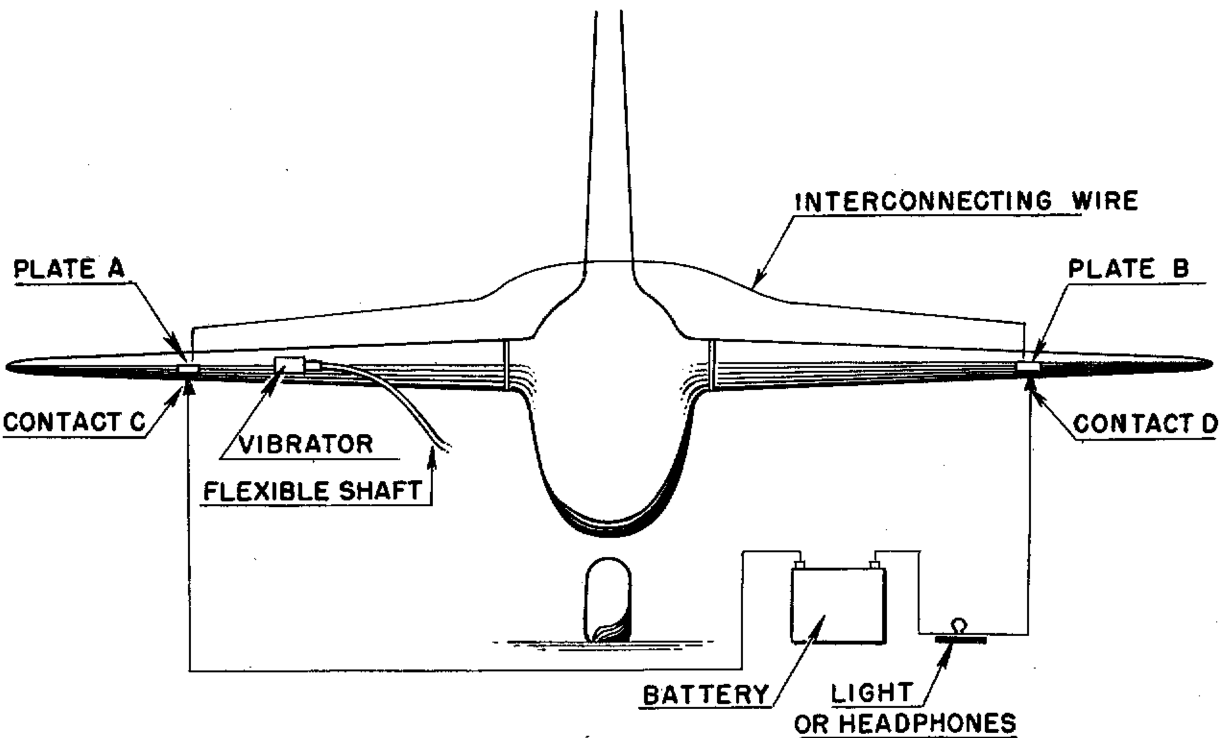


Figure 32.—Test set-up for determination of phase relationship.

A number of frequency trend curves are shown in figure 33. These will be expanded as more data become available. The wing and stabilizer data shown on this figure were obtained on unbraced surfaces. It should be appreciated that these trends are approximate and can serve as a rough guide only. Many factors, such as the type of construction involved, etc., will have a marked influence on the actual values which will be obtained for any particular design.

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

04.33 PROOF OF CONTROL SYSTEMS.

Structural analyses of control systems will be accepted as complete proof of compliance with *ultimate* load requirements only when the structure conforms with conventional types for which reliable analytical methods are available. Proof tests as defined in § 04.127 are required to prove compliance with *yield* load requirements.

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

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

04.330 In control system tests, the direction of test loads shall be such as to produce the most severe loading of the control system structure. The tests shall include all fittings, pulleys, and brackets used to attach the control system to the primary structure.

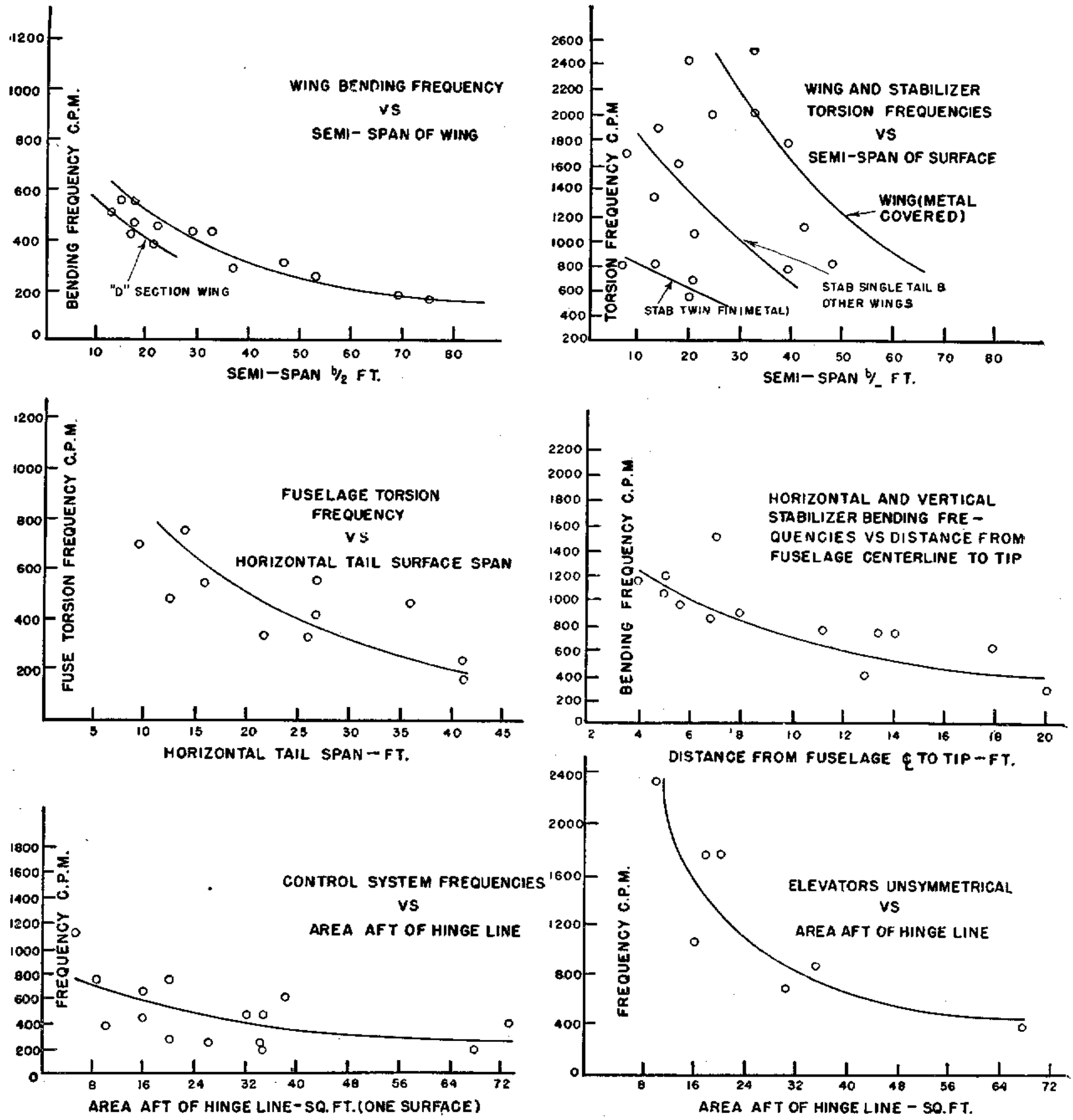


Figure 33.—Natural frequency trends.

04.331 Operation test. An operation test shall be conducted by operating the controls from the pilot's compartment with the entire system so loaded as to correspond to the minimum *limit* control force specified in item 3 of Table IX for the control system in question. In this test there shall be no jamming, excessive friction, or excessive deflection.

The system under the proof and operating tests should show no signs of excessive deflection or any permanent set of any connection, bracket, attachment, etc.

In order to insure that the surfaces will retain most of their effectiveness in flight under limit maneuvering loads the following check is recommended: The control surfaces and systems should be subjected to the limit loads as computed for the maneuvering flight conditions neglecting the minimum stick forces. If tabs are used, their effect shall be taken into account except that the *minimum* stick force limitations may be neglected. The angular deflections of the movable surfaces due to the above loads should be measured, and it is recommended that the average deflection for each surface not exceed one-half of the total one-way movement of the surface.

04.332 Analyses or individual load tests shall be conducted to demonstrate compliance with the multiplying factor of safety requirements specified in § 04.27 for control system joints subjected to angular motion.

04.34 PROOF OF LANDING GEAR.

Structural analyses of landing gear will be accepted as complete proof of compliance with load requirements only when the structure conforms with conventional types for which reliable analytical methods are available. Analyses may be used to demonstrate compliance with the energy absorption requirements in certain cases. When such analyses are not applicable, dynamic tests shall be conducted to demonstrate compliance with energy absorption requirements.

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

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

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

b. Tension loads are positive, compression loads negative.

c. Moments are represented by vectors according to the "right hand" rule.

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

e. In writing the equations of equilibrium, all forces are initially assumed to be tension, i. e., positive. The true nature of the forces will be indicated by the sign of the vector obtained in the final solution.

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

04.340 Energy absorption tests. When tests for energy absorption are required they shall be so conducted as to simulate the landing conditions for which energy absorption requirements are specified in § 04.440, and test data shall be obtained from which the maximum acceleration developed at the center of gravity of the airplane can be determined. When drop tests of wheels, tires and shock absorbers are conducted in a combination differing from that employed on the airplane, proper allowance and corrections shall be made for the errors thus introduced.

A. General

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

2. Many cases arise which involve approval of a higher gross weight, the necessary greater height of drop, and/or the use of different tires from those used in the original drop test. In some

such cases it may be possible to demonstrate compliance with the requirements without an additional drop test. In general, however, time and expense will be saved if such changes are anticipated and substantiated at the time of the original drop test.

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

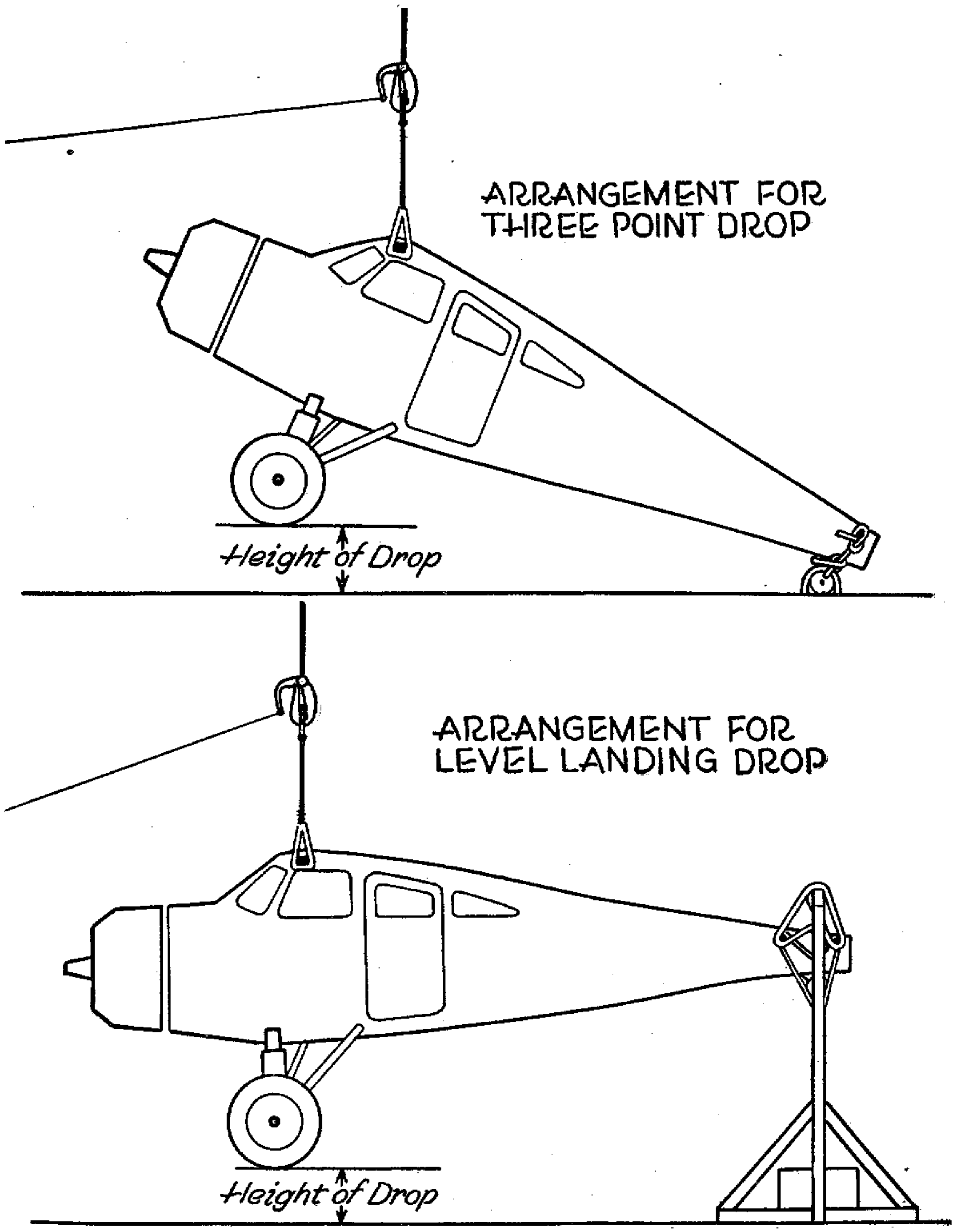


Figure 34.—Set-up for landing gear drop test.

B. Main Gear Tests—First Method

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

2. For the three-point landing test the rear end of the fuselage is held in place on the floor as shown in figure 34. For the level landing test the rear end of the fuselage is raised until the center of gravity of the loaded airplane is vertically above the wheel axles, or until the fuselage is inclined at a nose-down angle of 14°, whichever is reached first. The rear end of the fuselage is then held in this position, as shown in figure 34. Care should be taken, particularly in the level landing drop test, to restrain the rear end of the fuselage from rising as a result of the impact. When the airplane is in position for the drop it is advisable to place sand bags under the structure near the *CG* to minimize the damage in case of failure.

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

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

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

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

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

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

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

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

6. The final test should be witnessed by a CAA representative. The manufacturer's report should include, in addition to other data (see page 3), the accelerometer records or exact copies of them, with the magnitude of the maximum acceleration determined and marked thereon. A record of the maximum tire deflection should also be given.

C. Main Gear Tests—Second Method

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

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

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

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

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

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

a. Determine the value of K (ratio of the static load on the strut to the static load on the tire) for the critical condition being simulated in the test (See 1*b* above and *e* below for considerations involved).

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

c. Replace the original tire with a tire having a load deflection curve each ordinate (load) of which is K times the original value and each abscissa (deflection) of which is approximately $1/K$ times the original value, the original values being those for the tire actually used. In addition the maximum deflection of the test tire should be limited to $1/K$ times the maximum deflection of the original tire. It may be possible to obtain the above characteristics by changing the inflation pressure of the original tire and by using stops.

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

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

D. Tests of Tail Wheels and Tail Skids

1. Tests for the energy absorption capacity of the tail wheel assembly may be made in a manner similar to that used for testing a complete main gear assembly (see B "Main Gear Tests", page 79), except that the tests need be made only for the three-point condition. The test load may be obtained by loading the fuselage or by concentrating the required mass over the tail wheel.

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

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

E. Tests of Nose-Wheel Type Gear

1. In general, the tests of main wheel and nose wheel installations may be made in accordance with the methods outlined in A to C. The tests of each installation should be made for the most critical (most unfavorable with respect to shock absorption) of the conditions outlined in 2*a* through 2*e*, pages 47 to 48. Each of these conditions is assumed to be produced by the free drop from the height specified in 04.2411. In determining the critical conditions, consideration should be given to the value of the component of wheel travel (relative to the airplane) in the direction of the resultant external force and also to the magnitude of this force. In general, the higher the force and the smaller the travel, the more critical the condition. In cases where question arises as to the applicability of the design conditions used it is advisable to conduct actual landing and taxiing tests with one or more accelerometers installed in the airplane.

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

F. Tests at Provisional Weight

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

04.35 PROOF OF HULLS AND FLOATS.

Structural analyses of hulls and auxiliary floats will be accepted as complete proof of compliance with load requirements only when the structure conforms with conventional types for which reliable analytical methods are available. The strength of the structure as a whole and its ability to distribute water loads from the bottom plating into the main structural members shall be demonstrated. See Part 15 for the requirements for main floats.

04.36 PROOF OF FUSELAGES AND ENGINE MOUNTS.

Structural analyses of fuselages and engine mounts will be accepted as complete proof of compliance with load requirements only when the structure conforms with conventional types for which reliable analytical methods are available.

A. General

In addition to determining the loads in the main structural members of a fuselage, the local loads imposed by the internal weights which they support should not be overlooked. This applies particularly to members which serve both as a critical portion of the primary structure and as a means of support for some item of appreciable weight. Also, whenever critical, control system loads which occur in the specific flight or landing conditions should be combined with the primary loads. The combined stresses should be determined in such cases.

B. Stress Analysis Procedure

1. *Weight distribution.*—All major items of weight affecting the fuselage should be so distri buted to convenient panel points that the true center of gravity of the fuselage and its content is maintained. A suitable vertical di vision of loads should be included. The following rule should be followed in computing the panel point loads for conventional airplanes:

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

b. The weight of an item to the rear of the tail post or forward of the front structure should be represented in the table by a load and a horizontal couple at the tail post or front frame, as the case may be.

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

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

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

2. *Balancing (symmetrical conditions).*—Methods of balancing the airplane are discussed under "Balancing Loads," page 37. It will, in general, be satisfactory to apply directly the balancing loads found in the various flight conditions. The acceleration factor applied to each item of mass in the fuselage will be the net acceleration factor as determined from the balancing computations. The basic inertia force on any item will be parallel to the resultant external applied force and will not, in general, be perpendicular to the thrust line. In certain cases the chord components of the inertia forces (i.e., the components along the thrust line or fuselage centerline) can conveniently be combined into a single force applied at the nose of the fuselage. This procedure permits the use of vertical inertia loads but it should not be used unless it is obviously conservative for the critical fuselage members.

3. *Balancing (unsymmetrical conditions).*—In any condition involving angular acceleration about a given axis, the inertia force applied to the structure by any item of weight is proportional to the mass or weight of the item and to its distance from the axis of rotation. Each angular inertia force will act in a direction perpendicular to the radius line between the item and the axis of rotation. In order to facilitate the analysis of a condition involving both linear and angular acceleration, the loads produced by the linear acceleration should be determined separately from those produced by angular acceleration. When unbalanced external loads are applied this involves the determination of the magnitude of the net resultant external load and its moment arm about the proper axis through the *CG* of the airplane. It will usually be acceptable, in analyses of this nature, to represent the weights of major items such as wing panels, nacelles, etc., by assumed concentrated masses at the centers of gravity of the respective items. Figure 35 illustrates approximate methods by which the fuselage can be balanced for a typical unsymmetrical landing condition (one-wheel landing).

a. Figure 35a shows a level landing condition in which the resultant load does not pass through the center of gravity. In such a case it will generally be acceptable to apply a balancing couple composed of a downward force acting near the nose of the fuselage and an equal upward force acting at the same distance to the rear of the center of gravity. These arbitrary forces can be considered as approximately representing angular inertia forces and they may be divided between the nearest panel points, if desired.

b. Figure 35b indicates an acceptable method of balancing externally applied rolling moments about the longitudinal axis. The forces resisting angular acceleration are assumed to be applied by the wing. The arbitrary location shown is based on the fact that the effectiveness of any item is proportional to its distance from the center of gravity. The balancing loads may be assumed to be vertical, although they actually act normal to a radius line through the center of gravity of the airplane. If nacelles or similar items of large weight are attached to the wing, the balancing couples can be divided between the nacelles and wing panels in proportion to their effectiveness. This type of balancing applies also to side landing conditions, including those for seaplanes.

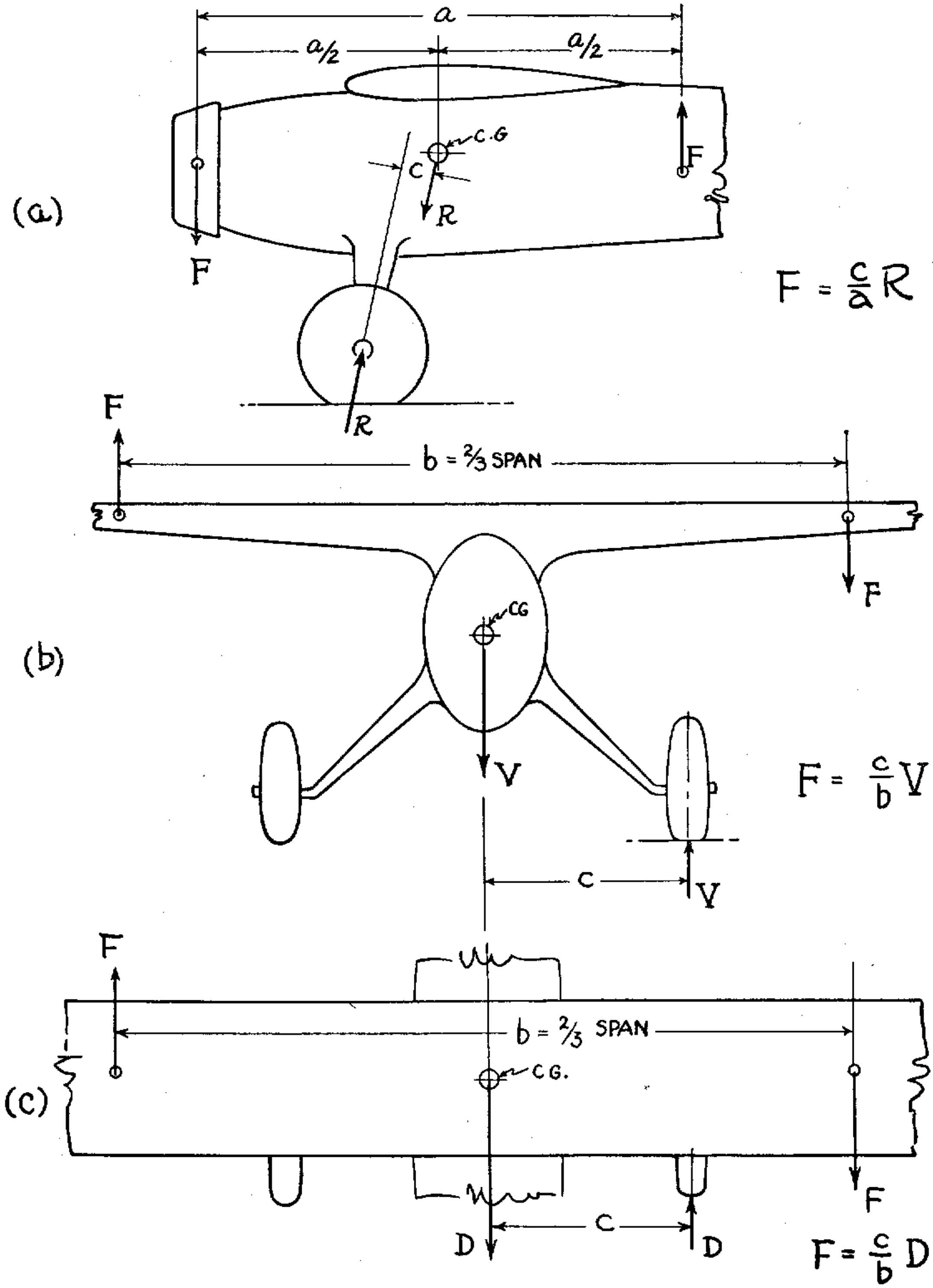


Figure 35.—Methods of balancing fuselage for unsymmetrical loads.

c. Figure 35c shows an approximate method for balancing a moment about a vertical axis This condition exists in a one-wheel landing. It is conservative (for the wing attachment members) to assume that the balancing couple is supplied entirely by the wing. The magnitude of the unbalanced moment about a vertical axis is, however, relatively small in the design conditions required. In order to secure ample rigidity against loads tending to twist the wing in its own plane, it may sometimes appear advisable to check the wing attachment members or cabane for a greater unbalanced drag load acting at one wheel, or for a side load acting at the tail.

d. It should be noted that the balancing couples shown on figure 35 will act *in addition* to the inertia loads due to linear acceleration. For instance, in figure 35b the load V shown as a reaction at the CG actually represents the inertia loads of all the components of the airplane. Those due to the wing weight will act uniformly on each wing panel and will be added arithmetically to the forces representing the angular inertia effects. This applies also to the other cases illustrated.

C. Special analysis methods

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

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

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

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

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

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

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

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

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

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

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

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

D. Analysis of Stressed-Skin Fuselages.

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

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

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

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

d. True location of neutral axis and stress distribution.

e. Applied and allowable loads for rings and bulkheads.

Unless a fuselage of this nature conforms closely to a previously constructed type, the strength of which has been determined by test, a stress analysis is not considered as a sufficiently accurate means of determining its strength. In all cases, the stress analysis should be supplemented by pertinent test data. Whenever possible it is desirable to test the entire fuselage for bending and torsion, but tests of certain component parts may be acceptable in conjunction with a stress analysis.

04.360 The end fixity coefficient used in determining critical column loads shall in no case exceed 2.0. A value of 1.0 shall be used for all members in the engine mount. In doubtful cases, tests are required to substantiate the degree of restraint assumed.

04.361 Baggage compartments. The ability of baggage compartments to sustain the maximum authorized baggage loads under all required flight and landing conditions shall be demonstrated.

04.37 PROOF OF FITTINGS AND PARTS.

Proof of strength of all fittings and joints of the primary structure is required. Where applicable, structural analysis methods may be used. When such methods are inadequate, a load test is required. Compliance with the multiplying factor of safety requirements for fittings (§ 04.27) shall be demonstrated.

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

The additional ultimate factor of safety of 1.20 for fittings (Table XII) is to account for various factors, such as stress concentration, eccentricity, uneven load distribution, and similar features which tend to increase the probability of failure of a fitting. As noted in the Table, this factor may be covered by several other factors so that when the ultimate factor of safety for any portion of the structure equals or exceeds 1.80 the fittings included in this portion are not subject to an increase in factor above the value used for the primary members.

04.370 Since the system of forces which designs a fitting does not necessarily include the forces which design the attaching members, all the forces acting in all the specified conditions shall be considered for every fitting. The strength of each part of a built-up fitting shall be investigated and proper allowance shall be made for the effects of eccentric loading when initially present or when introduced by deflection of the structure under load.

04.371 Bolts. The allowable bearing load assumed for the threaded portion of a bolt shall not exceed 25 percent of the rated shear strength of the bolt.

04.4 DETAIL DESIGN AND CONSTRUCTION

04.40 GENERAL.

The primary structure and all mechanisms essential to the safe operation of the airplane shall not incorporate design details which experience has shown to be unreliable or otherwise unsatisfactory. The suitability of all design details shall be established to the satisfaction of the Administrator. Certain design features which have been found to be essential to the airworthiness of an airplane are hereinafter specified and shall be observed.

04.400 Materials and workmanship. The primary structure shall be made from materials which experience or conclusive tests have proved to be uniform in quality and strength and to be otherwise suitable for airplane construction. Workmanship shall be of sufficiently high grade as to insure proper continued functioning of all parts.

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

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

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

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

04.401 Fabrication methods. The methods of fabrication employed in constructing the primary structure shall be such as to produce a uniformly sound structure which shall also be reliable with respect to maintenance of the original strength under reasonable service conditions.

04.4010 Gluing. Gluing may be used except in cases where inferior joints might result or where proper protection from moisture cannot be shown.

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

04.4011 Torch welding. Torch welding of primary structural parts may be used only for ferrous materials and for such other materials shown to be suitable therefor.

04.4012 Electric welding. Electric arc, spot, or seam welding may be used in the primary structure when specifically approved by the Administrator for the application involved. Requests for approval of the use of electric welding shall be accompanied by information as to the extent to which such welding is to be used, drawings of the parts involved, apparatus employed, general methods of control and inspection, and references to test data substantiating the strength and suitability of the welds obtained.

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

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

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

TABLE XIX.—Selection and Properties of Aircraft Wood

Species of wood	Strength properties as compared to Spruce	Permissible range of moisture content percent	Specific gravity (when oven dry)		Pounds per cu. ft. at 15 percent moisture content	Maximum permissible grain deviation (slope of grain)	Minimum number of annual rings per inch	Remarks
			Average	Minimum				
1	2	3	4	5	6	7	8	9
Spruce (Picea) Sitka (P. Sitchensis) Red (P. Rubra) White (P. Glauca).	100%	8–12	0.40	.36 (See note 1)	27	1–15 (See note 1)	6	Excellent for all uses. Considered as standard for this table.
Douglas Fir (Pseudotsuga Taxifolia).	Exceed Spruce	8–12	.51	.45 (See remarks)	34	1–20	8	May be used as substitute for Spruce in same sizes or in slightly reduced sizes providing reductions are substantiated. If used as direct substitute for Spruce in same sizes, a reduction in the minimum specific gravity (column 5) to .38 is permissible. Difficult to work with hand tools. Some tendency to split and splinter during fabrication and considerable more care in manufacture is necessary. Large solid pieces should be avoided due to inspection difficulties. Gluing satisfactory.
Noble Fir (Abies Nobiles).	Slightly exceed Spruce except 8 percent deficient in shear.	8–12	.40	.36	27	1–20		Satisfactory characteristics with respect to workability, warping, and splitting. May be used as direct substitute for Spruce in same sizes providing shear does not become critical. Hardness somewhat less than Spruce. Gluing satisfactory.
Western Hemlock (Tsuga Heterophylla).	Slightly exceed Spruce	8–12	.44	.40	29	1–20	8	Less uniform in texture than Spruce. May be used as a direct substitute for Spruce. Upland growth superior to lowland growth. Gluing satisfactory.
Pine, Northern White (Pinus Strobus).	Properties between 85% and 96% those of Spruce.	8–12	.38	.34	26	1–20		Excellent working qualities and uniform in properties but somewhat low in hardness and shock-resisting capacity. Cannot be used as substitute for Spruce without increase in sizes to compensate for lesser strength. Gluing satisfactory.
White Cedar, Port Orford (Charaecyparis Lawsoniana).	Exceed Spruce	8–12	.44	.40	30	1–20	8	May be used as substitute for Spruce in same sizes or in slightly reduced sizes providing reductions are substantiated. Easy to work with hand tools. Gluing difficult but satisfactory joints can be obtained if suitable precautions are taken.
Poplar, Yellow (Liriodendrow Tulipifera).	Slightly less than Spruce except in compression (crushing) and shear.	8–12	.45	.38	28	1–20	8	Excellent working qualities. Should not be used as a direct substitute for Spruce without carefully accounting for slightly reduced strength properties. Somewhat low in shock-resisting capacity. Gluing satisfactory.

NOTES

1. *Grain Deviation.* It will be noted that various specifications for aircraft lumber call for a maximum permissible grain deviation of 1 inch in 20 inches, with the exception of Spruce where 1 in 15 is specified. When the Spruce lumber specification was originated, it was anticipated that the proportion of the material having a slope no greater than 1 to 20 would be sufficient to provide such a slope in highly stressed members and that material having a slope between 1 to 20 and 1 to 15 would be used in other parts. Spruce having a slope of 1 to 20 or less should, therefore, be used wherever possible, however if it becomes necessary to utilize Spruce having a grain slope between 1 to 20 and 1 to 15 in highly stressed members, only material having a specific gravity (when oven dry) of 0.40 or greater should be used.

2. *Drying Methods.* Woods used in aircraft construction may be seasoned by air-drying, kiln-drying, or a combination of both. Some artificial drying is usually necessary to reduce the moisture content to the 8% to 12% range of values specified in the table. Kiln-drying has the advantage that atmospheric conditions are controlled, permitting the drying of wood without development of checks and other defects. Such defects are much more prevalent in air-dried lumber. Kiln-drying schedules may be obtained from U. S. Army Specification No. 82-13, Kiln Drying Process for Aircraft Lumber; Navy Department General Specification for Inspection of Material, Appendix IV, Lumber and Timber; or Department of Agriculture Forest Products Laboratory Report No. 1360, Aircraft Kiln Schedule. Western Hemlock should be kiln dried in accordance with Table 1 of those references.

3. *Sawing Methods.* In all species, quarter-sawed lumber is definitely preferable to flat-grain stock for aircraft use. Quarter-sawed stock (also referred to as edge-grain, rift-grain and rift-sawn) is defined as stock in which the rings are at an angle of 45° to 90° to the surface, and flat-grain stock (also referred to as plain-sawed or slash-grain) is stock in which the rings are at an angle of 0° to 45° to the surface. Quarter-sawed stock has less tendency to check and is less subject to bowing flatwise and to cupping, since it tends to shrink or swell equally with changes in moisture content. Resistance to indentation from tightening fittings, etc., is more uniform over an edge-grain surface. Experience with Douglas Fir in aircraft construction indicates that it should be quarter-sawed in all cases.

4. *Defects Permitted.*

(a) *Cross grain.* Spiral grain, diagonal grain, or a combination of the two is acceptable providing the grain does not diverge from the longitudinal axis of the material more than specified in column 7. A check of all four faces of the board is necessary to determine the amount of divergence. The direction of free-flowing ink will frequently assist in determining grain direction.

(b) *Wavy, curly, and interlocked grain.* Acceptable if local irregularities do not exceed limitations specified for spiral and diagonal grain.

(c) *Hard knots.* Sound hard knots up to ⅜ inch in maximum diameter acceptable providing: (1) they are not in projecting portions of I-beams, along the edges of rectangular or beveled unrouted beams or along the edges of flanges of box beams (except in lowly stressed portions); (2) they do not cause grain divergence at the edges of the board or in the flanges of a beam more than specified in column 7; and (3) they are in the center third of the beam and are not closer than 20″ to another knot or other defect (pertains to ⅜″ knots—smaller knots may be proportionately closer). Knots greater than ¼ inch should be used with caution.

(d) *Pin knot clusters.* Small clusters acceptable providing they produce only a small effect on grain direction.

TABLE XIX.—*Continued*

(e) *Pitch pockets.* Acceptable in center portion of a beam providing they are at least 14 inches apart when they lie in the same growth ring and do not exceed 1½ inches length by ⅛ inch width by ⅛ inch depth and providing they are not along the projecting portions of I-beams, along the edges of rectangular or beveled unrouted beams or along the edges of the flanges of box beams.

(f) *Mineral streaks.* Acceptable providing careful inspection fails to reveal any decay.

5. *Defects Not Permitted.*

(a) *Cross grain.* Not acceptable unless within limitations noted in 4 (a).

(b) *Wavy, curley, and interlocked grain.* Not acceptable unless within limitations noted in 4 (b).

(c) *Hard knots.* Not acceptable unless within limitations noted in 4 (c).

(d) *Pin knot clusters.* Not acceptable if they produce large effect on grain direction.

(e) *Spike knots.* These are knots running completely through the depth of a beam perpendicular to the annual rings and appear most frequently in quarter-sawed lumber. Wood containing this defect should be rejected.

(f) *Pitch pockets.* Not acceptable unless within limitations noted in 4 (e).

(g) *Mineral streaks.* Not acceptable if accompanied by decay (see 4 (f)).

(h) *Checks, shakes, and splits.* Checks are longitudinal cracks extending, in general, across the annual rings. Shakes are longitudinal cracks usually between two annual rings. Splits are longitudinal cracks induced by artificially induced stress. Wood containing these defects should be rejected.

(i) *Compression wood.* This defect is very detrimental to strength and is difficult to readily recognize. It is characterized by high specific gravity, has the appearance of an excessive growth of summer wood, and in most species shows but little contrast in color between spring wood and summer wood. In doubtful cases the material should be rejected or samples should be subjected to a toughness machine test to establish the quality of the wood. All material containing compression wood should be rejected.

(j) *Compression failures.* This defect is caused from the wood being overstressed in compression due to natural forces during the growth of the tree, felling trees on rough or irregular ground or rough handling of logs or lumber. Compression failures are characterized by a buckling of the fibers that appears as streaks on the surface of the piece substantially at right angles to the grain, and vary from pronounced failures to very fine hair lines that require close inspection to detect. Wood containing obvious failures should be rejected. In doubtful cases the wood should be rejected or further inspections in the form of microscopic examination or toughness tests made, the latter means being the more reliable.

(k) *Decay.* All stains and discolorations should be examined carefully to determine whether or not they are harmless or preliminary or advanced decay. All pieces should be free from rot, dote, red heart, purple heart, and all other forms of decay.

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

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

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

f. Drawings of parts to be welded.

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

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

04.4013 Brazing and soldering. The use of brazing and soldering in joining parts of the primary structure is prohibited except that brazing may be used in special cases when the suitability of the method and application can be definitely established to the satisfaction of the Administrator.

04.4014 Protection. All members of the primary structure shall be suitably protected against deterioration or loss of strength in service due to corrosion, abrasion, vibration or other causes. This applies particularly to design details and small parts. In seaplanes special precautions shall be taken against corrosion from salt water, particularly where parts made from different metals are in close proximity. All exposed wood structural members shall be given at least two protective coatings of varnish or approved equivalent. Built-up box spars and similar structures shall be protected on the interior by at least one coat of varnish or approved equivalent and adequate provisions for drainage shall be made. Due care shall be taken to prevent coating of the gluing surfaces.

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

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

04.4015 Inspection. Inspection openings of adequate size shall be provided for such vital parts of the aircraft as require periodic inspection.

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

04.402 Joints, fittings and connecting parts. In each joint of the primary structure the design details shall be such as to minimize the possibility of loosening of the joint in service, progressive failure due to stress concentration, and damage caused by normal servicing and field operations. (See § 04.271 for multiplying factors of safety required.)

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

a. Stress concentration, either from section changes or from welding or heat treating effects.

b. Adequate allowance for flexibility of parts being joined.

c. Specifying proper surface condition, i. e., a rough turning job on a highly stressed part invites cracking and failure.

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

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

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

04.4020 Bolts, pins and screws. All bolts and screws in the structure shall be of uniform material of high quality and of first-class workmanship. Machine screws shall not be used in the primary structure unless specifically approved for such use by the Administrator. The use of an approved locking device or method is required for all bolts, pins and screws.

Approved locking devices include cotter pins, safety wire, peening, and self-locking nuts listed on Civil Aeronautics Administration Product and Process Specification 4.[1] Dardelet Threaded parts are also approved subject to the following restrictions:

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

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

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

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

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

04.4021 Wood screws. The use of wood screws in the primary structure is prohibited except in special cases when the suitability of the particular application is proved to the satisfaction of the Administrator.

04.4022 Eyebolts. Special eyebolts and similar bolts shall have a fillet between the head and the shank of at least ¼ the diameter of the bolt when used in control surfaces or at other locations where they might be subjected to bending or vibration.

04.4023 Castings. Castings used in the primary structure shall incorporate the multiplying factor of safety specified in § 04.272 and shall be of such material and design as to insure the maximum degree of reliability and freedom from defects. The Administrator has the right to prohibit the use of castings where such use is deemed to be unairworthy.

Castings should be obtained from a reliable source with experience on similar type castings. Such castings should incorporate generous fillet radii, ample draft, and gradual changes of section.

[1] Copies of Product and Process Specification 4 may be obtained from the CAA Information and Statistics Service, A240, Department of Commerce, Washington 25, D. C.

Sound castings can only be secured by proper consideration of and allowance for the flow of molten metal in the mold. Casting drawings should be "load marked," i. e., the direction and approximate magnitude of the design loads should be shown. It is then possible for the foundry to cast the densest and soundest metal at the critical sections. Finished surfaces should end in radii at inside corners to prevent stress concentration. Some of the more important design and drafting considerations are given in Table XX. It should be emphasized that these are not given as requirements but merely as values and points found acceptable in general practice. Reference should be made to trade literature of the various metal and alloy producers for additional information.

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

TABLE XX.—Suggested Casting Practice

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

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

04.403 Tie-rods and wires. The minimum size of tie-rod which may be used in primary structure is No. 6–40. The corresponding minimum allowable size of single-strand hard wire is No. 13 (0.072-inch diameter).

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

04.4030 Wire terminals. The assumed terminal efficiency of single-strand hard wire shall not be greater than 85 percent.

04.4031 Wire anchorages. A fitting attached to a wire or cable up to and including the 3,400-pound size shall have at least the rated strength of the wire or cable, and the multiplying factor of safety for fittings (§ 04.271) is not required in such cases. In the case of fittings to which several tie-rods or wires are attached, this requirement applies separately to each portion of the fitting to which a tie-rod or wire is attached, but does not require simultaneous application of rated wire loads. The end connections of brace wires shall be such as to minimize restraints against bending or vibration.

04.4032 Counter wire sizes. (See also §§ 04.274 and 04.275.) In a wire-braced structure the wire sizes shall be such that any wire can be rigged to at least 10 percent of its rated strength without causing any other wire to be loaded to more than 20 percent of its rated strength. As used here "rated strength" refers to the wire proper, not the terminal.

04.404 General flutter prevention measures. When he deems it necessary in the interest of safety, the Administrator may require special provisions against flutter. For specific requirements see §§ 04.323, 04.413, 04.423, 04.424, 04.425, 04.426, 04.435, 04.436, and 04.707.

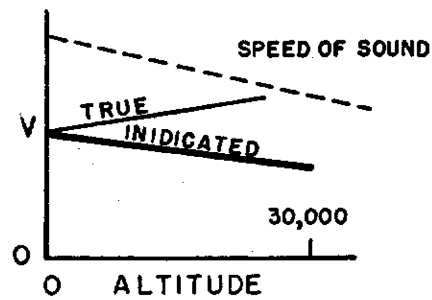
A. General

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

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

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

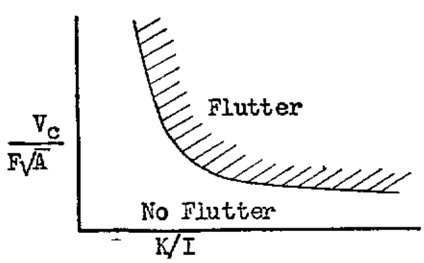
2. Flutter theory is in general based upon *true* velocity and sea level atmospheric conditions. For this as well as other reasons an ample margin, particularly on high performance aircraft, should be maintained between the *computed* flutter speed and the *actual* dive speed attained in testing. The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



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

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

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



where V_c = Flutter speed
F = Freq. fixed surface
A = Area movable surface
K/I = Balance coefficient of movable surface

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

$$V_c = \frac{FC}{K},$$

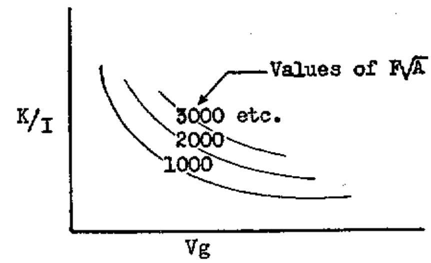
where F = Critical frequency

C = Wing chord

K = Reduced frequency coefficient (dependent on particular characteristics of the airplane)

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

5. By replotting the Air Corps data on the scales used in figure 37a, the following is obtained:



If the results of further study and experience warrant, suitable modifications to the $K/I = V_e$ relation shown on figures 37a and 37b will be made.

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

B. Rigidity

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

2. *Wings.*—The torsional rigidity of wings is highly important. This should be investigated by means of a wing torsion test (E, page 63) unless other adequate data are submitted. Figure 36, a development of an earlier curve of the same number, but based upon considerably more information including certain Navy data, indicates values of C_{TR} which have been found satisfactory for conventional designs. If the test is conducted on a fabric covered wing with taut fabric, an allowance of approximately 10 percent (dependent somewhat on the size of the wing) should be made for the effect of fabric aging. Since the actual torsional deflection of the wing will depend upon the moment coefficient of the airfoil employed, it is advisable to introduce the additional criterion that the maximum torsional deflection under the limit load critical for torsion not exceed 3°.

C. Mass Balancing

1. For methods of computing static and dynamic balance values see pages 102 to 106. (The weight and static hinge moment, or CG location, of finished movable surfaces should be checked to insure that the computed values are not unconservative.) See paragraph E "Detail Design", for notes on the installation of balance weights. Note that the specified dynamic balance coefficient values may in some cases be influenced by the frequency ratio (see Fig. 37c).

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

a. Ailerons.—When V_e is in excess of 150 mph the dynamic balance coefficient as computed

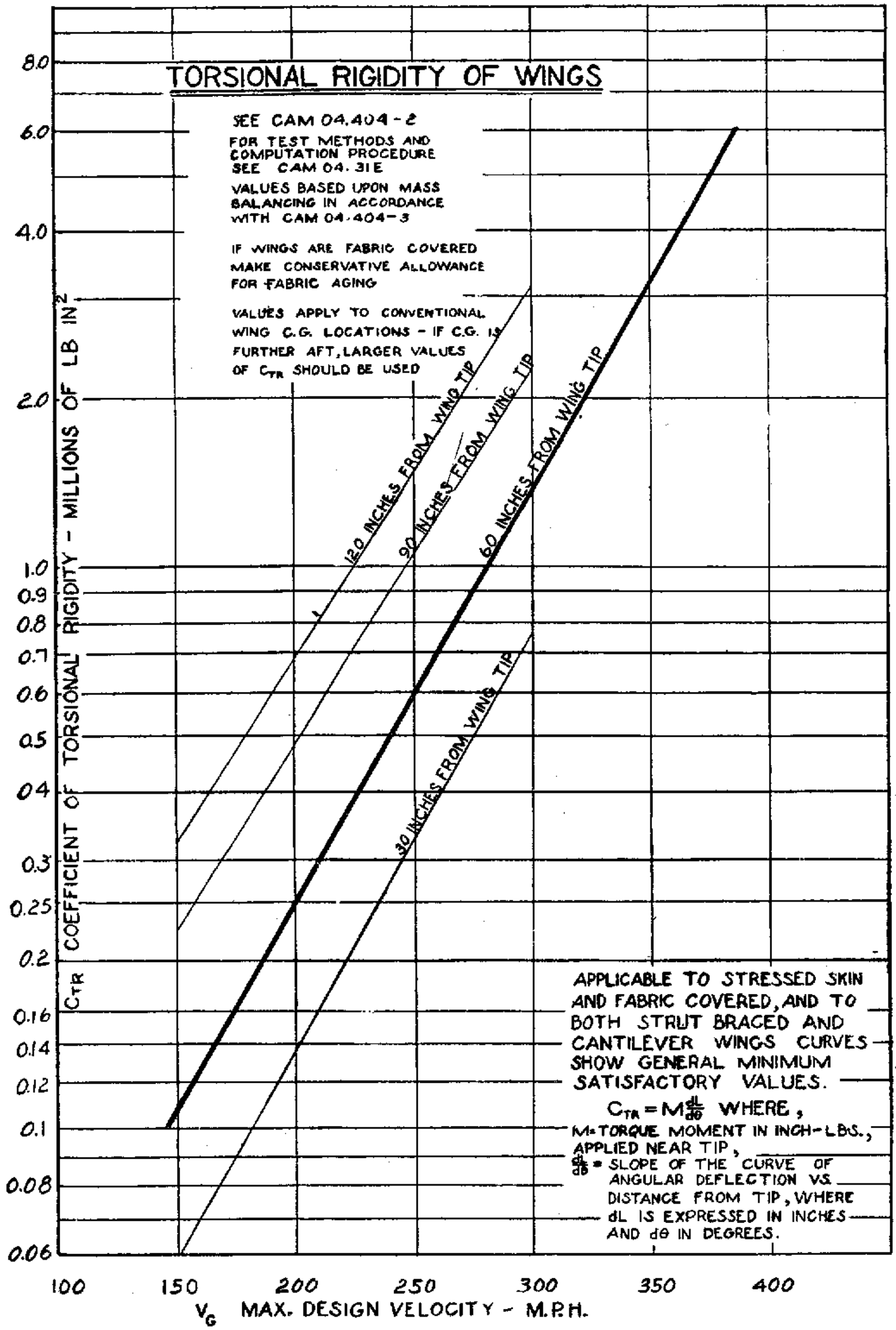


Figure 36.—Torsional rigidity of wings.

about the aileron hinge axis and the longitudinal axis of the airplane should not be greater than the following value

$$K/I = 1.6\ (3 - V_g/100) \qquad \text{(See Fig. 37a)}$$

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

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

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

$$K/I = 3.0 - V_g/250 \qquad \text{(See Fig. 37b)}$$

When the rudder(s) has (have) complete dynamic balance about a conservatively chosen axis, a special ruling may be obtained from the CAA regarding the elevator dynamic balance if the coefficient is greater than above specified. This ruling will be dependent on the general design of the entire tail unit.

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

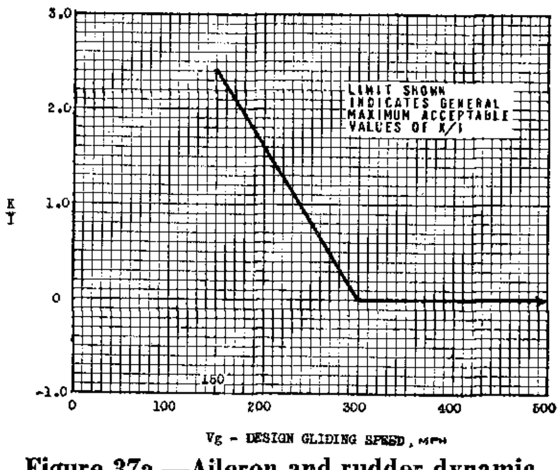


Figure 37a.—Aileron and rudder dynamic balance.

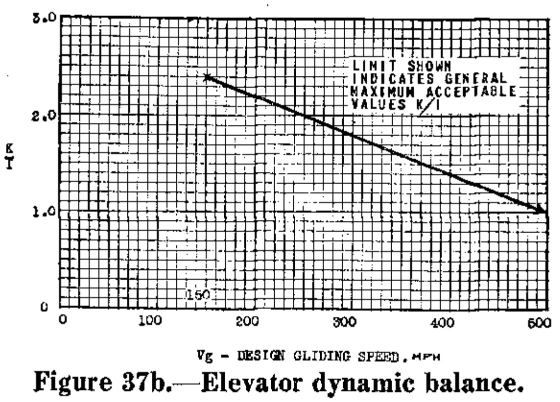


Figure 37b.—Elevator dynamic balance.

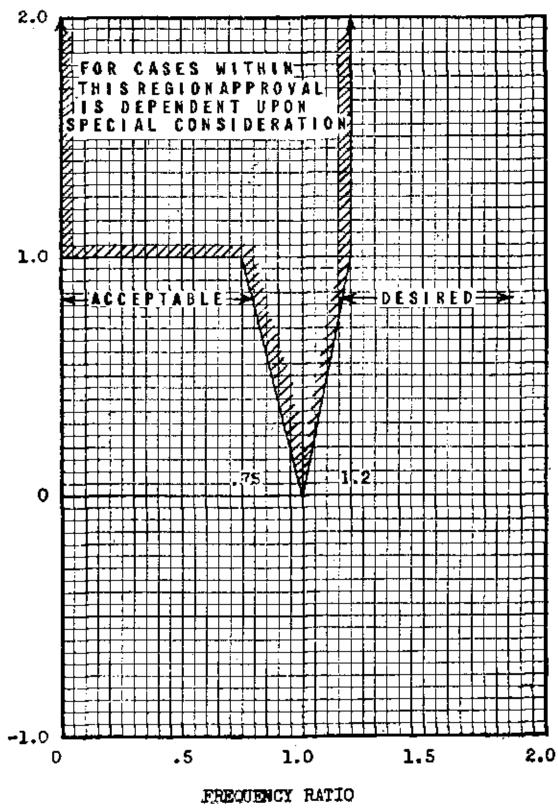


Figure 37c.—Dynamic balance vs. frequency ratio for control surfaces.

D. Frequency Ratio

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

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

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

Tests conducted by the Air Corps with flutter models have indicated that, when the frequency ratio involving a movable surface is greater than 1.0 the possibility of flutter in this mode is much reduced as compared with frequency ratios less than 1.0. A study of previous vibration test data and the service records of the aircraft tested, together with Air Corps test data, have indicated the desirability of using the frequency ratio as an additional limitation on the curves of the dynamic balance coefficient (K/I) versus design gliding speed (V_g), as shown in figure 37a for the aileron and rudder and in figure 37b for the elevator.

2. This limitation is expressed as a curve in figure 37c. The shaded portion marked "Approval dependent upon special considerations" is considered an undesirable range and approval is subject to consideration of the modes involved, actual values of frequencies, speed of the aircraft, value of K/I, etc. For this reason it is advantageous to submit to the Administration as early as possible in the development of a new model an outline of the proposed flutter prevention measures.

3. The following frequency ratios should be determined from the vibration test data (see pages 71 to 75 and Table XVII) and check against the K/I—frequency ratio limitations of figure 37c: (For description of particular modes see 04.323.)

*a.*** $\frac{\text{Rudder (as a unit)}}{\text{Fuselage Torsion}}$, or * $\frac{\text{Rudders (Unsym.)}}{\text{Fuselage Torsion}}$

*b.*** $\frac{\text{Rudder (as a unit)}}{\text{Fuselage Side Bending}}$, or * $\frac{\text{Rudders (Unsym.)}}{\text{Fuselage Side Bending}}$

c. $\frac{\text{Rudder (as a unit)}}{\text{Fin Bending}}$

*d.** $\frac{\text{Rudders (Sym)}}{\text{Stabilizer Bending}}$

*e.** $\frac{\text{Rudders (Unsym)}}{\text{Stabilizer Rocking about Fuselage Attachments}}$

*f.** $\frac{\text{Rudder (Unsym)}}{\text{Fin Bending}}$

g. $\frac{\text{Elevator (Sym)}}{\text{Fuselage Vertical Bending}}$

h. $\frac{\text{Elevator (Sym)}}{\text{Stabilizer Bending}}$

i. $\frac{\text{Elevator (Unsym)}}{\text{Fuselage Torsion}}$, or $\frac{\text{Elevator (Unsym.)}}{\text{Stabilizer Rocking about Fuselage Attachments}}$

j. $\frac{\text{Aileron (Sym)}}{\text{Wing Bending (Sym)}}$

k. $\frac{\text{Aileron (Sym)}}{\text{Wing Torsion (Unsym)}}$

l. $\frac{\text{Wing Bending (Unsym)}}{\text{Wing Torsion (Unsym)}}$

m. $\frac{\text{Stabilizer Bending}}{\text{Stabilizer Torsion}}$

n. $\frac{\text{Balance Weight and Arm Assembly}}{\text{Surface mode (see below)}}$

*Only for rudders *not* in the plane of symmetry. **Only for a rudder *in* the plane of symmetry.

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

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

o. $\frac{\text{Aileron (Unsym)}}{\text{Wing Bending (Unsym)}}$

p. $\frac{\text{Aileron (Unsym)}}{\text{Wing Torsion (Unsym)}}$

q. $\frac{\text{Wing Bending (Unsym)}}{\text{Wing Torsion (Unsym)}}$

r. $\frac{\text{*Rudder (Torsion)}}{\text{Fin Rocking about Stabilizer Attachments}}$

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

s. $\frac{\text{*Stabilizer Torsion}}{\text{Fuselage Torsion}}$

*Only for rudders *not* in the plane of symmetry.

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

E. Detail Design

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

a. The trailing edges of movable surfaces should be rugged to reduce the possibility of additional weight being added during field repairs with an adverse effect on the mass balance characteristics.

b. Tab mechanisms should be simple and rugged to avoid improper assembly, or the possibility of play developing due, for example, to open end (i. e., magneto type) ball thrust bearings being inserted backwards.

c. Provide adequate "carry through" structure to insure rigidity.

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

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

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

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

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

04.41 DETAIL DESIGN OF WINGS.

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

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

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

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

04.410 External bracing. When streamline wires are used for external lift bracing they shall be double unless the design complies with the lift-wire-cut condition specified in § 04.2161. (See also § 04.273.)

04.4100 Wire-braced monoplanes. If monoplane wings are externally braced by wires only, the right and left sides of the bracing shall be independent of each other so that an unsymmetrical load from one side will not be carried through the opposite wires before being counteracted, unless the design complies with the following conditions: (*a*) The minimum true angle between any external brace wire and a spar is 14 degrees. (*b*) The counter (landing) wires are designed to remain in tension at least up to the *limit* load. (*c*) The landing and flying wires are double.

04.4101 Multiple-strand cable shall not be used in lift trusses.

04.4102 Jury struts. When clamps are used for the attachment of jury struts to lift struts, the design shall be such as to prevent misalignment or local crushing of the lift strut.

04.411 Wing beams. Provisions shall be made to reinforce wing beams against torsional failure, especially at the point of attachment of lift struts, brace wires and aileron hinge brackets.

04.4110 Wing beam joints. Joints in metal beams (except pinned joints), and joints in mid-bays of wood beams shall maintain 100 percent efficiency of the beam with respect to bending, shear and torsion.

04.412 Drag truss. Fabric-covered wing structures having a cantilever length of overhang such that the ratio of span of overhang to chord at root of overhang is greater than 1.75 shall have a double system of internal drag trussing spaced as far apart as possible, or other means of providing equivalent torsional stiffness. In the former case counter wires shall be of the same size as the drag wires. (See also § 04.275.)

04.4120 Multiple-strand cable shall not be used in drag trusses unless such use is substantiated to the satisfaction of the Administrator.

04.413 Aileron and flap attachments. Aileron and flap attachment ribs or brackets shall be rigidly constructed and firmly attached to the main wing structure in order to reduce wing flutter tendencies.

04.414 Internally-braced biplanes. Internally-braced biplanes shall be provided with *N* or *I* struts to equalize deflections, and the effect of such struts shall be considered in the stress analysis.

04.415 Fabric covering. Fabric covering shall comply with the requirements of § 04.400 and shall be attached in a manner which will develop the necessary strength, with due consideration for slip-stream effects. (See § 04.-314.)

A. Cotton Fabric, Reinforcing Tape, Surface Tape and Lacing Cord

CAA grade A fabric.—Cotton fabric used on aircraft with wing loadings of 11 lbs. per sq. ft. or over, or design gliding speeds of 240 mph or over (see Fig. 38) should conform with or exceed the following specification. (Note: AN-CCC-C-399-1 exceeds this spec.).

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

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

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

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

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

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

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

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

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

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

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

CAA lightweight fabric.—Cotton fabric used on aircraft with wing loadings up to and including 8 lbs. per sq. ft., or design gliding speeds up to and including 150 mph (see figure 38) and having rib spacings in accordance with figure 39, has the same specifications as Grade A fabric except for the following:

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

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

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

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

Intermediate grade fabric.—A straight line variation between the physical properties of Grade A and lightweight fabrics should be assumed in selecting fabric for aircraft which have wing loadings between 8 and 11 lbs. or design gliding speeds between 150 and 240 mph. For example, fabric having a breaking strength of 60 lbs. per inch and a thread count of 90 is superior to lightweight fabric by 77 percent in thread count but only by 33 percent in strength. Thus the fabric is suitable for all aircraft having wing loadings of $8+.33\ (11-8)=9$ lbs. per sq. ft., or less, or design gliding speeds of $150+.33\ (240-150)=180$ mph, or less, as shown by the broken line in figure 38.

Reinforcing tape used with Grade A fabric should conform with or exceed the following specification:

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

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

Width—inches: ¼; ⅜; ½; ⅝; ¾; 1;
Breaking strength—lbs: 80; 120; 150; 170; 200; 250;

Reinforcing tape for lightweight fabric should conform with *a* above, and the breaking strength should not be less than one-half of the values listed for Grade A tape.

A straight line variation between the physical properties of the Grade A and lightweight reinforcing tapes should be assumed in selecting reinforcing tape for use with intermediate grade fabrics.

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

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

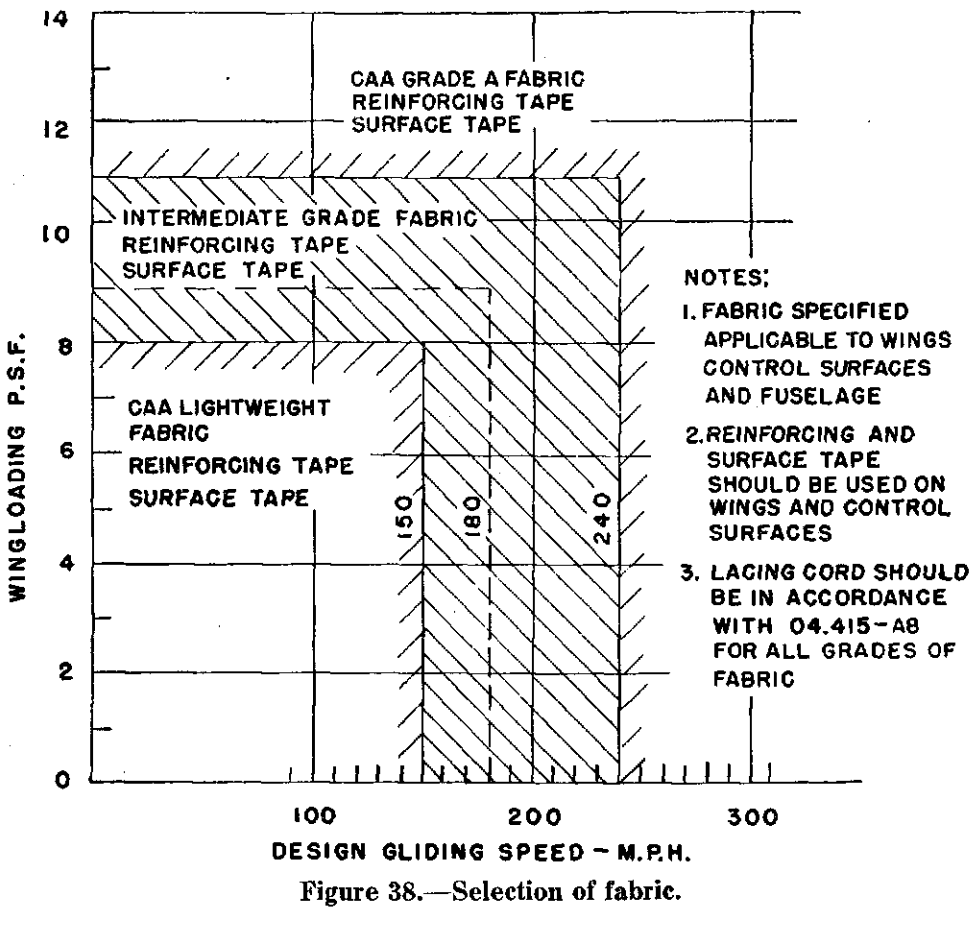


Figure 38.—Selection of fabric.

B. Other Covering Materials

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

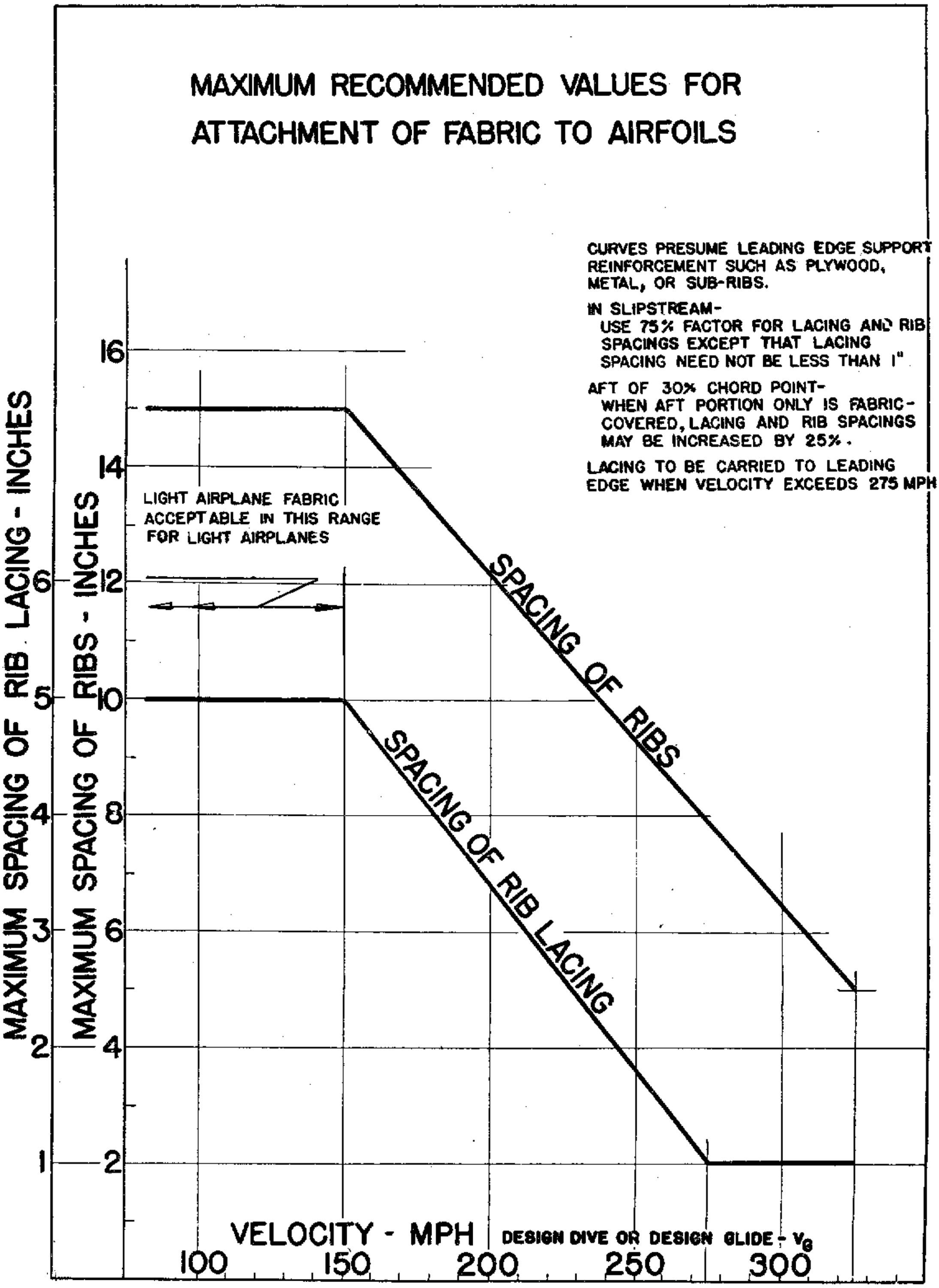


Figure 39.—Fabric attachment.

C. Technique of Covering

Seams.—All seams should be plain lap, folded fel or French fel seams (see fig. 40) machine stitched except as indicated below. Eight to ten stitches per inch should be used. The row of stitches nearest each folded edge of each seam should be approximately 1/16-inch from the edge of the fold and the rows should be ¼- to ⅜-inch apart.

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

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

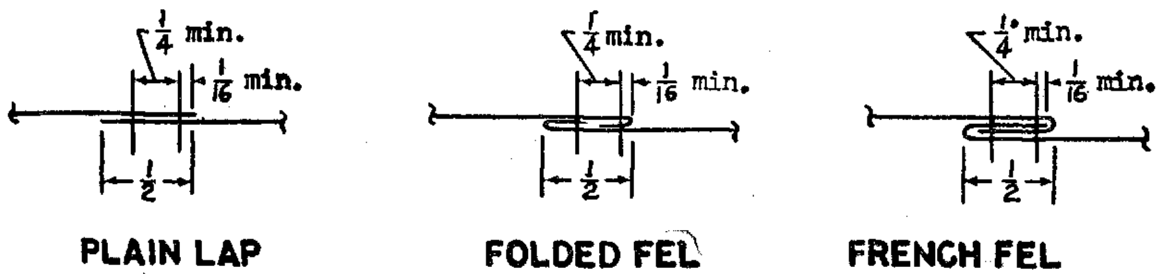


Figure 40.—Fabric stitching seams.

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

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

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

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

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

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

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

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

c. Stitch spacing should not exceed those shown in figure 39 which has been derived from extensive experience.

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

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

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

04.416 Metal-covered wings. The detail design of such wings shall incorporate suitable provision against buckling or wrinkling of metal covering as specified in §§ 04.201 and 04.314.

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

In an attempt to establish an empirical method of predicting panel sizes which will be free from unsatisfactory "oil-canning", figure 41 has been included as a proposal. In this case the skin thickness and unsupported panel width have been considered the main variables. Other important variables include stress (if appreciable) carried by skin, airspeed, wing loading, and workmanship. Comments and data on this subject are solicited.

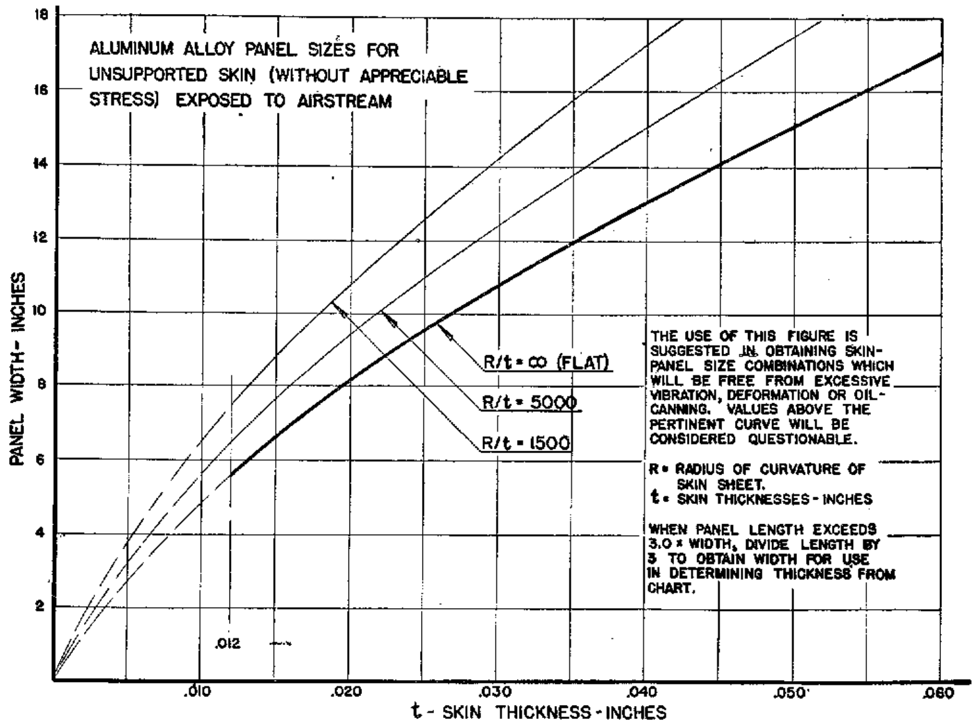


Figure 41.—Aluminum alloy panel sizes.

04.42 DETAIL DESIGN OF TAIL AND CONTROL SURFACES.

It is very important that control surfaces have sufficient torsional rigidity. No specific limits of permissible maximum deflection of the surface alone are offered, since these may vary widely with the type, size and construction of the surface. However, the behavior of the surface during proof tests should be closely observed. In addition the effect of the control system "stretch" on the total surface deflection under limit maneuvering loads should be considered from the standpoint of "surface usefulness", as described in paragraph 11, page 108.

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

External wire bracing on tails is subject to vibration and the design of the wire assembly and end connections should be such as to withstand this condition. Swaged tie rods are recommended, except that for use on light aircraft unswaged rods are acceptable if the points covered under "Tie rods and wires," page 89 are followed. Leading edges and struts should have adequate strength to withstand handling loads if handles or grips are not provided.

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.

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

04.421 Stops. When an adjustable stabilizer is used, stops shall be provided at the stabilizer to limit its movement, in the event of failure of the adjusting mechanism, to a range equal to the maximum required to balance the airplane.

Stops are specifically required in the case of adjustable stabilizers and elevator trailing edge tab systems (04.421 and 04.4210). Some form of stop should, however, be employed at all surfaces in order to avoid interferences and possible damage to the parts concerned and to limit the travel to the approved range. (See also "Stops," page 109.)

04.4210 Elevator trailing edge tab systems shall be equipped with stops which limit the tab travel to values not in excess of those provided for in the structural report. This range of tab movement shall be sufficient to balance the airplane under the conditions specified in § 04.704.

04.422 Hinges. Hinges of the strap type bearing directly on torque tubes are permissible only in the case of steel torque tubes which have a multiplying factor of safety as specified in § 04.276. In other cases sleeves of suitable material shall be provided for bearing surfaces.

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

a. Provision for lubrication should be made if self-lubricated or sealed bearings are not used.

b. The effects of deflection of the surfaces, such as in bending, should be allowed for, particularly with respect to misalignment of the hinges. This may also influence spacing of the hinges.

c. Sufficient restraint should be provided in one or more brackets to withstand forces parallel to the hinge centerline. Rudders, for instance, may be subjected to high vertical accelerations in ground operation.

d. Hinges welded to elevator torque tubes or similar components may prove difficult to align unless kept reasonably short and welded in place in accurate jigs.

e. Piano type hinges are acceptable with certain restrictions. In general only the "closed" type should be used, i. e., the hinge leaf should fold back under the attachment means. The attachment should be made with some means other than wood screws, and this attachment should be as close as possible to the hinge line to reduce flexibility. Piano hinges should not be used at points of high loading, such as opposite control horns, unless the reaction is satisfactorily distributed. Due to the difficulty in inspecting or replacing a worn hinge wire, it is better to use several short lengths than one long hinge.

04.4220 Clevis pins may be used as hinge pins provided that they are made of material conforming with, or the equivalent of, S. A. E. Specification 2330.

04.423 Elevators. When separate elevators are used they shall be rigidly interconnected.

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

04.424 Dynamic and static balance. All control surfaces shall be dynamically and statically balanced to the degree necessary to prevent flutter at all speeds up to the design gliding speed.

Dynamic balance.—A movable surface is dynamically balanced with respect to a given axis if an angular acceleration of the surface about that axis does not tend to cause the surface to swing about its own hinge line. A control surface which is dynamically balanced about a certain axis will therefore remain "neutral" with respect to a torsional vibration about that axis; that is, it will act as though rigidly connected with, and a part of, the fixed surface to which it is attached.

As the types of flutter likely to be encountered in aircraft structures involve both torsional and bending vibration, the type of balancing employed and the choice of a suitable reference axis for any given case will depend on the particular form of flutter to which the component is subjected.

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

a. When a surface is in complete static balance the numerical value of the product of inertia (*K*) is the same for any set of *parallel* oscillation axes. However, the sign of the product of inertia (*K*) will depend on the location of the oscillation axis with respect to the center of pressure (*CP*) of the surface.

b. It should be noted that when each section of a surface perpendicular to its hinge axis is statically balanced, the surface will be in complete dynamic balance for oscillation about any axis *perpendicular* to the hinge axis; i. e., $K/I=0$.

c. When the surface is statically balanced it will have some dynamic unbalance with respect to oscillations about an axis *parallel* to the hinge axis; i. e., $K/I=1.0$.

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

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

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

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

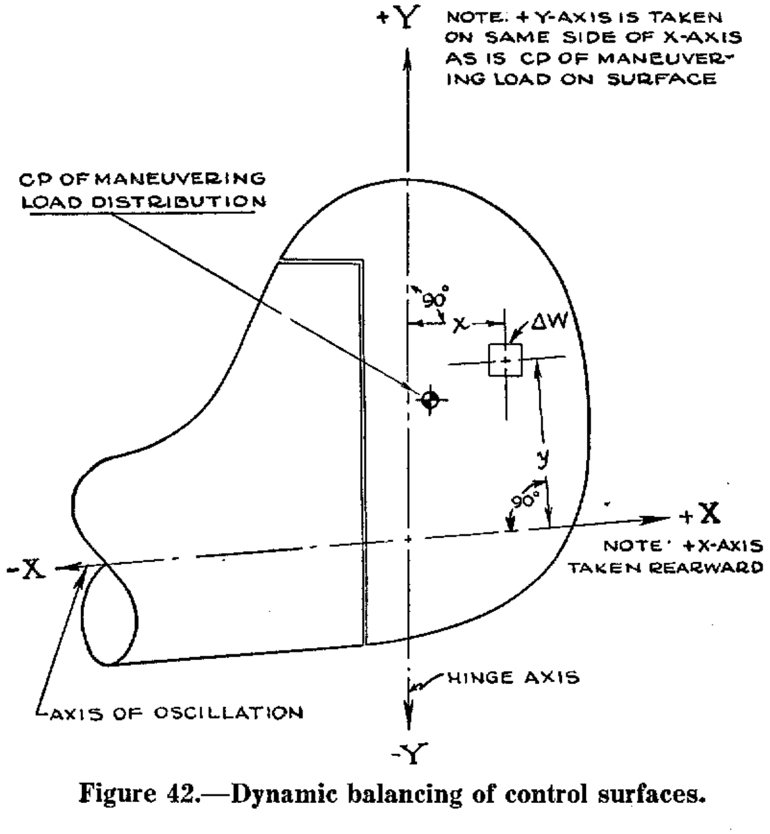


Figure 42.—Dynamic balancing of control surfaces.

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

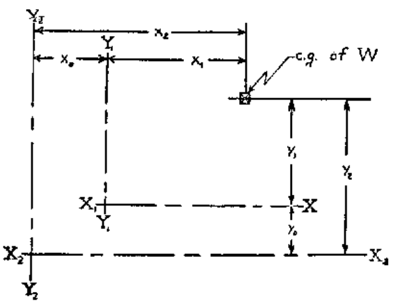
Product of inertia with respect to two axes that are mutually perpendicular.—In computing the dynamic balance coefficient, K/I, of a control surface for axes that are mutually perpendicular (within 15°), the following procedure may be used: Referring to figure 42.

a. Assume X-axis coincident with the assumed (or known) oscillation axis. Positive direction from the Y-axis is aft of control surface hinge axis, and negative forward of hinge.

b. Assume Y-axis coincident with the control surface hinge axis. Positive direction from X-axis is taken on the same side of the X-axis as is the center of pressure (CP) of the maneuvering load on the surface (see figs. 19 and 22 for the maneuvering load distribution). It should be noted that it is unnecessary to compute the position of the CP for these purposes, if the side of the X-axis on which it lies may obviously be determined by inspection.

c. After the reference axes have been established, the surface should be divided into relatively small parts and the weight of each such part (w) and the perpendicular distance from its CG to each axis (x to Y-Y axis and y to X-X axis) should be determined and tabulated. (See Table XXI.) *The weights and distances should be accurately determined.* The weights and CG locations of doped fabric and trailing edge material are sometimes underestimated with a resulting serious unbalance condition, and a larger value of K/I. In addition changes in service may tend to increase the unbalance. Referring to figure 42, the product of inertia of the item of weight, w, is equal to wxy. The product of inertia, K, of the complete surface is the sum of the individual products of inertia of each part. Hence, $K=wxy$. The weights should be expressed in pounds and the distance in inches. K is then in lbs.-inches².

d. The moment of inertia (I_{y-y}) of the control surface about its hinge axis may be computed from the data found for computing K (in paragraph c, above). I for a small part of the weight, w, is equal to wx^2, when x is the perpendicular distance from its CG to the hinge axis. Hence I_{y-y} is equal to the sum of the individual moments of inertia of each part and is equal to Σwx^2. The weight should be in pounds and the distance in inches, so that I_{y-y} will be in lbs.-inches². It should be noted that the correct value of I_{y-y} will only be obtained, if the weight items are broken down into a sufficient number of small parts, *especially in the chordwise direction.* This is particularly important for such items as fabric covering and tape, dope, metal skin, trailing edge tabs, tab operating mechanism, etc., *unless* the moment of inertia is first obtained about a parallel axis through the CG of the larger concentrated weight, w, and then transferred to the hinge axis; i. e., $I_{y-y}=I_{cz}+wd^2$ where d is the perpendicular distance in inches between the CG and the hinge axis.



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

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

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

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

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

TABLE XXI.—Mass Balance Computations

MODEL: ______________ SURFACE: ______________ DATE: ______________

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

Static unbalance = Algebraic sum of Σ (Col. 4) and Σ (Col. 5) =

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

Product of inertia with respect to two axes that are not mutually perpendicular.—This case might occur for some wing bending versus aileron mode of vibration, with, for example, the relations shown in figure 43. As shown in ACIC No. 711, the product of inertia for the inclined axes (*O-O* and *Y-Y*) can be obtained from the perpendicular axes (*X-X* and *Y-Y*) value by the use of the following equation:

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

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

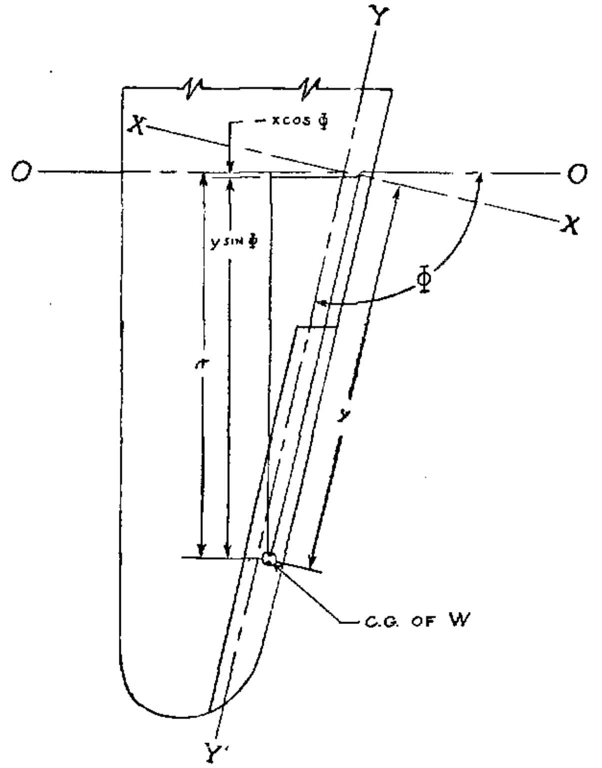


Figure 43.—Inclination of the axis for a typical wing-bending vs. aileron mode of vibration.

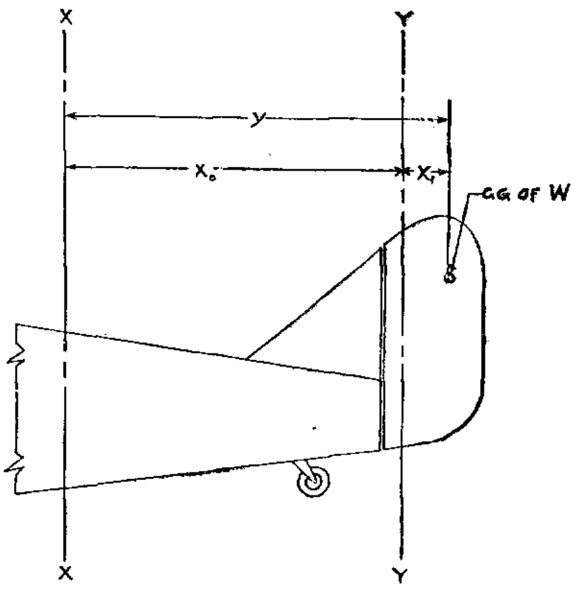


Figure 44.—Parallel axes for a typical fuselage-bending vs. rudder mode of vibration.

Product of inertia with respect to two parallel axes and in whose plane the control surface CG is located.—This case may be of importance in some of the wing torsion versus aileron and fuselage bending versus rudder or elevator modes of vibration. Using the same nomenclature as in the previous cases where *Y-Y* is the hinge line of the control surface and *X-X* is the axis of oscillation of the body as shown in figure 44 which represents a fuselage side bending versus rudder mode of vibration, then

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

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

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

The special case of parallel axes wherein the control surface CG is located outside of the plane of the axes, may in most cases be resolved into the above parallel axes case by projecting the *X*-axis to the plane through the control surface hinge resulting in a new *X'*-axis and resolving the *CG* reaction into components perpendicular and parallel to this plane. This may only be done when it can be shown that the *CG* of the control surface falls in the new plane through the *X'* and *Y*-axes which will be found true for most rudders and elevators. However, for the aileron, as shown in figure 45, where the hinge axis is usually near the bottom of the surface resulting in the *CG* being above an *X'-Y* plane, it will be necessary to consider the components of the *CG*, since an appreciable unbalance may be present even with a statically balanced aileron, for the true oscillation mode involving rotation about the *X*-axis.

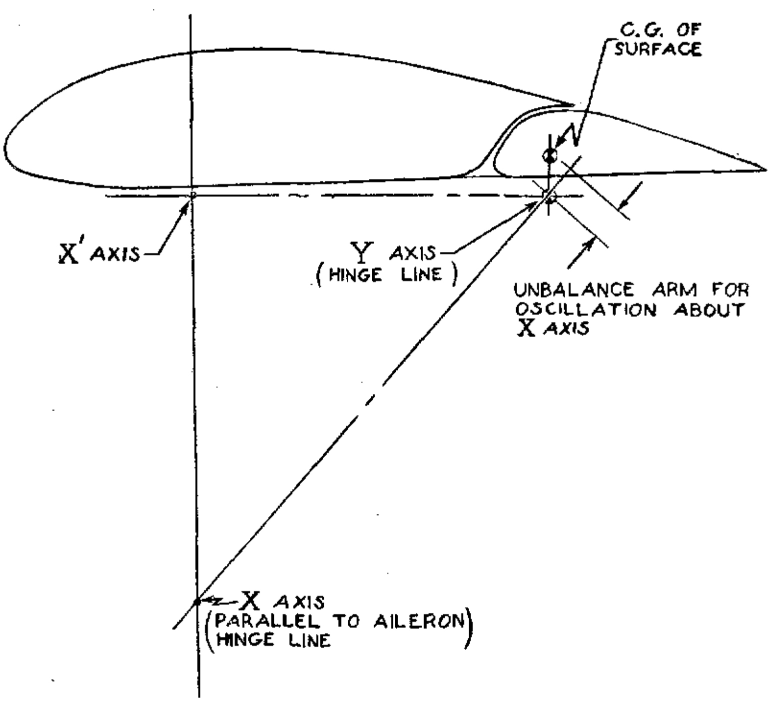


Figure 45.—Parallel axes with the control surface *CG* outside the plane of the axes.

04.425 Wing flaps. Flaps shall be so installed as not to induce flutter or appreciable buffeting.

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

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

04.426 Tabs. The installation of trim and balancing tabs shall be such as to prevent the development of any free motion of the tab. When trailing edge tabs are used to assist in moving the main surface (balancing tabs), the areas and relative movements shall be so proportioned that the main surface is not overbalanced at any time.

Minimum deflections and play are of first importance in the installation of these surfaces. Strength of the surface and anchorage should be sufficient to prevent damage or misalignment from handling. This is particularly true of thin sheet tabs which are set by bending to the proper position. See also "Balance" discussion pages 102 to 106.

04.43 DETAIL DESIGN OF CONTROL SYSTEMS.

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

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

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

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

5. *Cables.*—Control cables should be of the 6×19 or 7×19 extra flexible type, except that 6×7 or 7×7 flexible cable is acceptable in the $\frac{3}{32}$-inch diameter size and smaller provided that particular care is taken to prevent wear. Cable smaller than $\frac{1}{8}$-inch diameter should not be used in primary control systems, except that smaller sizes may be used for tab control systems where, in the event of cable failure, it is demonstrated that the airplane can be safely controlled in flight and landing operations. (See the following paragraph 7 regarding the use of fairleads.) For properties of control cable see Table 4-14 of ANC-5 and for cable terminal efficiencies see 4.530 of the same publication. End splices should be made by an approved tuck method such as that of the Army and Navy, except that standard wrapped and soldered splices are acceptable for cable less than $\frac{3}{32}$ inch in diameter. Approved swaged-type terminals are also acceptable. It should be remembered that cable sizes are governed by considerations of control system deflection as well as by strength requirements, particularly when long cables are used.

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

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

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

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

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

When dual controls are provided in aircraft to be used either for primary or advanced instruction, at least those controls necessary for normal safe operation, as for example, ignition switches, fuel valves, mixture and carburetor temperature controls, propeller pitch controls, etc., should be accessible from either control seat. (See CAR 20.81.) If the applicant wishes to provide for the carrying of a passenger in a control position, this may only be done if the controls are completely removable and are removed. Where a set of controls is removable, all controls, switches, valves, etc. must be accessible to the pilot at the permanent control. By removable controls is meant the removal of the control column or use of a throw-over type wheel or its equvalent, and the removal, disconnection or boxing-in of the dual rudder and brake pedals as to prevent possibility of interference by the passengers.

When dual controls are provided and all flight and power-plant controls cannot be normally operated from either seat, a placard should be installed similar to the following: To be flown solo from ---------- seat only.

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

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

13. *Power boost controls.*—Such controls should exhibit control force versus surface deflection curves which are smooth and free from discontinuities. (See also 04.75420.) Consideration should be given in the design and in test to the effects of the temperature variation to be expected in operation, in order to avoid the possiblility of jamming or excessive lag. Small changes in valve adjustments or other settings should not result in marked changes in operating or control characteristics.

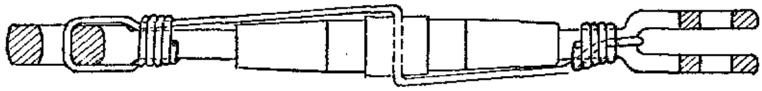
14. *Installation of turnbuckles.*—Fork ends of turnbuckles should not be attached directly to control surface horns or to bellcrank arms unless a positive means (such as the use of shackles, universal joints, etc.) is used to prevent binding of turnbuckles relative to the horns or bellcrank arms which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

Turnbuckle strength (AN std.)	Type of wrap	Diameter of wire	Material (annealed condition)
800	Single	0.040	Copper, brass, galv. steel [1]
1,600	Single	.040	Copper, brass, galv. steel [1]
2,100	Double	.040	Stainless steel.
2,100	Double	.040	Copper, brass, galv. steel.[1]
3,200 and greater	Single	.072	Copper, brass, galv. steel.[1]
	Double	.051	Copper, brass, galv. steel.[1]
	Double	.040	Stainless steel.

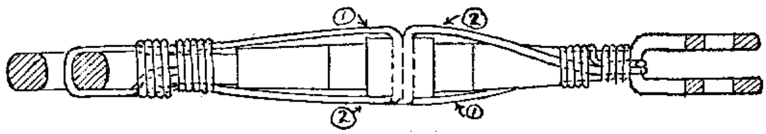
[1] Galvanized or tinned soft iron or tinned steel wires are also acceptable.

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

SINGLE WRAP



DOUBLE WRAP



NOTES: 1. Wire 1 is passed through the turnbuckle hole as shown, the two wire ends are passed through the right and left hand ends of the turnbuckle and are then bent back along the barrel of the turnbuckle.
2. Wire 2 is installed and wrapped (these wraps are next to the ends of the turnbuckle).
3. The two loose ends of wire 1 are then wrapped.

04.430 Installation. All control systems and operating devices shall be so designed and installed as to provide reasonable ease of operation by the crew and so as to preclude the probability of inadvertent operation, jamming, chafing, interference by cargo, passengers or loose objects, and the slapping of cables against parts of the airplane. All pulleys shall be provided with satisfactory guards.

Where water rudder controls are used, they should be so arranged to preclude possibility of the rudder jamming due to being damaged during taxiing or take-off or of freezing immediately after take-off in cold weather and thus interfering with the air controls. An acceptable preventive is a spring incorporated in the system between the water rudder and air controls which will permit some operation of the air rudder even though the water rudder may be fixed due to any accidental cause.

04.431 Stops. All control systems shall be provided with stops which positively limit the range of motion of the control surfaces. Stops shall be capable of withstanding the loads corresponding to the design conditions for the control system.

The stops in the control system should be located as close to each movable surface as is practicable to prevent changes in control system flexibility from affecting the ranges of travel, and to prevent interferences between the control surfaces under normal flight and ground operating conditions.

The degree of rigidity in a control system and interference of operating units may make it desirable to incorporate additional stops near the operating force. However, in such cases it should be demonstrated that these additional stops in no way prevent the attainment of the satisfactory movements under the conditions mentioned above.

Since the range of movement of the control surfaces is highly important for the safe operation of the aircraft, the stops controlling the movements should be preferably non-adjustable after the final approved movements are established. In lieu of non-adjustable stops, however, it will be acceptable if the manufacturer installs adequate warning against change of the adjustment without check against approved limits in a suitable placard installed adjacent to the adjustable stops.

04.432 Joints. Bolts with castellated nuts safetied with cotter pins or with an approved type of self-locking nut shall be used throughout the control system, except that the use of clevis pins in standard cable ends, thimbles and shackles is satisfactory for light airplanes.

Bolts, straight pins, taper pins, studs, and other fastening means should be secured with approved locking devices. (See page 88). Rivets should not be subjected to appreciable tension loads.

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

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

04.433 Welds. Welds shall not be employed in control systems to carry tension without reinforcement from rivets or bolts.

04.434 Flap controls. The flap-operating mechanism shall be such as to prevent sudden, inadvertent, or automatic opening of the flap at speeds above the design speed for the extended flap conditions. The time required to fully extend or retract flaps shall not be less than 15 seconds, unless it can be demonstrated to the satisfaction of the Administrator that the operation of the flaps in a lesser time does not result in unsatisfactory flight characteristics. Means shall be provided to retain flaps in their fully retracted position and to indicate such position to the pilot.

Undesirable flight characteristics, such as loss of lift and consequent settling, may result from too rapid operation of flaps which give appreciable lift. If flap retraction or extension time is less than 15 seconds, the behavior of the airplane must be such that during such operation of the flaps, the aircraft shall be easily and satisfactorily controllable with one hand. When the prime function of the flap is to act as a brake, however, slow operation is not so important. When more than one flap is installed, the control and means of interconnection should be such as to insure that the flaps function simultaneously, unless it can be demonstrated that no excessive rolling tendencies exist in the event of uneven operation or sudden partial failure.

04.434-T Flap controls. For transport-category airplanes, the flap control shall provide means for bringing the flaps from any position within the operating range to any one of three positions, designated hereinafter as landing, approach, and take-off positions, or to the fully retracted position, by placing the primary flap control in a single setting marked as corresponding to each such flap position, the flaps thereupon moving directly to the desired position without requiring further attention. If any extension of the flaps beyond the landing position is possible, the flap control shall be clearly marked to identify such range of extension.

The landing position, approach position, and take-off position, or any of them, may be made variable with altitude or weight by means of a secondary flap control provided for that purpose. Such a secondary control, if provided, shall operate independently of the primary control and in such manner that when it has been adjusted (for the effect of weight or altitude), the necessary flap position can thereafter be obtained by placing the primary flap control in the desired position. The secondary control shall be so designed and marked as to be readily operable by the crew.

The rate of flap retraction shall be such as to permit compliance with § 04.7540-T.

The ability of the airplane to comply with the maximum stalling speeds of 04.7530-T and the minimum rates of climb required by 04.7531-T depend critically upon specific flap positions. The operating rules for transport category airplanes contained in CAR 61.712 specify a maximum take-off weight such that with the "take-off" flap setting it is barely possible in the event of engine failure either to stop within the length of the runway or proceed on the remaining engine(s) and attain a height of 50 feet at the end of the runway. A lesser flap setting would improve the ability of the airplane to climb and therefore to reach the 50 ft. altitude, but would lessen its ability to stop within the field. A greater setting would improve the ability to stop but lessen the ability to climb to 50 feet. Similarly, the operating rules in effect specify a maximum landing weight such that with an "approach" and a "landing" flap setting, it is possible safely to execute an approach and landing, or, if necessary, to interrupt the landing process and "go around" for another attempt. Lesser flap settings would improve the ability of the airplane to "go around" but would require more distance to land and conversely for greater flap settings. Also, at some point during the complete process it is necessary to change from the "approach" to the "landing" flap setting, or from one or the other of these to "retracted" setting; i. e., it may be necessary that the crew select and obtain a precise flap setting under operating conditions such that the crew can devote a minimum of attention to the process or during an emergency, and, further, the safety of the operation demands that it be done surely.

It may also be noted that the maximum stalling speeds are specified as true indicated airspeeds and are, therefore, independent of altitude, while, at a given flap setting and weight, the rate of climb available ordinarily decreases with altitude. This suggests the possibility, where operation of the airplane at airports having various altitudes is contemplated, that optimum conditions in respect of carrying the maximum weight permitted by the stalling speed and rates of climb requirements would be obtained if the flap positions corresponding with the required flap control settings were made variable with altitude (i. e., the flap extension corresponding with a given control setting should decrease with increasing altitude of the field).

The purpose of this requirement is to specify a flap control which will permit the accomplishment of the use of the flap described above with the minimum of attention on the part of the pilot and co-pilot. While it is not required that it be possible to stop the flap at any point between the positions corresponding with any adjacent pair of the required control settings, which would provide more flexibility in the use of the flap than otherwise, nevertheless this is considered desirable unless the provisions necessary to accomplish this impair the reliability of the control mechanism to produce the flap positions corresponding with the required settings. There appears also to be no objection to providing for extension of the flaps beyond the position corresponding with the "landing" control setting provided the control clearly indicates it to be beyond this setting and so provides warning to the crew that the minimum rate of climb is not available. It should be also noted that 04.7533–T permits the use of such flap position in the determination of the landing distance.

04.435 Tab controls. Tab controls shall be irreversible and nonflexible, unless the tab is statically balanced about its hinge line. Proper precautions shall be taken against the possibility of inadvertent or abrupt tab operation and operation in the wrong direction.

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

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

04.4350 When adjustable elevator tabs are used for the purpose of trimming the airplane, a tab position indicator shall be installed and means shall be provided for indicating to the pilot a range of adjustment suitable for safe take-off and the directions of motion of the control for nose-up and nose-down motions of the airplane.

04.436 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 prohibited except under the following conditions: (*a*) The airplane shall 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 shall be of a type and design satisfactory to the Administrator. (*c*) Rubber cord shall not be used for this purpose.

04.437 Single-cable controls. Single-cable controls are prohibited except in special cases in which their use can be proved to be satisfactory.

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

04.438 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 shall be shown. (*a*) The locking device shall be so installed as to positively prevent taxiing the aircraft faster than 20 miles per hour, either intentionally or inadvertently, while the lock is engaged. (*b*) Means shall be provided to preclude the possibility of the lock becoming engaged during flights.

04.439–T Trim controls. For transport category airplanes, the trimming devices shall be capable of continued normal operation in spite of the failure of any one connecting or transmitting element in the primary control system. Trim controls shall operate in the plane and with the sense of the motion of the airplane which their operation is intended to produce.

The first sentence of this provision requires trim controls so designed that, in the event of failure of the corresponding primary control, the trim control will still continue to perform its normal function. This is considered a reasonable safety precaution because, in the event of primary control failure but with a trim control capable of continued operation, it may be possible, by means of the trim and powerplant controls, to land safely; whereas, if the trim control is also inoperative a safe landing would be very difficult to make and could be catastrophic particularly in the case of failure of the elevator control.

The second sentence in its effect requires standardization of the trimming controls for all transport category airplanes. This is required in an effort to avoid the possibility of improper operation, particularly during emergencies, or the shifting of crews from one type airplane to another, and to relieve the load upon the attention of the flight crew which the operation of an airplane of that type requires, by providing trim controls uniform in their operation. This is considered warranted in view of the present much greater dependence which is placed upon the trimming devices than when they were first generally introduced into airplane design.

The motion of the airplane to which the text of the requirement refers is a rotary motion; i. e., the motion involved in longitudinal or "elevator tab" trimming is a rotation about the lateral axis of the airplane and the plane of this motion is the plane of symmetry or a plane parallel to it. The sense of the motions is, therefore, a direction of rotation which the regulation requires to be the same as that of the airplane. Practically, the section requires that each trim control handle be either a wheel or a crank completely exposed and that it be located and operated as follows:

a. The longitudinal or "elevator" trim control must be so located as to rotate about a lateral axis and must rotate clockwise when viewed from the left to produce "nose up" trim, or to reduce the speed or to reduce the primary control force required to maintain a speed lower than the trimmed speed.

b. The lateral or "aileron" trim control must be so located as to rotate about a longitudinal or "fore and aft" axis and must rotate clockwise when viewed from aft to lower the right or starboard wing.

c. The directional or "rudder" trim control must be so located as to rotate about a normal or "vertical" axis and must rotate clockwise when viewed from above to change heading to the right.

It should be noted that 04.435 and 04.4350 also apply to the transport category and that, therefore, the installation of an elevator tab position indicator is required. Position indicators are recommended for all trimming tabs or for any other element actuated by the trimming device.

04.44 DETAIL DESIGN OF LANDING GEAR.

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

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

04.440 Shock absorption. All landing gear (including tail gear installations) shall be provided with shock-absorbing systems which will permit the airplane to be landed under the conditions specified in § 04.2411 and § 04.2420 without exceeding the ultimate load used in the analysis of any landing gear member. (See § 04.340 for proof of absorption capacity.) If the design of the shock-absorbing system is such that the above method of specifying the required energy absorption capacity appears to give irrational results, an alternate method will be considered upon presentation of pertinent data.

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

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

Where t=component of wheel travel in the direction of the resultant external force.
h=specified height of drop,
n=load factor, and
η=absorber efficiency.

Thus when a certain height of drop h must be met without exceeding a load factor n, the recommended minimum wheel travel for any absorber efficiency may be computed. While absorber efficiencies as high as 0.85 have been developed, it should be noted that such shock absorbers tend to give bouncing and undesirable taxiing characteristics. This may be obviated by ample travel in combination with an absorber which does not develop high loads in the first part of travel but rather "builds-up" gradually to a peak load only when near the fully deflected position. In such cases, an efficiency of 0.60 to 0.70 may be expected. The effect of the tire in altering the above relationship will in general not be large because, while it provides additional energy absorption, its deflection increases the energy to be dissipated. Structural deflections, while not usually of importance, may in some cases appreciably reduce load factors.

04.441 Shock-absorbing systems. The shock-absorbing systems employed shall incorporate suitable means for absorbing the shocks developed in taxiing or running over rough ground.

04.442 Wheels. Main landing gear wheels shall be of a type or model certificated by the Administrator in accordance with the provisions of Part 15 and shall not be subjected to static loads in excess of those for which they are certificated. Tail wheels may be of any type or model and are not certificated. Nose wheels are subject to special rulings to be made by the Administrator.

04.4420 For the purpose of these regulations main landing gear wheels are considered as those nearest the airplane center of gravity with respect to fore-and-aft location.

04.4421 For the purpose of these regulations a tail wheel is considered as one which supports the tail of a conventional airplane in the three-point landing attitude. A nose wheel is considered to be a wheel supporting the nose of the airplane when the two main wheels are located behind the center of gravity.

04.443 Tires. A landing gear wheel may be equipped with any make or type of tire, provided that the tire is a proper fit on the rim of the wheel and provided that the tire rating of the Airplane Tire Committee of the Tire and Rim Association is not exceeded.

A wheel appended to a previously approved tail skid installation will not be classed as a "landing gear wheel." See "Minor Changes," page 11 for an acceptable procedure of use in making such a change.

04.4430 When specially constructed tires are used to support an airplane, the wheels shall be plainly and conspicuously marked to that effect. Such markings shall include the make, size, number of plies and identification marking of the proper tire.

04.444 Retracting mechanism. When retractable landing wheels are used visual means shall be provided for indicating to the pilot, at all times, the position of the wheels. Separate indicators for each wheel are required when each wheel is separately operated unless a single indicator is obviously satisfactory. In addition, landplanes shall be provided with an aural or equally effective indicator which shall function continuously after the throttle is closed until the gear is down and locked.

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

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

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

04.4440 A positive lock shall be provided for the wheels in the extended position, unless a rugged irreversible mechanism is used.

The usual reduction ratios of screw and nut, and of worm and worm wheel combinations, are considered to provide irreversibility. Detents or other means should be provided however if there is appreciable creeping. Some types of swinging arms which move slightly past dead center to a position against a stop are also acceptable, but the effect of bouncing on landing should be considered. Consideration should be given to providing a lock or other device that requires a positive manual effort being made to retract the landing gear.

04.4441 Manual operation of retractable landing gears shall be provided for.

04.445–T Brakes. Transport category airplanes shall be equipped with brakes certificated in accordance with the provisions of Part 15 for the maximum certificated landing weight at sea level and the power-off stalling speed, V_{s_0} as defined in § 04.7511–T. The brake system for such airplanes shall be so designed and constructed that in the event of a single failure in any connecting or transmitting element in the brake system, or the loss of any single source of hydraulic or other brake operating energy supply, it shall be possible, as shown by suitable test or other data, to bring the airplane to rest under the conditions specified in § 04.7533–T with a mean negative acceleration during the landing roll of at least 50 percent of that obtained in determining the landing distance under that section.

This requirement is based upon the fact that compliance with the operating rules of CAR 61.712 will require great dependence upon the presence and proper functioning of brakes unless the runways involved are unusually long.

The nature and extent of the "test or other data" required to show compliance with this requirement will necessarily depend upon a great many things such as, for example, the general arrangement of the landing gear, the design of the brake system, the extent to which the capacity of the brakes is used in establishing the landing distance required by 04.7533–T, the amount of available performance data for the brakes, etc. The simplest possible procedure would appear to be to determine the average deceleration during a landing ground roll using no brakes and then to establish the landing distance required by 04.7533–T by using the brakes to the extent necessary to double the mean deceleration so established. It appears likely, however, that this procedure would result in excessive landing distance and might seriously limit the use of the airplane in scheduled operation.

If it is desired by the applicant to make the maximum possible use of the brakes in establishing the landing distance, and if also the contribution of the brakes to the total deceleration is relatively large, it will be necessary so to design the brake system as to permit the application of slightly less than half the braking deceleration so developed under the conditions specified in this section. The following dual system is recommended: dual wheel elements (drums or disc units), transmitting elements, power sources, master cylinders, etc., connected to a single pedal on each rudder pedal, such that the failure of any single one of these would leave half the total braking capacity symmetrically disposed about the plane of symmetry of the airplane. With such a system it should be possible to show compliance with this section by means of calculation based upon the test data necessary to establish the landing distance plus those obtained as a part of the certification process for the brake (See CAR 15.104).

If the system be so designed that under the conditions here specified appreciably less than half the total braking capacity remains or if the remaining capacity be asymmetrically disposed, it will almost certainly require tests to determine that half the mean deceleration may in fact be developed and/or that the airplane may be safely controlled directionally while doing so.

04.45 HULLS AND FLOATS. (See also § 04.46.)

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

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

04.450 Buoyancy (main seaplane floats). Main seaplane floats shall have a buoyancy in excess of that required to support the gross weight of the airplane in fresh water as follows: (*a*) 80 percent in the case of single floats; (*b*) 90 percent in the case of double floats.

It should be noted that Canadian requirements specify that twin-float seaplanes shall have at least 100 percent reserve buoyancy in the floats.

04.4500 Main seaplane floats for use on aircraft of 2,500 pounds or more maximum authorized weight shall contain at least five watertight compartments of approximately equal volume. Main seaplane floats for use on aircraft of less than 2,500 pounds maximum authorized weight shall contain at least four such compartments.

04.451 Buoyancy (boat seaplanes). The hulls of boat seaplanes and amphibians shall be divided into watertight compartments in accordance with the following requirements: (*a*) In aircraft of 5,000 pounds maximum authorized weight or more the compartments shall be so arranged that, with any two adjacent compartments flooded, the hull and auxiliary floats (and tires, if used) will retain sufficient buoyancy to support the gross weight of the aircraft in fresh water; (*b*) in aircraft of 1,500 to 5,000 pounds maximum authorized weight the compartments shall be so arranged that, with any one compartment flooded, the hull and auxiliary floats (and tires, if used) will retain sufficient buoyancy to support the maximum authorized weight of the aircraft in fresh water; (*c*) in aircraft of less than 1,500 pounds maximum authorized weight watertight subdivision of the hull is not required; (*d*) bulkheads may have watertight doors for the purpose of communication between compartments.

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

04.452 Water stability. Auxiliary floats shall be so arranged that when completely submerged in fresh water, they will provide a righting moment which is at least 1.5 times the upsetting moment caused by the aircraft being tilted. A greater degree of stability may be required in the case of large flying boats, depending on the height of the center of gravity above the water level, the area and location of wings and tail surfaces, and other considerations.

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

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

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

D=total displacement of the seaplane in cubic feet.

04.453 Float design. In designing the bow portion of floats and hulls suitable provision shall be made for the effects of striking floating objects.

04.46 FUSELAGE AND CABINS.

04.460 Provision for turn-over. The fuselage and cabins shall be designed to protect the passengers and crew in the event of a complete turn-over and adequate provision shall be made to permit egress of passengers and crew in such event. This requirement may be suitably modified when the possibility of a complete turn-over in landing is remote.

04.461 Doors. Closed cabins on all aircraft carrying passengers shall be provided with at least one adequate and easily accessible external door.

04.4610 No passenger door shall be located in the plane of rotation of an inboard propeller, nor within 5 degrees thereof as measured from the propeller hub.

04.462 Exits. Closed cabins on aircraft carrying more than 5 persons shall be provided with emergency exits in addition to the one external door required by § 04.461, consisting of movable windows or panels or of additional external doors which provide a clear and unobstructed opening, the minimum dimensions of which shall be such that a 19-inch by 26-inch ellipse may be completely inscribed therein. The location and the method of operation of emergency exits shall be approved by the Administrator. If the pilot is in a compartment separate from the cabin, passage through such compartment shall not be considered as an emergency exit for the passengers. The number of emergency exits required is as follows: (*a*) Aircraft with a total seating capacity of more than 5 persons, but not in excess of 15, shall be provided with at least one emergency exit or one suitable door in addition to the main door specified in § 04.461. This emergency exit, or second door, shall be on the opposite side of the cabin from the main door. If desired, an additional emergency exit may be provided in the top of the cabin, but such an installation shall not obviate the necessity for an exit on each side; (*b*) aircraft with a seating capacity of more than 15 persons shall be provided with an additional emergency exit or door either in the top or side of the cabin for every additional 7 persons or fraction thereof above 15, except that not more than 4 exits, including doors, will be required if the arrangement and dimensions are suitable for the purpose intended.

04.463 Pilot's compartment. The pilot's compartment shall be so constructed as to afford suitable ventilation and adequate vision to the pilot under normal flying conditions. In cabin aircraft the windows shall be so arranged that they may be readily cleaned or easily opened in flight to provide forward vision for the pilot. The ventilation requirements of § 04.467 shall also apply to the pilot's compartment.

In providing for adequate vision and movement of controls it is necessary that sufficient head room, clearance for controls, and space for movement of the hands and feet should be provided when parachutes are being worn. The above are essential for aircraft designed for acrobatic use or for demonstration of flying ability in connection with pilot's flight tests.

If provided, an adjustable pilot's seat should be equipped with a suitable lock which is readily operable, positive in action and of such construction that it will neither fail under the maximum loads, which can be applied by the pilot to any of the controls, nor become disengaged due to taxiing over rough ground or water.

04.4630 The pilot and the primary control units, excluding cables and control rods, shall be so located with respect to the propellers that no portion of the pilot or controls lies in the region between the plane of rotation of any propeller and the surface generated by a line passing through the center of the propeller hub and making an angle of 5 degrees forward or aft of the plane of rotation of the propeller.

04.4631 A metal identification plate shall be permanently affixed in a visible location in the pilot's comapartment of each airplane. This plate shall contain the manufacturer's name, the date of manufacture, the manufacturer's serial number and the model designation. The manufacturer shall specify the fuel capacity of each fuel tank on the manufacturer's identification plate, or on or adjacent to the fuel shut-off valves in the pilot's compartment.

04.4632 Means shall be provided by which the operating personnel is suitably informed of all operation information and limitations deemed necessary by the Administrator.

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

1. Instruments should be marked for airspeed not to be exceeded in glide or dive (red); level flight or climb (yellow); with flaps extended (green), and the rpm and manifold pressure not to be exceeded in take-off (red); in climb (yellow); in all other operations (green).
2. Acceptable methods of marking include:
   a. Pointers, adjustable on the ground only.
   b. Sectors or lines properly marked and outlined on the face of the dial, under the glass.
   c. Lines painted on the glass face of the instrument when *a* or *b* above is impracticable and when the glass is adequately secured against rotation. Such lines should be painted over a suitably etched or scratched line on the glass itself. This etching or scratching is considered advisable for more serviceable markings.
3. When considered necessary by the CAA, operating information and limitations such as the following should be displayed on an appropriate placard in full view of operating personnel, and/or included in a manual, or its equivalent, which must be carried in the pilot's compartment and be accessible at all times:
   a. Emergency ceiling and conditions under which it may be obtained.
   b. All other information or limitations considered necessary by the CAA properly to inform operating personnel of the conditions necessary for operation in compliance with the Civil Air Regulations.

04.4633 The windows and windshields of the pilot's compartment in airplanes certificated for air transportation service shall be so arranged as to provide satisfactory forward vision and protection under all conditions and, to accomplish this, particular attention shall be paid to the following detail requirements: (*a*) Sufficient data specifying the windshield material, number of laminations, binder if any, size and shape of panes, angle of panes to flight path and method and rigidity of mounting, shall be forwarded to the Administrator for rulings as to the acceptability of the windshield from the standpoint of strength; (*b*) Windshields shall be so installed that they can be easily opened in flight and shall be so arranged that the air stream and snow or rain are deflected across the opening, or to provide equivalent results; (*c*) The pilot's compartment shall be so constructed and arranged as to prevent glare or reflections which would interfere with the vision of either pilot, particularly while flying at night. The aircraft will be flown by a representative of the Administrator during hours of darkness to determine compliance with this provision.

04.4634 The pilot's compartment in airplanes certificated for air transportation service shall be so constructed as to prevent any leakage into it when the airplane is flying in rain or snow.

04.4635 When a second pilot is required (§ 61.520), two seats shall be installed side-by-side in the pilot's compartment of airplanes certificated for air transportation service from either of which the airplane shall be fully and readily controllable. If any difference exists as to convenience of the instruments and controls necessary for safe flight such difference should favor the left-hand seat. The left-hand seat shall be known as the first pilot's seat and the right-hand one as the second pilot's seat.

04.4636 The navigation instruments for use by the pilot in airplane certificated for air transportation service shall be so installed as to be easily visible to him with the minimum practicable deviation from his normal position and line of vision when he is looking out and forward along the flight path and they shall also be visible to the second pilot.

04.4637 All airplanes certificated for air transportation service shall be provided with a door or an adequate openable window between the pilot's compartment and the passengers' cabin. When a door is provided it shall be equipped with a locking means which shall prevent passengers from opening such door while in flight.

04.464 Passenger chairs. Seats or chairs for passengers shall be securely fastened in place in both open and closed airplanes, whether or not the safety belt load is transmitted through the seat. (See Part 15 and § 04.2640 for safety belt requirements.)

04.465 Baggage compartments. Each baggage and mail compartment shall bear a placard stating the maximum allowable weight of contents, as determined by the structural strength of the compartment (§ 04.265) and by flight test (§ 04.742). Suitable means shall be provided to prevent the contents of mail and baggage compartments from shifting.

04.466 Reinforcement near propellers. Surfaces near propeller tips shall be suitably stiffened against vibration and the effects of ice thrown from the propeller. (See § 04.611 for clearance requirements.)

04.467 Passenger compartments. A suitable ventilation system shall be provided which will preclude the presence of fuel fumes and dangerous traces of carbon monoxide in each passenger compartment.

04.5 EQUIPMENT

04.50 GENERAL.

The equipment required shall be dependent upon the type of operation for which certification is to be made. The requirements specified herein (§ 04.5) shall be the basic equipment requirements and such additional equipment as may be specified in other sections of the Civil Air Regulations for specific special cases shall be supplemental hereto unless otherwise specified.

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

04.501 An approved life preserver or flotation device is one approved by the Administrator for such usage on sea-going vessels.

04.502 Fire extinguishing apparatus approved by the Underwriters Laboratories is considered to be of an approved type.

04.51 NON-AIR-CARRIER (NAC) AIRPLANES.

Airplanes which are certificated as non-air carriers shall have at least the following equipment:

04.510 NAC landplanes—Visual-contact day flying (within 100 miles of a fixed base):

(*a*) One air-speed indicator. (See § 04.5800 for installation requirements.)
(*b*) One altimeter.
(*c*) A tachometer for each engine.
(*d*) An oil-pressure gauge when an oil-pressure system is employed.
(*e*) A water thermometer for each water-cooled engine.
(*f*) An oil thermometer for each air-cooled engine.
(*g*) A manifold-pressure gauge, or equivalent, for each altitude engine.
(*h*) A fuel quantity gauge. (See § 04.624 for requirements.)
(*i*) Certificated safety belts for all passengers and members of the crew. (See Part 15 for belt requirements and § 04.5810 for installation requirements.)
(*j*) A portable fire extinguisher, which extinguisher shall be of an approved type, which shall have a minimum capacity, if carbon tetrachloride, of one quart, or, if carbon dixoide, of two pounds, or, if other, of equivalent effectiveness; except that any extinguisher of not less than half the above capacity may be used in an airplane equipped with an engine whose maximum rating is 40 horsepower or less. (See § 04.5811 for installation requirements.)
(*k*) Landing gear position indicator for retractable main landing gear. (See § 04.444 for requirements.)
(*l*) A device for measuring or indicating the amount of oil in the tanks. (See § 04.633 for requirements.)
(*m*) A first aid kit.
(*n*) A log-book for the airplane and one for each engine. (See Part 01 for requirements.)
(*o*) Rigging information for airplanes with wire-braced wings, either in the form of a sketch or listed data, which shall include sufficient information to permit proper rigging.

04.511 NAC landplanes—Visual-contact day flying (unlimited distance). Airplanes of this category shall have the equipment specified in § 04.510 and, in addition, there shall be installed: (*a*) A magnetic compass. (See § 04.5803 for installation requirements.)

04.512 NAC landplanes—Visual-contact night flying. Airplanes of this category shall have the equipment specified in § 04.511 and, in addition, there shall be installed:

(*a*) A set of certificated standard forward position lights in combination with a certificated tail light. (See Part 15 for light requirements and § 04.5827 for installation requirements.)
(*b*) Two electric landing lights if the aircraft is operated for hire: *Provided, however,* That only one such landing light shall be required for any airplane certificated for a weight of less than 1,500 pounds. (See § 04.5825 for installation requirements.)
(*c*) Certificated landing flares as follows, if the aircraft is operated for hire beyond an area within a circle with a radius of 3 miles drawn from the center of the airport of take-off (see Part 15 for flare requirements and § 04.5813 for installation requirements):

Airplanes of 3,500 pounds maximum authorized weight or less—5 Class 3 flares or 3 Class 2 flares.
Airplanes of between 3,500 pounds and 5,000 pounds maximum authorized weight—4 Class 2 flares.
Airplanes of 5,000 pounds maximum authorized weight or more—2 Class 1 flares or 3 Class 2 flares and one Class 1 flare.
If desired, airplanes of less than 5,000 pounds maximum authorized weight may carry the flare equipment specified for heavier airplanes.

(*d*) A storage battery suitable as a source of energy supply for such lights and radio as are installed. (See § 04.5821 for installation requirements.)
(*e*) Radio equipment, if the aircraft is operated in a control zone (§ 60.103), as follows: A radio range and weather broadcast receiver operating within the frequency range of 200 to 400 kilocycles. Under normal atmospheric conditions this receiver must be capable of receiving with a range of 100 miles intelligence emanated from a radio range or weather broadcast station the equivalent of an SBRA installation.
(*f*) A set of spare fuses. (See § 04.5822 for installation requirements.)

04.513 NAC landplanes—Instrument day flying. Airplanes of this category shall have the equipment specified in § 04.511 and, in addition, there shall be installed:

(*a*) Radio equipment: Same as § 04.512 (*e*), whether the aircraft is operated for hire or not, and, in addition, a radio transmitter operated on 3105 kilocycles with a power output sufficient to establish communication at a distance of at least 100 miles under normal atmospheric conditions. Additional frequencies may be employed subject to approval of the Federal Communications Commission.

(*b*) A gyroscopic rate-of-turn indicator.

(*c*) A bank indicator. (Instruments (*b*) and (*c*) may be combined in one instrument if desired.)

(*d*) A sensitive altimeter which shall be adjustable for changes in barometric pressure and compensated for changes in temperature.

(*e*) A clock with a sweep-second hand.

(*f*) A storage battery suitable as a source of energy supply for the radio equipment installed. (See § 04.5821 for installation requirements and § 04.5823.)

(*g*) A generator.

(*h*) A set of spare fuses. (See § 04.5822 for installation requirements.)

(*i*) A rate-of-climb indicator.

04.514 NAC landplanes—Instrument night flying. Airplanes of this category shall have the equipment specified in §§ 04.512 and 04.513 combined. The storage battery shall be suitable as a source of energy supply for both the radio equipment and the lights.

04.515 NAC seaplanes and amphibians. The equipment requirements for seaplanes and amphibians shall be the same as specified for landplanes (§ 04.510 through § 04.514) except that seaplanes and amphibians shall not be certificated for operation over water out of sight of land unless they have at least the equipment specified in § 04.511, and except that all certificated seaplanes and amphibians shall also have an approved life preserver or flotation device for each person for which there is a seat, and except that all seaplanes and amphibians certificated for night operation shall also have a white anchor light. (See § 04.5824 for installation requirements.)

04.53 AIR CARRIER AIRPLANES—PASSENGERS (ACP).

Airplanes certificated for use by an air carrier in passenger service shall have installed at least the following equipment:

04.530 ACP landplanes—Visual-contact day flying. The same as specified in § 04.511 and, in addition, the following:

(*a*) An electrically heated pitot tube, or equivalent, for the air-speed indicator.

(*b*) One additional portable fire extinguisher of the type specified in § 04.510 (*j*). (See § 04.5811 for installation requirements.)

(*c*) Fixed fire extinguishing apparatus of an approved type for each engine compartment.

(*d*) Type certificated radio equipment as specified in Part 40.

(*e*) A set of spare fuses. (See § 04.5822 for installation requirements.)

(*f*) A rate-of-climb indicator.

(*g*) A storage battery—Same as § 04.513 (*f*).

(*h*) A means for providing, without continuous manual operation, vision through the windshield adequate for executing take-offs and landings in rain.

04.531 ACP landplanes—Visual-contact night flying. The same as specified in § 04.530 and, in addition, the following:

(*a*) A set of certificated air-carrier airplane position lights. The forward lights may be air-carrier forward position lights or a combination of standard forward position lights and a set of auxiliary forward position lights. (See Part 15 for light requirements and section 04.5827 for installation requirements.)

(*b*) A storage battery of sufficient capacity for such lights and radio as are installed. (See § 04.5821 for installation requirements and § 04.5823.)

(*c*) Two electric landing lights. (See § 04.5825 for installation requirements.)

(*d*) Certificated landing flares as follows: 2 Class 1 flares or 3 Class 2 flares and one Class 1 flare. (See Part 15 for flare requirements and § 04.5813 for installation requirements.)

(*e*) Instrument lights. (See § 04.5826 for installation requirements.)

(*f*) Cabin lights in all passenger cabins and compartments.

(*g*) A generator. (See § 04.5823 for requirements.)

(*h*) Radio equipment same as § 40.235.

04.532 ACP landplanes—Instrument day flying. The same as specified in § 04.530 except § 04.510 (*b*) and, in addition, the following:

(*a*) A gyroscopic rate-of-turn indicator combined with a bank indicator.

(*b*) A gyroscopic instrument showing bank and pitch.

(*c*) A gyroscopic direction finder.

(*d*) Two sensitive type altimeters, both of which shall be adjustable for changes in barometric pressure and compensated for changes in temperatures; *Provided*, That aircraft in use on or before January 1, 1939, and thereafter replacements and additions of aircraft of the same make and model may, for purposes of standardization, be deemed to have met this requirement if there are installed in each such aircraft, one sensitive type altimeter and one standard type altimeter provided each is adjustable for changes in barometric pressure, and compensated for changes in temperature.

(*e*) A free air thermometer of the distance type, with an indicating dial in the cockpit.

(*f*) A clock with a sweep-second hand.

(*g*) A vacuum gauge, installed in the lines leading to instruments (*a*), (*b*), and (*c*).

(*h*) Type certificated radio equipment as specified in Part 40.

(*i*) Means shall be provided to indicate icing conditions, or the probability thereof, in the carburetor if the de-icing device specified in § 04.6291 requires the manual manipulation of controls.

(*j*) A storage battery suitable as a source of energy supply for the radio equipment installed. (See § 04.5821 for installation requirements and § 04.5823.)

(*k*) A generator. (See § 04.5823 for installation requirements.)

04.533 ACP landplanes—Instrument night flying. The same as specified in §§ 04.531 and 04.532 combined. The storage battery, in this case, shall be of sufficient capacity for all radio equipment and all lights installed.

04.534 ACP seaplanes and amphibians. The same as specified for landplanes (§ 04.530 through § 04.533) and including the life preservers specified in § 04.515, except that when certificated for night operation they shall also have installed the anchor light specified in § 04.515.

04.58 INSTALLATION REQUIREMENTS.

The following regulations apply to the installation of specific items of equipment and are additional to the regulations of § 04.50.

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

04.5800 Airspeed indicator. This instrument shall be so installed as to indicate true airspeed at sea level with the maximum practicable accuracy but the instrument error shall not be more than plus or minus 3 percent, except that it need not be less than plus or minus 5 miles per hour, at the level flight speed corresponding to the design power (§ 04.105), at V_L (§ 04.111), or at the maximum attainable level flight speed, whichever is lowest.

This requirement necessitates that the airspeed indicator error at the top level flight speed for the power conditions of 04.5800 shall not exceed ± 5 mph for top level flight speeds up to 167 mph, and $\pm$ 3 percent for speeds 167 mph or over. It is considered desirable that the airspeed indicator error be within ± 5 mph throughout the speed range. It is also considered to be in the interest of safety, and therefore desirable, that any error present be such that the indicator reads high at comparatively high speeds and low at comparatively low speeds.

Any of the methods of calibration described here are acceptable to the CAA. The calibration should generally be conducted with the airplane fully loaded and with the center of gravity in an intermediate position. *If flaps are installed on the airplane, a separate calibration covering at least five speeds should be made with the flaps extended.* If the landing gear is retractable, the calibration with the flaps retracted should be made with the landing gear also retracted and that with the flaps extended should be made with the landing gear also extended.

The calibration may be conducted by means of a number of methods. The following are considered acceptable for use in conducting type inspections and are listed in the order of their desirability regarding accuracy and consistency of results obtained:

1. Suspended or trailing bomb.
2. Measured speed course.
3. Pacing by means of another airplane.

A. Calibration by Means of a Trailing Bomb

Because of the impracticability of conducting airspeed indicator calibration tests over a measured course at speeds approaching the stalling speed of the airplane being tested, the calibration at these low speeds should be made with a trailing bomb. The bomb may be used at any altitude and therefore lends itself readily to calibrations at speeds which would be considered unsafe at the altitudes at which calibration runs over a measured course are made. Reasonable accuracy may be expected from the use of a trailing bomb when the following precautions are taken:

The bomb itself should be calibrated at intervals not to exceed six months by conducting tests at various speeds with the bomb suspended from an airplane in order to obtain the bomb error. This process requires an accurate calibration by other independent means of the airspeed indicator of the airplane to which the bomb is attached or that the calibration of the bomb be made by flying over a measured course. It also requires calibration of the instrument to which the bomb is attached.

In order to eliminate the possibility of position error, it is recommended that both during the calibration of the bomb over a speed course and during calibration of the airplane indicator by means of the bomb, the length of the bomb cable paid out bear a constant relation to the span of the airplane used in each case. Thus, for example, if the airplane, the indicator for which is to be calibrated, has a span of 100 feet and the airplane used to calibrate the bomb a span of 36 feet, and also if 36 feet of cable be used during the bomb calibration, then 100 feet of cable should be used during the calibration of the indicator on the test airplane. The exact factor relating cable length to span (1.00 in the above example) should be based upon consideration of the total length of cable available and the greatest span of any airplane likely to be tested.

B. Calibration Over a Speed Course

A speed course consists of two ground stations easily identifiable from the air at an accurately known distance apart, preferably on level terrain. This distance should be at least two miles. The ground stations should be of such nature that it is possible to observe with considerable precision the instant the airplane is directly above them. Roads at right angle to the course direction are an example of such stations.

To insure accurate results, course calibrations should in general not be conducted if the wind speed exceeds approximately five mph or if the air is appreciably turbulent. In the case of a cross wind, the airplane should be headed parallel to the course and allowed to drift if necessary. If it be attempted to fly directly along the course by yawing the airplane into the wind, the resultant airspeed will be in error by an amount depending upon the airspeed and the cross wind velocity.

At least one run in each direction should be made at each of the selected speeds. In each run the altitude and the necessary power setting should become stabilized for a considerable distance before entering the speed course. Neither power (rpm or mp) nor altitude should be altered throughout the particular run. Each run should be made at as low an altitude as is considered to be safe, but in general should not be made at an altitude less than 50 feet above the ground. In conducting the "full throttle" run, when fixed or adjustable pitch propellers are installed, the pilot should not permit dangerously high rpm to be reached. In general, the rpm at full throttle should not exceed the rated rpm of the engine.

The following data should be observed:

a. The outside air temperature and the average pressure altitude of all runs should be observed and recorded.

b. The indicated airspeed should be recorded at frequent intervals throughout each run and the average of these recordings taken as the speed of that particular run.

c. The average manifold pressure, tachometer reading, and carburetor air temperature should be observed for each run and recorded as precisely as possible. "Average speed" should be determined as the average of the measured course ground speed for each direction. Do not average the time intervals.

C. Calibration by Means of Pacing With Another Airplane

This method consists essentially in flying another airplane, the airspeed calibration for which is known, alongside the test airplane at the same speed as indicated by their relative positions and reading simultaneously the airspeed indicator in each airplane. The method is obviously open to considerable error since it requires that zero relative speed between two airplanes be determined by direct observation. It also requires a reliable means of communication between two airplanes and, obviously, that the tests be conducted only in very smooth air. In general, the method should be regarded as a last resort.

D. General Discussion

It is impossible to place too much emphasis upon the necessity to obtain the most accurate possible airspeed calibration. Since all test data are to be reported in terms of true indicated airspeed and these reported data will necessarily cover the entire range from the stalling speed to the design gliding speed, no tests for the purpose of obtaining quantitative performance data should be conducted until a satisfactory and acceptable airspeed indicator calibration has been accomplished. Particular care should be exercised in determining the calibration at speeds near the stalling speed because of the dependence of the required rate of climb and the showing of compliance with the cooling, the landing speed, and flight characteristics requirements upon the calibrated speeds in the neighborhood of the stalling speed.

The calibration made at the full load condition for the airplane will in most cases be satisfactory for all of the tests. In certain cases, however, in which there are appreciable differences in weight between the full load and the foremost or rearmost *CG* loading conditions, it may be necessary to apply a correction to the calibration obtained at the full load condition for the difference in weight. Such correction is based upon the assumption that the airspeed indicator position error is a function of angle of attack of the airplane. This is equivalent to assuming it to be a function of lift coefficient, i. e., the indicator error will have the same magnitude at the same lift coefficient no matter what the speed involved. Since for level flight at a given lift coefficient the speed varies as the square root of the weight, this correction may be made by plotting the calibration as true indicated airspeed over the square root of weight of airplane during test against indicated airspeed over the square root of weight of airplane during test. (See below.)

$$\frac{\text{True Indicated Airspeed}}{\sqrt{\text{Weight of Airplane During Test}}} \text{ vs. } \frac{\text{Indicated Airspeed}}{\sqrt{\text{Weight of Airplane During Test}}}$$

The true indicated airspeed corresponding with a given indicator reading obtained with the airplane at another weight may then be obtained by calculating indicated airspeed over the square root of the new weight, entering the calibration and reading the corresponding TIAS over the square root of the weight of the airplane, and multiplying this by the square root of the new weight. The result is TIAS at the new weight.

In the case of modification of a previously certificated airplane, which requires a determination of a speed as a part of the tests necessary to determine the status of the altered airplane with regard to continued compliance with the Civil Air Regulations, it is considered desirable that an airspeed calibration be made of the test airplane even though the calibration of the indicator on the prototype, which was tested for the original certification, may have been made and may be available. Our experience indicates that there may be appreciable differences in the calibration for different individual airplanes of the same type.

04.5801 Powerplant instruments and controls. (See §§ 04.650 and 04.651.)

04.5802 Fuel quantity gauge. (See § 04.624.)

04.5803 Magnetic compass. This instrument shall be properly damped and compensated and shall be located where it is least affected by electrical disturbances and magnetic influences.

04.5804 Navigation instruments. Navigation instruments for use by the pilot shall be so installed as to be easily visible to him with the minimum practicable deviation from his normal position and line of vision when he is looking out and forward along the flight path and they shall also be visible to the second pilot.

04.5805 Gyroscopic instruments. All gyroscopic instruments shall derive their energy from engine-driven pumps or from auxiliary power units. Each source of energy supply and its attendant complete installation shall comply with the instrument manufacturer's recommendations for satisfactory instrument operation. On multi-engine aircraft each instrument shall have two separate sources of energy, either one of which shall be capable of carrying the required load. Engine-driven pumps, when used, shall be on separate engines. The installation shall be such that failure of one source of energy or breakage of one line will not interfere with proper functioning of the instruments by means of the other source.

04.581 Safety equipment installation.

04.5810 Safety belts. Safety belts shall be so attached that no part of the attachment will fail at a load lower than that specified in §04.2640.

04.5811 Fire extinguishers. The portable fire extinguisher specified in § 04.510 shall be so installed as to be accessible to the passengers. The two portable fire extinguishers specified in §04.530 shall be so installed that one is readily available to the crew and the other is near the main external cabin door where it shall be readily available to passengers and ground personnel.

04.5812 Safety belt signal. When a signal or sign is used to indicate to passengers the times that seat belts should be fastened, such signal or sign shall be located in a conspicuous place and so arranged that it can be operated from the seat of either pilot.

04.5813 Landing flares. Landing flares shall be releasable from the pilot's compartment. Structural provision shall be made for the recoil loads.

04.5814 De-icers. Positive means shall be provided for the deflation of all wing boots.

During the official type inspection tests, the actual operation and effect upon performance and flight characteristics will be observed for any de-icing devices provided for wings, tail surfaces, propellers, etc. No hazardous effects on flight characteristics shall result during their operation.

04.582 Electrical equipment installation.

04.5820 General. Electrical equipment shall be installed in accordance with accepted practice and suitably protected from fuel, oil, water and other detrimental substances. Adequate clearance shall be provided between wiring and fuel and oil tanks, fuel and oil lines, carburetors, exhaust piping and moving parts.

04.5821 Battery. Battery shall be easily accessible and adequately isolated from fuel, oil and ignition systems. Adjacent parts of the aircraft structure shall be protected with a suitable acid-proof paint if the battery contains acid or other corrosive substance and is not completely enclosed. If the battery is completely enclosed, suitable ventilation shall be provided. All batteries shall be so installed that spilled liquid will be suitably drained or absorbed without coming in contact with the airplane structure.

04.5822 Fuses. Fuses shall be so located that they can readily be replaced in flight. They shall break the current in a generating system at a sufficiently small current flow to adequately protect the lights, radio equipment and other parts of the circuit.

04.5823 Generator. When a generator is specified it shall have sufficient capacity to carry the entire running load. Such generator shall be engine-driven unless an approved equivalent system is provided. Auxiliary power units will be approved in lieu of batteries and engine-driven generators, provided that they are at least two in number and that the supply system is capable of carrying the entire running load with any one unit out of action.

04.58230 Running load. The running load shall be defined as the electric consumption of all lights, radio equipment and other electrical devices except those which are designed only for occasional intermittent use. Examples of devices regarded as intermittent are radio broadcasting equipment, landing lights and electrically operated landing gears and wing flaps. Radio range signal receivers and all other lights are considered a part of the constant load.

04.5824 Anchor lights. The anchor light specified for seaplanes and amphibians shall be so mounted and installed that, when the airplane is moored or drifting on the water, it will show a white light visible for at least two miles at night under clear atmospheric conditions.

04.5825 Landing lights. Electric landing lights shall be so installed on multiengine aircraft that at least one shall be not less than 10 feet to the right or left of the first pilot's seat and beyond the swept disk of the outermost propeller. On single-engine aircraft such lights shall be so installed that no visible portion of the swept disk of the propeller, if of the tractor type, is illuminated thereby. Individual switches for each light shall be provided in the pilot's compartment.

04.5826 Instrument lights. Instrument lights shall be so installed as to provide sufficient illumination to make all flight instruments easily readable and shall be equipped with rheostat control for dimming unless it can be shown that a non-dimming light is satisfactory.

04.5827 Position lights shall be installed so that, with the airplane in normal flying position, the forward red position light is displayed on the left side and the forward green position light on the right side, each showing unbroken light between two vertical planes whose dihedral angle is 110 degrees when measured to the left and right, respectively, of the airplane from dead ahead. Such forward position lights shall be spaced laterally as far apart as practicable. One rear position light shall be installed on the airplane at the rear and as far aft as possible and shall show a light visible aft throughout a dihedral angle of 140 degrees bisected by a vertical plane through the longitudinal axis of the airplane. Such light shall emit (a) in the case of a non-air carrier airplane, either a continuous white light as specified in CAR 15.2014 or alternate red and white flashes as specified in CAR 15.2015, and (b) in the case of an air carrier airplane, alternate red and white flashes as specified in CAR 15.2015. In lieu of such a single flashing rear position light, an airplane may carry two rear position lights, one red and one white, spaced as closely as possible to each other with one unit above the other and in combination emitting the red and white flashes specified in CAR 15.2015.

04.5828 Master switch. Electrical installations shall incorporate a master switch easily accessible to a member of the crew.

04.589 Miscellaneous equipment installation.

04.5890 Seats. Seats or chairs, even though adjustable, in open or closed airplanes, shall be securely fastened in place whether or not the safety belt load is transmitted through the seat.

04.5891 Accessories. Engine-driven accessories on multiengine aircraft shall be distributed among two or more engines.

04.6 POWERPLANT INSTALLATION

04.60 ENGINES.

Engines shall be of a type and design which has been type certificated, or found eligible for use in certificated aircraft, in accordance with the requirements of Part 13 or shall have been approved as airworthy in accordance with previous regulations.

04.61 PROPELLERS.

Propellers shall be of a type and design which has been certificated as airworthy in accordance with the requirements of Part 14 or shall have been approved as airworthy in accordance with previous regulations, except that wood propellers of a conventional type for use in light airplanes need not be certificated. In certain cases maximum engine bore limitations are also assigned to propellers. Propellers may be used on any engine provided that the certified power ratings, speed ratings, and bore of the engine are not in excess of the limitations of the propeller as certificated, and further provided that the vibration characteristics of the combination are satisfactory to the Administrator.

The propeller as installed on the aircraft must permit the clearance provided in 04.611. The "Minimum Permissible Diameter" which will be listed for propellers on the aircraft specification is that of the smallest propeller tested, multiplied by .98. This 2 percent margin is arbitrarily chosen as being the maximum permissible reduction in diameter of a given propeller which will not noticeably reduce performance. It also provides a margin to allow dressing down of metal propellers in service.

Fixed or adjustable pitch propellers.—The Aircraft Specification for each newly certificated aircraft equipped with a fixed or adjustable pitch propeller will specify the following propeller limitations:

Static rpm at maximum permissible throttle setting.	Not more than ______ rpm Not less than ______ rpm
Diameter________________________________	Not more than ______ inches Not less than ______ inches

Normally propellers made from wood will require a different range of static rpm limits than metal propellers. The basis for the preceding limitations is summarized here for convenient reference.

Maximum static rpm.—The maximum permissible static rpm will be the lower of the two static rpm values determined by the methods shown below:

a. Using the propeller having *both* the smallest diameter and lowest static rpm with which compliance with the Steady Rate of Climb, Take-off Distance and First Minute Climb requirements can be met:

Maximum permissible static rpm = Rated engine rpm − (rpm at the best rate of Climb Speed − Lowest Static rpm).

b. Using a propeller having *both* the largest diameter and highest static rpm for which approval is desired, the power-off rpm at placard V_g speed must not exceed 110 percent of the rated rpm of the engine at METO power.

Minimum static rpm.—The minimum static rpm will be established by the propeller having *both* the smallest diameter and lowest static rpm used in demonstrating compliance with the Steady Rate of Climb, Take-off Distance, and First Minute Climb requirements.

Maximum diameter.—The maximum diameter will be the smaller of the two values determined by the methods shown below:

a. Maximum diameter which will permit the clearances specified in 04.611. (Landplane 9 inches ground clearance, seaplane 18 inches water clearance. At least 1-inch clearance between propeller tips and structure.)

b. Maximum diameter used in placard V_g dive tests described in (*b*), "Maximum Static rpm".

Minimum diameter.—The minimum diameter will be the diameter of the smallest propeller used to demonstrate compliance with the Steady Rate of Climb, Take-off Distance and First Minute Climb requirements as described in "Minimum Static rpm" multiplied by a factor of 0.98.

To demonstrate that the vibration characteristics of the propeller are satisfactory in any given installation the blade vibration stresses should be measured under flight conditions. Past experience has indicated that this procedure is necessary with all propellers, except wood types. In some cases it is possible to determine the effect of changes or slightly new combinations by ground tests or comparisons of previous data, but in most cases flight tests for this purpose are necessary. The propeller manufacturers have the stress measuring equipment necessary to accomplish these tests and will furnish the results to the CAA together with their recommendations with respect to approval.

04.610 Controllable pitch. The control mechanism shall be designed and equipped with a positive stop which shall limit the minimum pitch so that the take-off crankshaft speed for which the aircraft is certificated is not exceeded during take-off with take-off power unless it is necessary to so locate the stop that a higher crankshaft speed may be used in an emergency. The means provided for controlling the pitch shall be so arranged as to minimize the attention required from a pilot to prevent the engines from exceeding their crankshaft speed limitations under any flight condition.

Controllable propellers.—The high pitch stop should be such that 110 percent of METO rpm will not be exceeded in power-off dives at placard "Never Exceed" speeds.

Constant speed propellers.—A control stop should be provided which will limit the low pitch range to the maximum permissible take-off rpm.

All stops should, when practicable, be located at the propeller governor or hub in order to reduce to a minimum the adjustments required in service.

Full feathering propellers.—Comments regarding requirements for control stops on constant speed propellers are also pertinent to full feathering propellers. The mechanism should be operated in flight to demonstrate satisfactory operation to full feathering position. These checks can usually be made during tests for one engine inoperative performance.

04.611 Propeller clearance. Propellers shall have a minimum ground clearance of 9 inches when the airplane is in a horizontal position with the landing gear deflected as it would be under the maximum authorized weight of the airplane. Propellers on seaplanes shall clear the water by at least 18 inches when the seaplane is at rest under the maximum authorized load condition. A clearance of at least 1 inch shall be provided between the tips of propellers and any part of the structure.

If the airplane is so designed that the normal ground attitude is more critical, from the standpoint of decreasing propeller clearance, than the horizontal attitude then the clearance should be measured in the normal ground attitude. The location of the center of gravity position should be such as to produce the most critical condition and shock struts and tires should be checked for recommended pressures before making measurements.

A minimum clearance of ¼-inch between the blade or hub and the cowling is recommended, and if variable pitch propellers are used this clearance should exist under the feathered or highest pitch condition.

04.62 FUEL SYSTEMS.

04.620 Capacity and feed. The fuel capacity shall be at least 0.15 gallon per maximum (except take-off) horsepower for which the airplane is certificated. Air-pressure fuel systems shall not be used. Only straight gravity feed or mechanical pumping of fuel is permitted. The system shall be so arranged that the entire fuel supply may be utilized in the steepest climb and at the best gliding angle and so that the feed ports will not be uncovered during normal maneuvers involving moderate rolling or side slipping. The system shall also feed fuels promptly after one tank has run dry and another tank is turned on. If a mechanical pump is used, an emergency hand pump of equal capacity shall be installed and available for immediate use in case of a pump failure during take-off. Hand pumps of suitable capacity may also be used for pumping fuel from an auxiliary tank to a main fuel tank.

04.621 Tank installation. No fuel tank shall be placed closer to an engine than the remote side of a firewall. At least one-half inch clear air space shall be allowed between the tank and firewall. Spaces adjacent to the surfaces of the tank shall be ventilated so that fumes cannot accumulate or reach the crew or passengers in case of leakage. If two or more tanks have their outlets interconnected they shall be considered as one tank and the air space in the tanks shall also be interconnected to prevent differences in pressure at the air vents of each tank of sufficient magnitude to cause fuel flow between tanks. Mechanical pump systems shall not feed from more than one tank at a time except by special ruling from the Administrator.

04.622 Tank construction. Each fuel tank shall be provided with either a sump and drain located at the point which is lowest when the airplane is in a normal position on the ground or outlets at the bottom of the tank provided with large mesh finger strainers. If a sump is provided, the main fuel supply shall not be drawn from the bottom of this sump. If no sump is provided the system drain shall be controllable from the pilot's compartment and shall act as a tank drain. Each tank shall be suitably vented from the top portion of the air space. Such air vents shall be so arranged as to minimize the possibility of stoppage by dirt or ice formation. When large fuel tanks are used, the size of the vent tubes should be proportioned so as to permit rapid changes in internal air pressure to occur and thereby prevent collapse of the tanks in a steep glide or dive. Tanks of 10 gallons or more capacity shall be provided with internal baffles unless suitable external support is provided to resist surging.

04.623 Tank strength. Fuel tanks shall be capable of withstanding an internal test pressure of 3½ pounds per square inch without failure or leakage. Fuel tanks of large capacity which have a maximum fuel depth greater than 2 feet shall be investigated for the pressure developed during the maximum *limit* acceleration with full tanks. Tanks shall be so designed, and the rivets or welds so located, as to resist vibration failures or leakage.

04.624 Gauge. A satisfactory gauge shall be so installed on all airplanes as to indicate readily to a pilot or flight mechanic the quantity of fuel in each tank while in flight. When two or more tanks are closely interconnected and vented, and it is impossible to feed from each one separately, only one fuel-level gauge need be installed. If a glass gauge is used, it shall be suitably protected against breakage.

04.625 Lines and fittings. All fuel lines and fittings shall be of sufficient size so that under the pressure of normal operation the flow is not less than double the normal flow required for take-off engine power. A test for proof of compliance with this requirement shall be made. All fuel lines shall be so supported as to prevent excessive vibration and should be located so no structural loads can be applied. Bends of small radius and vertical humps in the lines shall be avoided. Copper fuel lines which have been bent shall be annealed before installation. Parts of the fuel system attached to the engine and to the primary structure of the airplane shall be flexibly connected thereto. Flexible hose connections and fuel lines shall have metal liners or the equivalent. Fittings shall be of a type satisfactory to the Administrator.

04.626 Strainers. One or more strainers of adequate size and design, incorporating a suitable sediment trap and drain, shall be provided in the fuel line between the tank and the carburetor and shall be installed in an accessible position. The screen shall be easily removable for cleaning.

04.627 Valves. One or more positive and quick-acting valves that will shut off all fuel to each engine shall be within easy reach of the first pilot and the second pilot or of the flight mechanic. In the case of airplanes employing more than one source of fuel supply, suitable provision shall be made for independent feeding from each source.

04.6270 Dump valves. When fuel tanks are equipped with dump valves, the operating mechanism for such valves shall be within convenient reach of the first pilot and the second pilot, or of the flight mechanic. Dump valves shall be so installed as to provide for safe and rapid discharge of fuel.

04.628 Drains. One or more accessible drains shall be provided at the lowest point on the fuel systems to completely drain all parts of each system when the airplane is in its normal position on level ground. Such drains shall discharge clear of all parts of the airplane and shall be equipped with suitable safety locks to prevent accidental opening.

04.629 Miscellaneous fuel system requirements.

04.6290 Filler openings. All filler openings in the fuel system shall be plainly marked with the capacity and the word "fuel." Provision shall be made to prevent any overflow from entering the wing or fuselage.

04.6291 An adequate means shall be provided for preventing the formation of ice in the engine carburetors. (See also § 04.532 (*i*).)

04.63 LUBRICATION SYSTEMS.

04.630 General. Each engine shall have an independent oil supply. The oil capacity of the system shall be at least 1 gallon for every 25 gallons of fuel but shall not be less than 1 gallon for each 75 maximum (except take-off) rated horsepower of the engine or engines. A special ruling concerning the capacity will be made by the Administrator when oil may be transferred between engines in flight or when a suitable reserve is provided. The suitability of the lubrication system shall be demonstrated in flight tests in which engine temperature measurements are obtained. The system shall provide the engine with an ample quantity of oil at a temperature suitable for satisfactory engine operation.

04.631 Tank installation. Oil tanks shall be suitably vented and shall be provided with an expansion space which cannot be inadvertently filled with oil. Such expansion space shall be at least 10 percent of the total tank volume, except that it shall in no case be less than one-half gallon.

04.632 Tank strength. Oil tanks shall be capable of withstanding an internal test pressure of 5 pounds per square inch without failure or leakage. Tanks shall be so designed and the rivets or welds so located as to resist vibration failures and leakage.

04.633 Gauge. A suitable means shall be provided to determine the amount of oil in the system during the filling operation.

04.634 Piping. Oil piping shall have an inside diameter not less than the inside diameter of the engine inlet or outlet and shall have no splices between connections. Connections in the oil system shall be of a type satisfactory to the Administrator.

04.635 Drains. One or more accessible drains shall be provided at the lowest point on the lubricating systems to drain completely all parts of each system when the airplane is in its normal position on level ground. Such drains shall discharge clear of all parts of the airplane and shall be equipped with suitable safety locks to prevent accidental opening.

04.636 Oil temperature. A suitable means shall be provided for measuring the oil temperature at the engine inlet.

04.637 Filler openings. All filler openings in the oil system shall be plainly marked with the capacity and the word "oil."

04.64 COOLING SYSTEMS.

04.640 General. The cooling system shall be of sufficient capacity to maintain engine temperatures within safe operating limits under all conditions of flight during a period at least equal to that established by the fuel capacity of the aircraft, assuming normal engine power and speeds. Compliance with this requirement shall be demonstrated in flight tests in which engine temperature measurements are obtained under critical flight conditions including flight with one or more engines inoperative.

04.641 Radiators. Radiators shall be so mounted as to reduce vibration and eliminate strains causing distortion.

04.642 Piping. Piping and connections shall conform to accepted standards and shall not transmit vibration to the radiator or the structure of the aircraft.

04.643 Drains. One or more accessible drains shall be provided at the lowest points on the cooling system to drain completely all parts of such system when the airplane is in its normal position on level ground. Such drains shall discharge clear of all parts of the airplane and shall be equipped with suitable safety locks to prevent accidental opening.

04.644 Filler openings. All filler openings in the cooling system shall be plainly marked with the capacity of the system and the name of the proper cooling liquid.

04.65 POWERPLANT INSTRUMENTS, CONTROLS, AND ACCESSORIES.

04.650 Instruments. The engine instruments required are specified in § 04.5. The installation requirements for navigation instruments in § 04.5804 shall apply to tachometers and manifold pressure gauges. All other instruments shall be visible in flight to the pilot and co-pilot or to the flight mechanic. If the manifold pressure gauges and tachometers are not visible to the flight mechanic, he shall be provided with a duplicate set of these instruments.

04.651 Controls. All powerplant controls, including those of the fuel system, shall be plainly marked to show their function and method of operation.

04.6510 Throttle controls. Throttle controls shall be easily accessible to both pilots and shall be so arranged as to afford a positive and immediately responsive means of controlling all engines separately or simultaneously. Flexible throttle control systems shall be of a certificated type. A forward movement shall open the throttle.

04.6511 Ignition switches. Ignition switches shall be easily accessible to both pilots. A positive means for quickly shutting off all ignition of multiengine aircraft, by grouping of switches or otherwise, shall be provided.

04.6512 Propeller pitch controls. Separate pitch controls shall be provided for each propeller.

04.652 Accessories (Air carrier airplanes). (See § 04.5891.)

04.66 MANIFOLDING, COWLING AND FIREWALL.

04.660 General. All manifolds, cowling and firewalls shall be so designed and installed as to reduce to a minimum the possibility of fire either during flight or following an accident and shall therefore comply with accepted practice in all details of installation not hereinafter specified.

04.661 Manifolds. Exhaust manifolds shall be constructed of suitable materials, shall provide for expansion, and shall be arranged and cooled so that local hot points do not form. Gases shall be discharged clear of the cowling, airplane structure and fuel system parts of drains. They shall not blow back on the carburetor air intake or the pilot or passengers, nor cause a glare ahead of the pilot at night. No exhaust manifolding shall be located immediately adjacent to or under the carburetor or fuel system parts liable to leakage.

04.662 Air intakes. Carburetor air intakes shall be suitably drained and shall open completely outside the cowling unless the emergence of back-fire flames is positively prevented. The drain shall not discharge fuel in the path of possible exhaust flames.

04.663 Engine cowling. All cowling around the powerplant and on the engine side of the firewall shall be made of metal and shall be so arranged that any accumulations of dirt, waste or fuel may be observed without complete removal of the cowling. It shall fit tightly to the firewall, but openings may be provided if the airplane surface within 15 inches thereof is protected with metal or other suitable fireproofing material. The cowling shall be completely and suitably drained in all attitudes of flight and on the ground, with separate drains provided for the parts of the fuel system liable to leakage. All such drains shall be so located as to prevent fuel or oil from dripping onto the exhaust manifold or any parts of the aircraft and from permeating any material of a cellular nature.

04.664 Firewall. A firewall shall be provided unless the engine is mounted in an isolated nacelle with no fuel tanks. Such fire bulkhead shall be constructed in one of the following approved manners: (*a*) A single sheet of terne-plate not less than 0.028-inch thick. (*b*) A single sheet of stainless steel not less than 0.015-inch thick. (*c*) Two sheets of aluminum or aluminum alloy not less than 0.02-inch thick fastened together and having between them an asbestos paper or asbestos fabric sheet at least ⅛-inch thick.

04.6640 The firewall shall completely isolate the engine compartment and shall have all necessary openings fitted with close-fitting grommets or bushings. Adjacent inflammable structural members shall be protected by asbestos or an equivalent insulating material and provision shall be made for preventing fuel and oil from permeating it.

04.665 Heating systems. Heating systems involving the passage of cabin air over or in close proximity to engine exhaust manifolds shall not be used unless adequate precautions are incorporated in the design to prevent the introduction of carbon monoxide into the cabin or pilot's compartment. They shall be constructed of suitable materials, be adequately cooled and be susceptible to ready disassembly for inspection.

04.69 MISCELLANEOUS POWERPLANT REQUIREMENTS.

04.690 Materials. Fuel, oil and cooling systems shall be made of materials which, including their normal or inherent impurities, will not react chemically with any fuels, oils or liquids that are likely to be placed in them.

04.7 PERFORMANCE

04.70 PERFORMANCE REQUIREMENTS.

All airplanes shall comply with the performance requirements set forth in §§ 04.707 and 04.708. All airplanes except those certificated in the transport category shall comply with §§ 04.700 through 04.706, inclusive. Compliance with such performance requirements shall be shown in standard atmosphere, at all weights up to and including the standard weight (§ 04.102) and under all loading conditions within the center of gravity range certified (§ 04.742): *Provided*, That demonstration of compliance with landing-speed requirements, and with those relating to take-off time and distance, may be limited to an intermediate range of center of gravity positions if it can be shown that it is possible for the airplane to continue flight with one engine inoperative, and that passengers or other load can be easily and rapidly shifted while in flight to permit the realization, at the pilot's discretion, of a center of gravity position within the range covered by this demonstration. There shall be no flight or handling characteristics which, in the opinion of the Administrator, render the airplane unairworthy.

The phrase, "possible for the airplane to continue flight with one engine inoperative", is interpreted to mean that the best rate of climb with one engine inoperative shall be not less than 200 feet per minute at sea level in standard air. This is to be determined with the inoperative propeller in the minimum drag condition, the airplane at standard weight with flaps and gear retracted, and with maximum except take-off power on the remaining engine(s).

The phrase "passengers or other load can be easily and rapidly shifted while in flight", is interpreted to mean that load other than passengers should be capable of being as easily moved as are passengers. That is, it should require no more effort nor attention, on the part of the crew, than that involved in effecting the removal of the necessary number of passengers from seats, which they may occupy, to those necessary to be occupied in order to bring the center of gravity within the required limits. However, no means by which this can be so accomplished is known to be in use at present. "Other Load" has been included in order to take care of possible future development.

CG Positions

a. The applicant should test the aircraft for the most forward and most rearward *CG* positions at which it will meet the minimum flight requirements and still not exceed the limits for which the structure has been investigated. This will avoid the necessity for additional flight testing in the event of subsequent alterations. It is important, however, that the most rearward *CG* position is not realized at the expense of good stall characteristics. This is particularly true of aircraft in the lower weight class which are generally flown by comparatively inexperienced pilots.

b. In all cases where *controls are arranged in tandem* and the aircraft can be flown from either position, the loading must be such as to simulate a condition with the pilot in the front seat in the tests of most forward *CG* condition and in the rear seat in the rearward *CG* condition, otherwise, a suitable placard must be installed restricting solo flights to one seat.

c. Compliance throughout the range of weight and *CG* location specified in 04.70 is ordinarily considered to be established by a showing of compliance at "critical" values of weight and *CG* position. The validity of this procedure depends of course upon the certainty with which "critical" conditions can be selected.

Flight Testing Methods (General)

a. Prior to conducting any official flight test, the *loading* of the airplane should be verified, i. e., check to see that the proper amounts of fuel and ballast (if required) are installed to obtain the desired *CG* position. Further, all ballast should be securely fastened and spin chutes, droppable ballast containers, emergency door removal mechanism, etc., should work satisfactorily.

b. Since certain of the flight characteristics are materially affected by *weather conditions*, care should be exercised to conduct tests under conditions which will give every assurance of the results being accurate. Smooth air is essential for the tests required for quantitative performance data, stability, trim, and stalls. With due regard for safety, suitably smooth air should be sought as near as possible to the pressure altitude at which the engine will deliver its rated METO power. Spin recoveries are not so readily effected in air of low density as in air of high density. Therefore, when an airplane has marginal spin recovery in air above 10,000 feet pressure altitude, final decision should be reserved until verified at an altitude less than 10,000 feet.

c. Whenever practicable, it is desirable that the applicant's test pilot and/or other representative(s) be members of the *crew* during official tests. This not only facilitates the recording of accurate data, but also aids in the case of unforeseen exigencies. In all cases the equipment, instruments, etc., must be suitable and accurate for obtaining the data desired.

CHART FOR DETERMINATION OF DENSITY AND EQUIVALENT ALTITUDES

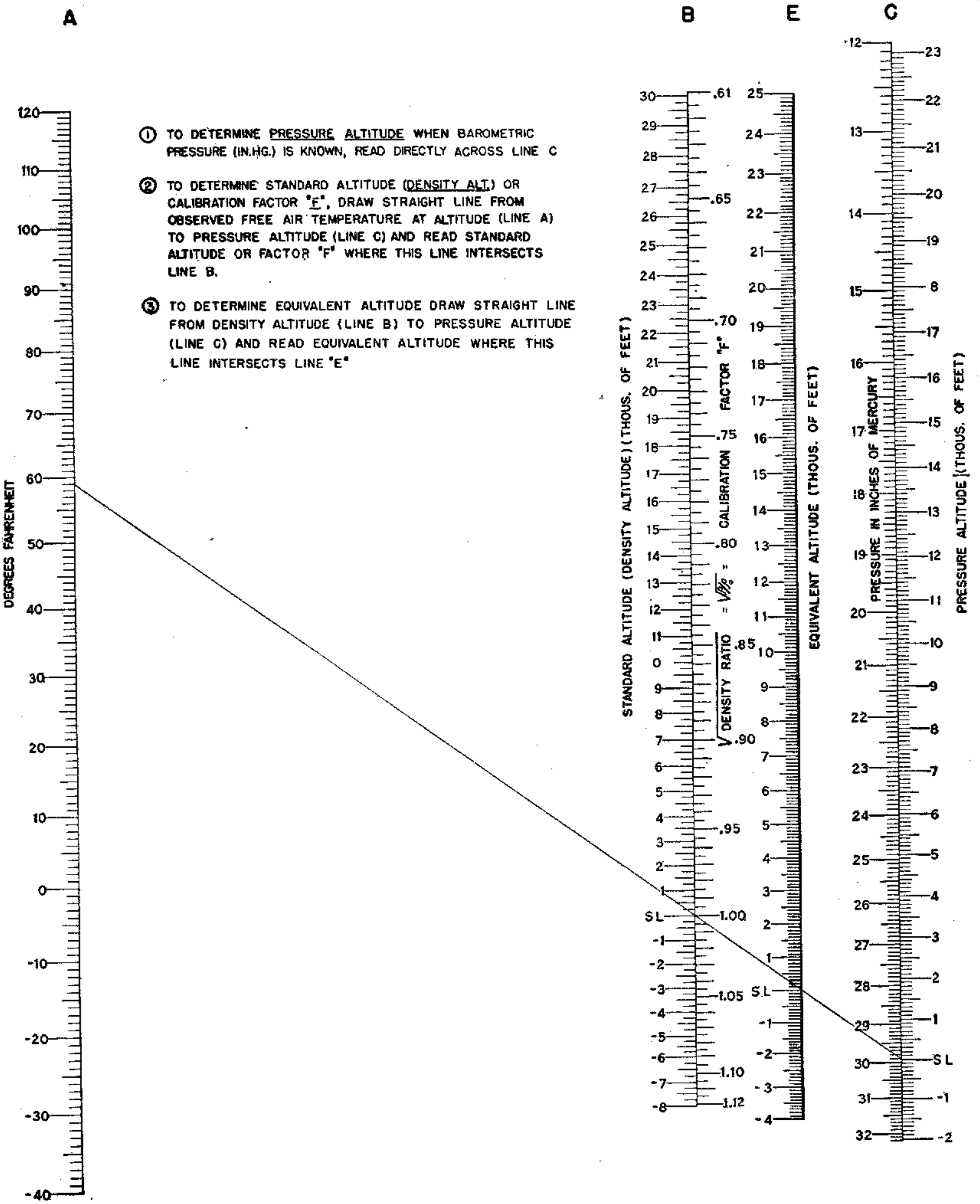


Figure 46.

d. Since the performance tests to demonstrate compliance are dependent upon the accuracy of the instrument installations, the following suggestions are offered regarding the most commonly used instruments:

Airspeed indicator.—The airspeed indicator must be so installed that errors due to change of attitude and cabin or cockpit pressures are not abnormal. The practice of venting the static line at the cabin or cockpit has been generally unsatisfactory. Such installations must be investigated for the effect of moderate slips and cabin ventilators or windows opened to various degrees. The instrument must be calibrated in all cases where accurate values of speed are required. (Also see page 119 for acceptable methods of calibration.)

Altimeter.—A sensitive type altimeter, adjustable for changes of barometric pressure, must be installed during all performance tests. This instrument should be vented to the airspeed indicator static line unless another arrangement is proved to be equally satisfactory. All altitudes reported should be in terms of "pressure altitude". This can be accomplished by setting the barometric pressure scale of the instrument to 29.92 inches prior to taking readings. The following method may be used in cases where a barometric pressure scale is not provided or when the scale is suspected of being in error: Obtain from the local weather bureau station, or other reliable source, the *surface* pressure at the airport immediately prior to the take-off. The corresponding *pressure altitude* of the airport may then be obtained from standard altitude table, or from figure 46. Adjust the altitude scale to correspond to the existing pressure altitude on the ground. All indications will then be in terms of pressure altitude without further corrections for changes of barometric pressure.

Rate of climb indicator.—This instrument has been found to be generally unreliable for use in measuring accurately rate of ascent or descent due to its inherent lag characteristics. It is of some value in trimming the aircraft for level flight and in providing approximate values of vertical speed. Best results will usually be obtained from this instrument by venting it to the airspeed static line.

Pressure gauges.—Such instruments as manifold, fuel and oil pressure gauges have been proven generally satisfactory unless subjected to pressure above the normal range, or otherwise abused.

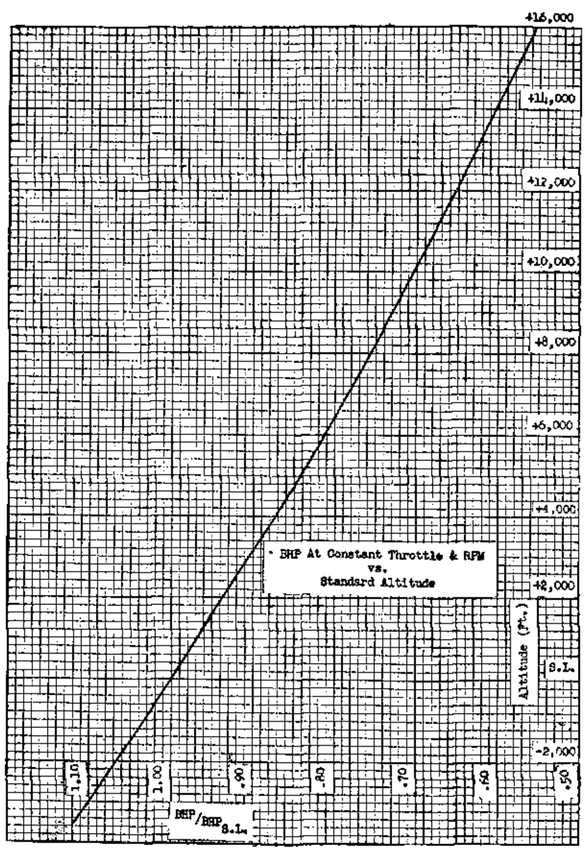


Figure 47.

Temperature gauges.—Thermometers used in measuring outside air temperature during performance and cooling tests must be so installed that the expansion elements are not affected by engine heat. Readings on the ground will usually not indicate true atmospheric temperatures unless taken immediately following a landing.

Thermocouple indicator installations.—Thermocouple indicators for cylinder head and barrel temperature readings must be calibrated at frequent intervals. The thermocouple elements must be firmly fastened to the engine to insure accurate results. A loose spark plug may introduce errors as high as 50° F in head temperature readings.

Oil temperature indicators.—Oil temperature indicators of the standard bulb type are generally reliable. The dial on test instruments must be graduated so as to be easily readable to within 5° F.

04.700 Landing speeds. The landing speed with power off, in standard calm air at sea level, shall not exceed a value determined as follows:

(*a*) Airplanes certificated for passenger carrying: 65 miles per hour for airplanes of 20,000 pounds *standard* weight or less, 70 miles per hour for airplanes of 30,000 pounds *standard* weight or more, and a linear variation with *standard* weight shall apply for airplanes between 20,000 and 30,000 pounds.

(*b*) Airplanes which are certificated for the carriage of goods only: The above landing speed values may be increased 5 miles per hour.

Critical Weight and *CG* Position

The "critical" weight for landing is of course the maximum weight.

The "critical" *CG* location is almost always foremost.

Marginal Cases

If the landing speed is within 5 mph of the maximum permissible, at least three trials will be made during official tests, and the average speed of three consecutive landings will be used to determine compliance with the regulation. In such marginal cases and in any case where there is uncertainty as to the accuracy of the airspeed indicator at low speeds, the applicant may be requested to provide for a more accurate determination of the speed than may be made by means of the indicator such as, for example, photographic means. Measurement of the landing roll is desirable.

Acceptable Method

a. The landing speed should be measured during a normal approach and landing with power off (throttle(s) closed) and should be the minimum speed obtainable under such conditions. Flaps, if present, should be in the position intended for landing. The propeller, if controllable, should be placed in low pitch during the approach and left in this position thereafter until the landing is complete. If the landing roll is also to be measured, the most vigorous safe brake application should be made during the roll. The airspeed indicator should be tapped lightly prior to contact in order to reduce any lag in the instrument due to friction.

b. The airspeed indicator should be read during the approach glide and at the instant of contact. The anemometer should be located near the runway, at approximately the point where the airplane is brought to a stop and at the approximate height of the airplane wing(s) and should be read from the time the airplane is approximately 50 feet above the runway until the airplane is stopped. The actual flap position used should be noted.

c. If the flap retraction or extension time is less than the 15 seconds required by 04.434, the behavior of the airplane during flap retraction should be investigated by putting the airplane in a simulated approach glide, with flaps extended and throttles closed, at the minimum speed recommended for approach and initiating flap retraction. The airplane should be easily and satisfactorily controllable with one hand and appreciable loss of altitude avoided by using the throttle(s).

Corrections of Flight Data

a. If the landing speed has been measured by means of the airspeed indicator, no correction beyond that involved in the calibration need be made since the true indicated airspeed is the true airspeed at sea level standard air. If the speed has been measured by other means involving the separate measurement of ground speed and wind velocity, the observed airspeed (ground speed ± wind velocity) is the true airspeed under the test conditions and must be corrected to true indicated airspeed by multiplying it by the factor "F" read from figure 46 precisely as "average speed" is corrected to the true indicated airspeed in the airspeed calibration.

b. The landing roll must be corrected for both wind and altitude (density) in exactly the same manner as in the case of the take-off run (see section IV, D of Form ACA 283, "Type Inspection Report."), i. e.:

Wind.—K_w is read from figure 48 at the value of V_w/V_{Lnd_s} appropriate to the test conditions, where V_{Lnd_t} is the true airspeed under the test conditions (true indicated landing speed divided by "F" from figure 46).

Density.—K_D is read from figure 49 at h_D of the test.

Summary.—The corrected landing roll under calm sea level standard air conditions is: Observed Roll $\times K_W \times K_D$.

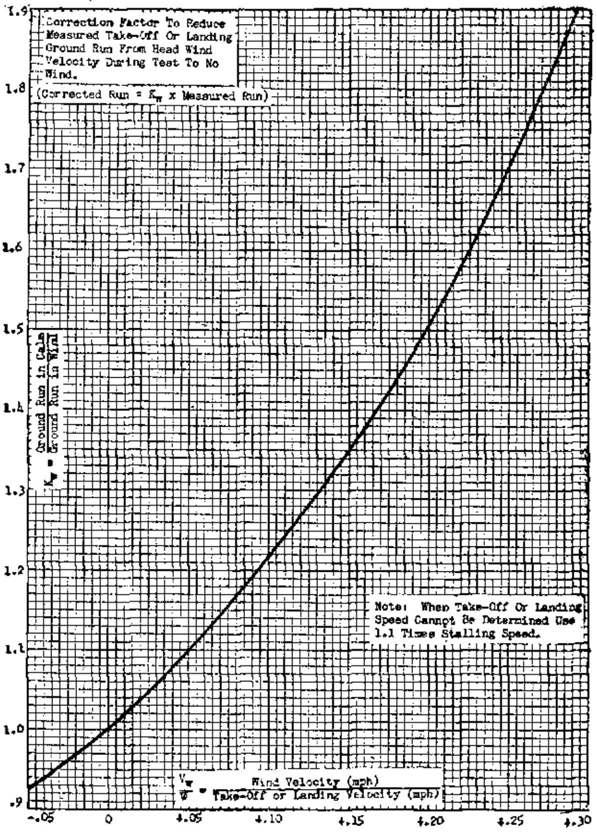


Figure 48.

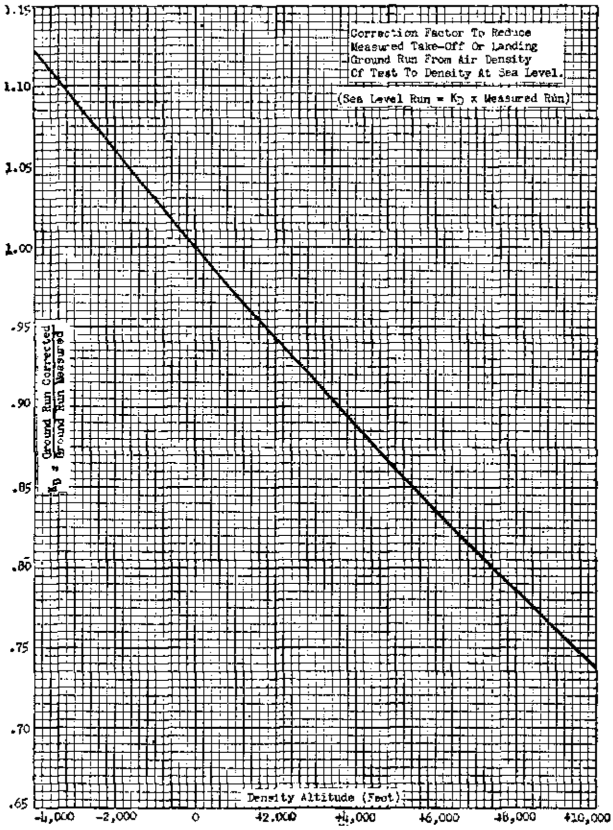


Figure 49.

04.701 Take-off. Take-off at sea level: (*a*) Within 1,000 feet for landplanes; (*b*) Within 60 seconds in calm air for seaplanes.

General

When wing flaps are used to demonstrate compliance with the take-off requirement, the airplane shall be so placarded as to inform the pilot of the flap setting necessary to realize the take-off distance demonstrated.

Critical Weight and *CG* Position

In the case of take-off distance, there appears little question that the maximum weight is critical. The critical *CG* position is not so certain but, in the absence of evidence to the contrary, it may be assumed to be foremost for both tail and nose wheel landing gears.

Marginal Cases

If the take-off distance is in excess of 900 feet, at least three trials will be made during official tests, and the average corrected result will determine compliance.

Acceptable Method

a. It is desirable that the *wind* velocity during the test does not exceed 5 mph. The anemometer should be at approximately the height of the airplane wing when the airplane is on the ground and should be located as near the runway of take-off as practicable and just beyond the anticipated point of take-off. The anemometer reading should be begun as the take-off starts and completed just before the airplane passes the anemometer station in order to avoid any possible influence of the slip stream upon the reading.

b. The tests should be conducted on a smooth level and firm surface for land planes; and for seaplanes, the water surface should be reasonably smooth. During official tests the flight engineering inspector will ordinarily ride in the airplane, observe and record the test data. In the case of land planes, three other observers are desirable, one to read the anemometer, thermometer, and barometer and the other two to observe the point at which the airplane leaves the take-off surface and to determine the distance from the start of the take-off to this point.

c. The full throttle static rpm should be measured, with the engine thoroughly warm and the airplane on the ground headed 90° from the direction of any wind, by opening the throttle wide, waiting for conditions to stabilize and reading the tachometer. The propeller static rpm is required in order to establish the propeller limits. See "Propellers," page 123.

Corrections of Flight Data

Corrections for Landplane Tests.—In order to obtain the take-off distance under sea level calm standard air conditions, it is almost always necessary to apply corrections to the measured run to reduce it from the conditions existing during the actual testing to these standard conditions. These necessary corrections are discussed below.

Wind.—The take-off ground run is the distance required to accelerate the airplane from zero ground speed to the take-off airspeed and is, for a given airplane, approximately proportional to the difference between the squares of the take-off ground speed and the initial ground speed. In the presence of a head wind component, the take-off ground speed is equal to take-off airspeed minus the velocity of the wind component. The initial ground speed is unaffected by wind and is zero. It follows that a head wind component produces a shorter take-off ground run or conversely, if the distance is measured in a head wind component, the distance in a calm will be greater.

The magnitude of the necessary correction depends upon the ratio of the velocity of the wind component along the runway to the take-off airspeed and the correction factor, K_w, by which the observed distance must be multiplied to obtain the corrected distance in a calm, may be read directly from figure 48 at the appropriate value of V_w/V_{T-o}. It should be noted that positive (+) values of this ratio correspond with head wind components and negative (−) values with tail components. Also that V_w is the component along the runway of the total wind velocity, i. e., it is the wind velocity times the cosine of the angle between the wind direction and the runway. It is not the wind velocity unless the wind blows directly along the runway. Finally, it should be noted that V_{T-o} is the true indicated airspeed under the test conditions.

Density (altitude).—Aside from its effect upon power which in turn affects the take-off distance and which is separately treated hereunder, the effect of altitude upon the take-off is to increase the take-off airspeed (true speed) which increases the distance approximately as the square of this speed. Since the indicated minimum take-off airspeed is independent of altitude, the actual speed increases with increasing altitude as $\sqrt{\rho_o/\rho}$ and the distance as ρ_o/ρ. The sea level take-off distance is then the observed distance @ ρ multiplied by ρ/ρ_o. Figure 49 is a plot of K_D $(=\rho/\rho_o)$ against the "density" altitude of the test. The "density" altitude may be obtained from figure 46 by connecting the observed "pressure" altitude on the "C" scale by a straight line with the observed temperature on the "A" scale and reading the intersection of this line with the "B" scale.

Power.—Since power, at a given throttle setting and rpm, decreases with altitude above the critical altitude of the engine, the take-off distance is correspondingly increased if the take-off is made above the critical altitude. Figure 50 is a plot of the factor, K_p, by which the take-off distance observed under such conditions must be multiplied in order to obtain the corresponding distance at sea level, against the "pressure" altitude of the test. As is indicated by the figure, this factor depends upon the take-off power loading, i. e., the weight of the airplane during the test divided by the (sum of the) rated take-off power(s) of the engine(s). It should be noted that:

(*a*) If the altitude of the test is less than the critical altitude for take-off power of the engine, no correction for power is necessary.

(*b*) If the altitude of the test is greater than the critical altitude for take-off power, the observed distance is to be multiplied by the ratio of K_p at the altitude of the test to K_p at the critical altitude, and

(*c*) For a sea level engine, correction is always necessary unless the test is conducted at precisely sea level, in which case $K_p=1$.

It should also be noted that this correction is based upon the variation of power with altitude in standard air and does not therefore account for departure of air temperature from standard values.

Outside air temperature.—The effect of carburetor inlet air temperature upon engine power is such that:

$\frac{BHP}{BHP_s}=\sqrt{\frac{T_s}{T}}$, where T, T_s=absolute temperature, i. e., power decreases with increasing temperature. It follows that take-off distance increases with increasing temperature or that a take-off distance measured under temperatures higher than standard must be reduced to obtain the distance corresponding with the standard temperature.

Figure 51 is a plot of the temperature correction factor K_t by which the observed distance must be multiplied to accomplish this reduction. It should be noted that the curve for each "pressure" altitude intersects the line K_t=1.00 at the standard temperature for that altitude.

Summary.—The finally corrected take-off distance which must not exceed 1,000 feet is then: Observed Distance $\times K_w \times K_D \times K_p \times K_t$.

Corrections for seaplane tests.—It is also necessary that take-off time for seaplanes be corrected for wind and temperature and factors for this purpose will be furnished later.

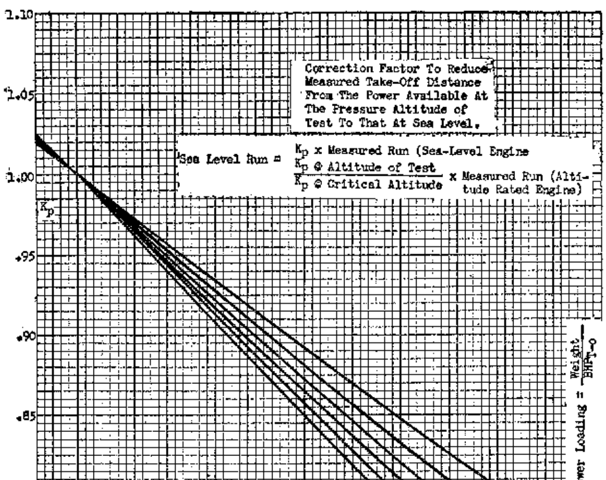


Figure 50.

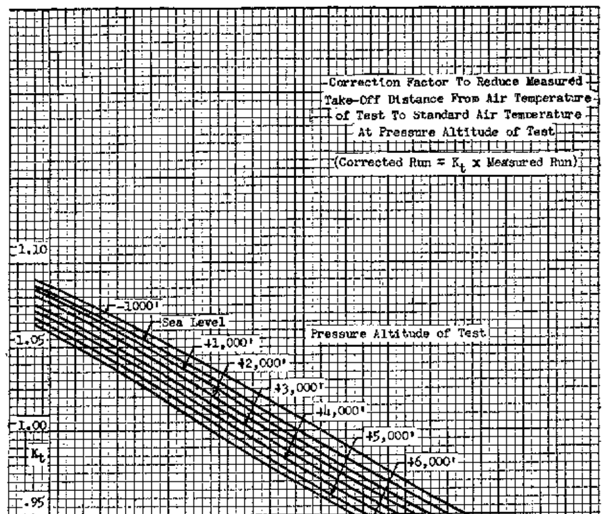


Figure 51.

04.702 Climb. The average rate of climb for the first minute after the airplane leaves the take-off surface in accordance with § 04.701, and the rate of steady climb at sea levels with not more than maximum except take-off power, shall not be less in feet per minute than: (*a*) Landplanes: Eight times the measured power-off stalling speed in miles per hour with the flaps and landing gear retracted, or 300 feet per minute, whichever is greater; (*b*) Seaplanes: Six times the measured power-off stalling speed in miles per hour with the flaps retracted, or 250 feet per minute, whichever is greater.

1. Best Rate of Climb—General

a. The purposes of testing for determination of compliance with 04.702 are as follows:

(1) To determine the best rate of climb speed which must not be exceeded during the cooling test required by 04.640 nor during the first-minute climb test required by 04.702.

(2) To determine the best rate of climb with all engines operating at METO power at on

altitude near sea level which, when corrected to sea level and standard conditions, must exceed the minimum required by 04.702.

(3) The absolute ceiling and corresponding speed or alternatively, the variation of the best rate of climb and corresponding speed with altitude over a range of altitude great enough to enable the best rate of climb (see item 2 above) and the first-minute climb (see item 1 above) to be predicted under standard conditions at sea level and to indicate the speeds to be used at altitude during the cooling test.

b. In order to accomplish these purposes, it is necessary that sawtooth climbs be conducted. For the first two of these purposes, a single, five speed sawtooth near sea level will ordinarily suffice. For the third purpose, two or more additional five speed sawtooth climbs are necessary depending upon the type of engine installed. For a sea level engine, two such additional sawtooth climbs should be conducted at higher altitudes at intervals of altitude equal to about one third of the estimated ceiling. For altitude rated engines, a sufficient number of such additional sawtooth climbs should be conducted at such altitudes as to determine the slope of the best rate of climb versus altitude curve on either side of each critical altitude as well as such testing as is necessary to determine the actual values of these critical altitudes for the engine(s) as installed. For a single stage supercharger for example, one additional sawtooth below the critical altitude and three above will ordinarily be necessary.

c. Reference is made to Flight Engineering Report No. 3, "Airplane Climb Performance," for a discussion of the nature of climb performance and the various factors affecting it.

2. Best Rate of Climb—Acceptable Method

a. For sawtooths, the airspeeds selected should bracket the speed for best rate of climb which, for preliminary purposes, may be estimated as 140 percent of the power-off stalling speed. The lowest speed should be as near the stalling speed as it is considered safe to fly. The interval between speeds should be smaller at the low speed end of the range, where it should normally not exceed 5 mph, and should increase with increase in speed. Thus, for example, for an airplane having a stalling speed of 60 mph, 65, 70, 80, 90, and 105 mph would be a good selection.

b. In order to obtain usable results, it is essential that these tests be conducted in smooth air. Even slight turbulence may produce such errors in the observed climb as to render the data inconclusive both in respect of the best rate of climb and of the speed for best rate. The observed rates when plotted should define a curve of the general shape of that in figure 10 of Flight Engineering Report No. 3. If they fail of this they are probably inadequate. Each climb should be started at an altitude far enough below the chosen altitude to permit speed and engine rpm to become stabilized. Each climb should be continued five minutes or through 2,000 feet of altitude, whichever occurs first. It is preferable to observe and report pressure altitude at not less than one-minute intervals throughout the steady portion of the climb.

3. Best Rate of Climb—Corrections

a. If a satisfactory set of data have been obtained, the speed for best rate of climb may be approximated closely enough by plotting these observed data. The corrections necessary to determine the actual rate of climb are described in Flight Engineering Report No. 3.

4. First-Minute Climb—General

a. The first-minute climb must be measured during a continuation of the test conducted to establish the take-off distance. That is, it is the gain in altitude during the minute beginning at the time of leaving the ground during the take-off test. The airspeed during this minute should approach as closely as possible but should at no point exceed the speed for best rate of climb determined in accordance with paragraphs 1, 2, and 3 immediately above. If there is a time limitation on the use of take-off power from the engine, this limitation must be observed by throttling back to METO power at the expiration of the time from the start of the *take-off*. If, for example, the limit is one minute and the take-off has taken 20 seconds, the climb should be continued 40 seconds at take-off power and thereafter at METO power.

b. For the purpose of demonstrating compliance with this requirement, a fixed or adjustable pitch propeller must be of such pitch that, at the maximum permissible throttle setting, 95 percent of METO rpm is not exceeded if the engine is a sea level engine rated 100 bhp (METO) or less or METO rpm for sea level engines of higher power. The low pitch setting for a controllable pitch propeller must be such that take-off rpm is not exceeded. For constant speed propellers neither take-off rpm nor mp shall be exceeded.

c. If the first-minute climb is less than 50 feet per minute in excess of that required on official tests, not less than three take-offs and climbs will be made and the average result used for determining compliance with 04.702.

d. The weight and *CG* location will, of course, be those selected for the take-off test. (See page 131.)

e. In addition to the first-minute climb, it should be noted that the rate of steady climb at sea level with not more than METO power must also meet the requirements of 04.702.

5. First-Minute Climb—Acceptable Method

a. The airplane should be so flown, within the limitations outlined above, as to gain the greatest possible altitude within the minute following leaving the ground.

b. Take-off speed should not exceed the Best Rate of Climb speed.

6. First-Minute Climb—Corrections

a. In order to establish the first-minute climb at sea level under standard air conditions, it is necessary to apply certain corrections to the observed data. These are discussed hereunder.

Outside air temperature.—The observed first-minute climb is actually an observed difference in pressure (measured by means of the altimeter). The corresponding difference in altitude depends upon the temperature of the air, being greater the higher the tempreture, and is directly proportional to the absolute temperature. For this reason the actual climb under the test conditions is the observed climb multiplied by (459+the average outside air temperature) and divided by (459+the standard temperature) at the mean "pressure" altitude of the test. This standard temperature may be read from figure 46 by placing a straight edge on the altitude on both the "B" and "C" scales and reading the temperature at the intersection of the straight edge with the "A" scale, or, it may be calculated as follows:

$$t=59-0.003566h$$

Where:

h=altitude in feet.

All temperatures are in degrees Fahrenheit.

Altitude.—The effect of altitude (temperature and pressure) upon the climbing performance is such that the rate of climb generally decreases with altitude. This is due essentially to the fact

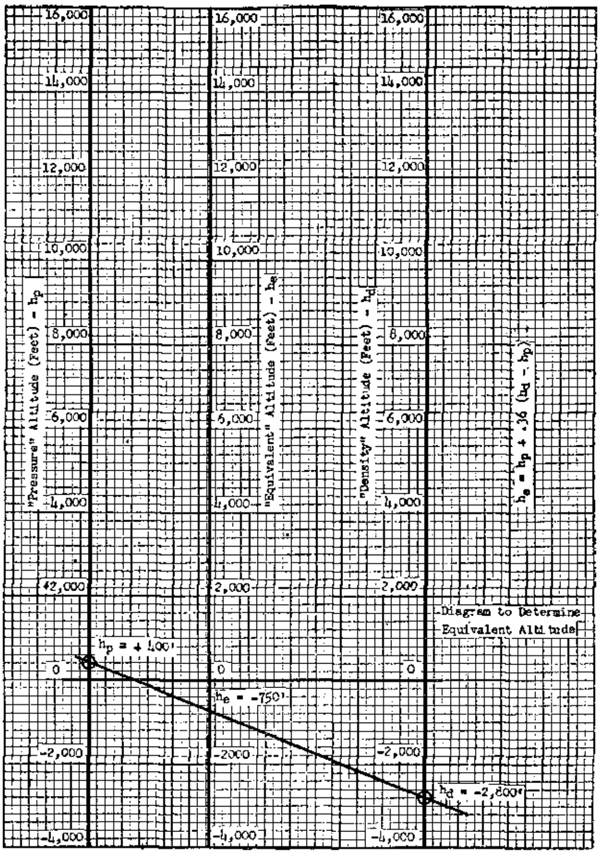


Figure 52.

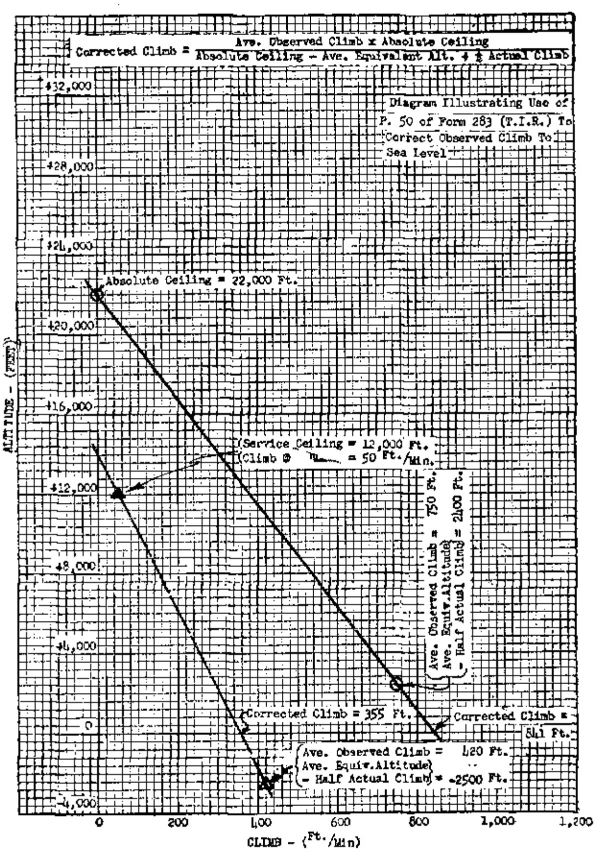


Figure 53.

that at a given true indicated airspeed the true airspeed increases which increases the power required for level flight (drag × velocity) which reduces the power available for climb and to the fact that the available power decreases with altitude.

For a sea-level engine installation.—The correction for altitude may be made by the "equivalent" altitude method. The "equivalent" altitude is the altitude in standard air at which the measured actual climb under the test conditions would be obtained under standard conditions. It is the "pressure" altitude of the test, plus 0.36 × the difference between the "density" altitude of the test and the "pressure" altitude of the test. It may be read directly from figure 52 . The actual first-minute climb at sea level in standard air may then be obtained by plotting the absolute or usable ceiling and the measured actual first-minute climb. The latter should be plotted at the mean "equivalent" altitude of the test minus half the measured actual climb. Then draw a straight line between this point and the absolute or usable ceiling point. The intersection of this line with zero altitude is the sea level first-minute climb. Figure 53 illustrates the use of the diagram for two cases.

For an altitude rated engine.—The correction process is much more complicated. If the mean "equivalent" altitude of the test is below the critical altitude of the engine, an approximate correction may be made by adding 10 feet per minute for each 1,000 feet this "equivalent" is above sea level. Otherwise correction may be made by following the procedures outlined in Flight Engineering Report No. 3.

Wind.—If, as is usually the case, there is a wind velocity gradient with height (i. e., the wind velocity increases with height), a climb made into the wind for a given airplane at a given weight, speed, and power will be a higher rate (i. e., more altitude will be gained in a given time) than the same climb in still air. This effect of a wind gradient is discussed, in respect of the glide in landing, in Flight Engineering Report No. 1 which indicates it to be proportional to wind velocity, the increase of wind velocity with altitude, and the rate of climb. In practice, it is very difficult to determine the second of these and even when it is known, the correction is rather difficult to make. For these reasons it is recommended that tests be conducted in as nearly calm conditions as possible.

04.703 Controllability and maneuverability. All airplanes shall be controllable and maneuverable under all power conditions and at all flying speeds between minimum flying speed and the maximum certified speed. All airplanes shall have control adequate for an average landing at minimum landing speed with power off.

1. Controllability—General

a. By controllability is meant the ability of the pilot through a proper manipulation of the controls to maintain or alter the attitude of the airplane with respect to its flight path. It is intended in the design of the airplane that it be possible to "control" the attitude about each of three axes, the longitudinal, the lateral, and the normal axes. Angular displacements about the longitudinal axis are called "roll." Those about the lateral axis are called "pitch" and those about the normal axis are called "yaw." Controllability of an airplane is usually rated as "good," "fair," "poor," or "unsatisfactory," in each of the various flight conditions which are investigated for each of the three loading conditions. The particular rating which is appropriate must necessarily depend to a great extent upon the judgment and previous experience of the operator. If he finds this to be as good as or better than that present in the best airplane of a similar type and size that he has flown, it may be identified as "good." If it be his opinion it is no better than that of the average airplane with which he has had experience, it should be identified as "fair." If it be typical of the poorest of type certificated airplanes with which he has had experience, it should be identified as "poor." If the controllability is so inadequate that a dangerous condition might easily occur, it should be considered "unsatisfactory" and is unacceptable as a showing of compliance with the regulations.

b. It should be noted that if a spring device is installed in the control system, the airplane must be satisfactorily controllable with and without the use of the spring device in accordance with 04.436.

2. Maneuverability—General

a. By maneuverability is meant the ability of the pilot through a proper manipulation of the controls to alter the direction of the flight path of the airplane. Obviously, in order to accomplish this, it is necessary that the airplane be controllable, since a change in attitude is necessary in order to change a direction of flight. It should also be noted that any change in the direction of flight involves an acceleration normal to the flight path.

b. Maneuverability is so closely related to controllability as to be inseparable in any real motion of the airplane. It is also similarly purely qualitative in its nature and should be treated in the same manner as has been suggested for controllability above.

c. It should be noted that if a spring device is installed in the control system, the airplane must be satisfactorily maneuverable with and without the use of the spring device in accordance with 04.436.

d. For the current conventionally designed airplanes, the following loading and flight conditions are likely to be critical with respect to controllability and maneuverability:

(1) Take-off transition from ground control to aerodynamic control for maximum load at rearward or forward *CG* for steerable wheel(s) aft or forward respectively.

(2) At or near the stall with rearward *CG* and power on.

(3) At placard *V* airspeed irrespective of power and *CG* loading.

(4) For multi-engine airplanes with power on when the critical inoperative engine(s) suddenly becomes inoperative.

3. Controllability and Maneuverability—Flight Conditions

a. 04.703 is interpreted to mean that the aircraft must be satisfactorily controllable (with and without use of spring devices) about each of its axes for any flight and power condition and with any loading condition for which approval is sought. This also includes during landings and take-offs.

b. In achieving satisfactory controllability, the applicant should attempt to so design the control surfaces that the degree of response from movement of the various surfaces is approximately equal. Further, the manual force required should neither be excessive nor too light and should be proportional for each surface to that force which can be readily exerted by an operator's arms and legs.

4. Multi-Engine Aircraft—Best Rate of Climb Airspeed

a. 04.703 is interpreted to require that it be possible to maintain the wings level during, and to recover safely from, suddenly executed changes in heading up to 15° during steady flight at the one engine inoperative best rate of climb speed, with any engine inoperative, the inoperative propeller in its minimum drag condition, with the landing gear retracted and flaps in the position used to show compliance, with the airplane loaded at the rearmost *CG* (using maximum weight obtainable at this *CG* position) for which certification is sought and with METO power on the remaining engines. This section of the regulations is further interpreted to require that it be possible to execute 15° banked turns with or against the inoperative engine, starting from steady flight at the one engine inoperative best rate of climb speed with any engine inoperative, the inoperative propeller in its minimum drag condition, with landing gear and flaps in the fully extended position and with METO power on the remaining engines, except that, if the applicant so elects, the airplane may be placarded for a flap position less than fully extended, provided only that such reduced flap extension must not be less than that used in demonstrating compliance with 04.700.

The fundamental purpose of the investigation of suddenly executed changes in heading is to insure that the rudder is unlikely to lock in the fully deflected position or become heavily overbalanced, thus producing an uncontrollable airplane.

b. Acceptable method.—The airplane should be properly loaded, taken off and climbed on all engines to the altitude at which the tests are to be conducted. This altitude should in no case be less than 1500 feet or more than 1500 feet above the lowest standard altitude above which METO power cannot be obtained by opening the throttles. The critical engine should then be made inoperative and it should be attempted to trim the airplane under the required conditions. Once the airplane has been trimmed, the behavior of the airplane following sudden changes in heading should be investigated, beginning with comparatively small changes and gradually increasing these until a maximum of 15° has been reached or dangerous flight characteristics encountered. Following this, the landing gear and flaps should be extended, the airplane retrimmed at the appropriate speed and the execution of 15° bank turns attempted.

5. Multi-Engine Aircraft—Best Angle of Climb Airspeed

a. Further, 04.703 is interpreted to require that it be possible safely to recover to straight flight with wings level in the event of the failure of any engine during steady flight at the one engine inoperative best angle of climb speed with take-off or the maximum available power on all engines, with the flaps and landing gear retracted and with the airplane loaded at the rearmost *CG* (using maximum weight obtainable at this *CG* position) for which certification is sought, without unnecessary delay and without the necessity to exert a rudder pedal force in excess of 180 pounds, or to throttle the remaining engine. The above is not intended to prohibit the momentary dropping of a wing which almost invariably follows failure of an engine on a multi-engine airplane.

b. Acceptable method.—The test should be conducted at an altitude not less than 1500 feet. During official tests, the applicant's pilot should fly the airplane with the examining inspector in the co-pilot's position. Steady flight should be established under the specified conditions, the throttles concealed from the pilot and one of them, selected by the examining inspector, should be suddenly closed. The examining inspector will thereafter observe whether or not the requirement is met. It is believed that there is little pilot reaction time involved in applying corrective rudder pressure under circumstances such as the above test involved. The inspector will observe carefully the applicant's pilot's reaction time and satisfy himself that there are no peculiarities of control or cockpit arrangement which, in operation, might tend to make the pilot's reaction time excessive. It is desirable that some visual means be provided by which the rudder pedal force may be measured. If this is impossible, the pilot and co-pilot should exchange positions and the maneuver be repeated, in order that the inspector may estimate the rudder pedal forces required to recover.

04.7030 Controllability at the stall. With power off and with 75% maximum-except-take-off power, with flaps and landing gear in any position, the airplane shall have sufficient directional and lateral control so that when the airplane is stalled, the downward pitching motion following the stall shall occur prior to any uncontrollable roll or yaw. Any such pitching motion shall not be excessive and recovery to normal flight shall be possible by normal use of the controls after the pitching motion is unmistakably developed, without excessive loss of altitude.

1. General

a. Investigation of the stalling characteristics is required, not only in order to measure the speed at which the stall occurs in terms of which various of the requirements are stated, but also in order to insure that the stalling characteristics are such that recovery to normal flight is possible with only a reasonable skill on the part of the pilot and that such recovery is possible without excessive loss of altitude, excessive speed or awkward or dangerous airplane altitudes which cannot be prevented by normal use of the controls. In order to insure that the stalling characteristics are satisfactory under any configuration in which the airplane is likely to be flown, these characteristics are to be investigated with power on and power off and with landing gear and flaps, if present, up and also in the position normally used for landing for each of the power conditions. "Power-on" is arbitrarily defined as the 75% of METO power, or the nearest possible approach to it. "Power-off" is defined in 04.704.

b. This flight condition normally is of very infrequent occurrence during the life of the airplane except possibly when the airplane is used for the purpose of pilot training. It is an accelerated rather than a steady condition and for this reason duration is without significance. The regulations are interpreted to require that the stalling characteristics of the airplane must be such that it is reasonably probable that a pilot of the degree of skill likely to be found among those subsequently to fly the airplane may safely recover from an inadvertent stall.

c. It is desirable that the airplane have satisfactory stalling characteristics with the maximum power which can be drawn. In no case will an airplane be approved which invariably starts to spin following a stall which has been approached from a normal flight attitude. It is required that the first evidence of the onset of the stall be the loss of longitudinal control; i. e., the nose should drop before a wing drops, and that there be no excessive pitch prior to recovery. Recovery to normal flight is required by normal use of the controls after pitching motion is unmistakably developed, without excessive loss of altitude.

d. The rearward *CG* loading condition is most likely to be critical with respect to the magnitude of uncontrollable roll or yaw and therefore with respect to the tendency to spin upon being stalled.

2. Acceptable Method

a. The altitude selected for the "power on" condition should be as low as is considered to be safe for this purpose. For the power off condition, any altitude may be used. Steady flight should be established at or about the chosen altitude under the specified power condition at a conveniently low speed. The speed should then be slowly reduced by pulling back on the elevator control until the stall is reached. During this approach to the stall the aileron and rudder controls should be used in a normal fashion to maintain the wings level and the initial heading. The stall speed will be that noted immediately prior to and concomitantly with the pitching motion or other outward evidence of broken air flow. In view of the rapidity with which the stall may develop and the subsequent recovery occur, it may be necessary to repeat this procedure several times for each airplane configuration and power condition, noting each of the pertinent items of the observed data in turn in order to arrive at representative values for all.

b. It should be remembered that the characteristics being investigated are those which a comparatively inexperienced pilot may encounter inadvertently and therefore no violent measures should be used in either inducing or recovering from the stall.

04.704 Balance. As used in these regulations the term "balanced" refers to steady flight in calm air without exertion of control force by the pilot or automatic pilot. Lateral and directional balance is required at cruising speed, which for this purpose shall be taken as 90 percent of the high speed in level flight. Longitudinal balance is required under the following flight conditions: (a) Power on: In level flight, at all speeds between cruising speed and a speed 20 percent in excess of stalling speed. In a climb, at maximum (except take-off) horsepower and a speed 20 percent in excess of stalling speed; (*b*) Power off: In a glide, at a speed not in excess of 140 percent of the maximum permissible landing speed or the placard speed with flaps extended, whichever is lower, under the forward center of gravity position approved with maximum authorized load and under the most forward center of gravity position approved, regardless of weight.

1. General

a. Although 04.704 is fairly self-explanatory, it is further interpreted to include balance being required for the flight and power conditions specified and with any loading condition obtainable for which approval is sought. In addition, cruising speed is arbitrarily defined as 90 percent of the top level flight speed attainable with METO power or full throttle power for sea level engines.

b. For the current conventionally designed airplanes, the following loading and flight conditions normally determine the limits of travel for the longitudinal trimming device:

Nose heavy condition.—(1) Power-off glide with maximum forward *CG* and maximum loading. (2) METO power-on climb at 120% V_s with same loading as in (1).

Tail heavy condition.—(1) Cruising power level flight with maximum rearward *CG* and maximum loading.

c. It should be noted that 04.704 requires lateral and directional balance at cruising speed for all loading conditions.

2. Acceptable Method

a. Steady linear flight should be established at the chosen altitude and with the specified flight configuration and power condition. Pertinent data should be observed and recorded after the procedure below.

b. It should then be attempted to balance the airplane by so adjusting the flight and trimming controls that no control force is required to maintain the speed and flight attitude. Since all control systems have a certain amount of friction inherent in them, it is usually necessary in order to balance the airplane to manually excite the control surfaces in small rapid oscillations which do not disturb the general trim but do permit the surfaces to adapt themselves to their trimmed position. This manual excitation should be done by manipulating one control at a time from the cockpit and adjusting the trimming device, if any, provided for the particular control being excited. By so doing, it will be possible to trim out any possible unbalance about any one axis due to some other control operating about one of the other axis. When it is apparent that the airplane is balanced or as nearly balanced for straight flight as is possible, it should be permitted to continue its flight path for an appreciable length of time in order to assure its having obtained equilibrium. (Also see 04.704 discussion).

3. Multi-Engine Airplanes

a. 04.704 is interpreted to require that it be possible to trim the airplane at the one engine inoperative best rate of climb speed with any of its engines inoperative, the inoperative propeller in the minimum drag condition, with flaps and landing gear retracted and with the remaining engines operating at METO power.

b. Further, 04.704 is interpreted to require, for airplanes with four or more engines, that it be possible to trim the airplane in straight flight at an indicated airspeed 10 mph less than the indicated maximum level flight speed possible, with all engines on the critical side of the airplane inoperative, the inoperative propellers in the minimum drag position, with flaps and gear up and with the remaining engines operating at METO power. Airplanes unable to maintain altitude under this condition at the maximum weight for which certification is sought will be tested at a sufficiently reduced weight that level flight is possible.

c. Acceptable method.—The demonstration of compliance with this requirement may necessitate some preliminary investigation in order to determine the maximum weight at which there is a speed range of at least 10 mph in level flight. Once this has been accomplished, however, the procedure may be substantially the same as that discussed under 04.703.

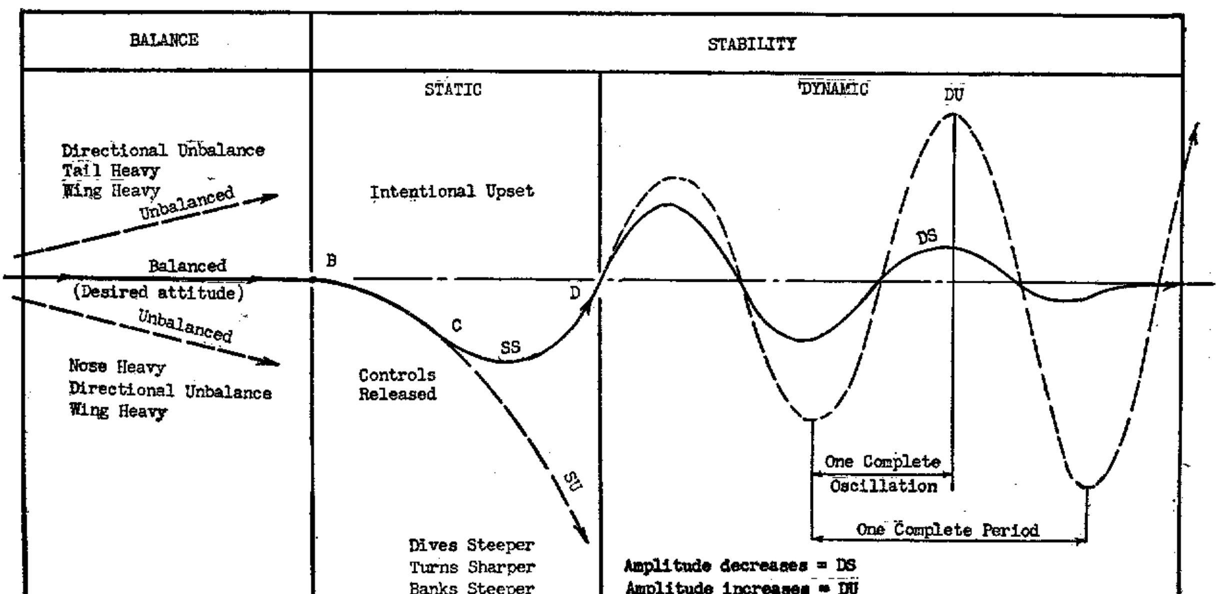
04.705 Stability. Under all power conditions all airplanes shall be longitudinally, laterally and directionally stable. An airplane will be considered to be longitudinally stable if, in stability tests, the amplitude of the oscillations decreases.

1. General

a. By stability is meant the tendency of the airplane to return to a "balanced" steady flight condition if it is disturbed either by an external force or by manipulation of the controls. Stability is also considered to be involved in the motions parallel to and about each of the three principal axes, although any real motion of the airplane following a disturbance almost invariably involves simultaneous motion about more than one of these axes and the forces set up by the displacements and velocities upon which stability depends are for any one motion to some extent influenced by all of the others. The nature of these motions is almost invariably oscillatory, i. e., a path of the motion tends to pass back and forth through a mean position or a path of the velocity back and forth through a mean value which gives rise to two types of stability. The first of these is called "static stability" which is concerned solely with the existence or absence of a tendency for the airplane to return toward the initial condition without regard to any subsequent oscillation. Thus, for example, an airplane trimmed for and flown steadily in level flight is said to be statically stable longitudinally if, when the speed is reduced by pulling on the elevator control and the control then released, the speed begins to increase toward the trim value. If, on the other hand, the speed continues to decrease after releasing the control, the airplane is said to be "statically unstable." Assuming the above airplane to be "statically stable" the speed will ordinarily continue to increase beyond the initial trim value until some maximum is reached, whence it will again decrease toward the initial value, but will continue through this to a lower value, etc. The airplane is said to be "dynamically stable" if succeeding maxima and minima of velocities depart less and less from the initial trim value, because, under these conditions, the motion will eventually be "damped out" and the airplane will return to equilibrium in the initial trim condition. If, on the other hand, these maxima and minima tend to depart further and further from the initial trim value the airplane is said to be "dynamically unstable" and the motion, if allowed to continue, will eventually result in stalling the airplane or reaching an excessive velocity, either of which is obviously undesirable.

b. The above discussion has been based upon what is called "longitudinal stability," i. e., the tendency of the longitudinal axis to return to its initial attitude. There are also "lateral" stability involving the rolling attitude of the lateral axis and "directional" stability involving the yawing attitude of the longitudinal and lateral axes. Whereas it is possible for the motion involved in "longitudinal" stability to exist without those involved in "lateral" and "directional" stability, it is not possible for either of the latter two to exist alone, since the airplane invariably rolls if it yaws or vice versa.

c. The nature of stability may be further described by referring to figure 54. Suppose, for example, an airplane is flying along path A–B, where the ordinate represents altitude. It is nosed down (or up) at B and the force of the control column is released at point C. If the aircraft follows path SS, it is statically stable longitudinally. If it follows path SU, it is statically unstable longitudinally. The figure may also be used to illustrate directional static stability by assuming that it represents the paths of an aircraft as seen from above. Likewise for lateral static stability, it may represent an airplane the wings of which either remain in a banked position, or bank further,



SS = Statically Stable DS = Dynamically Stable

SU = Statically Unstable DU = Dynamically Unstable

Figure 54.

or change in direction of bank. Another indication of static stability is to be found in the nature of the cockpit control forces. This is discussed below.

d. The question of dynamic stability is pertinent *only* when the airplane is statically stable as shown in the illustration. If from point D the airplane returns to its original attitude in oscillations the amplitude of which decrease, as shown by the line DS, it is dynamically stable. If the amplitude of the oscillations increases, as shown by the line DU, the airplane is dynamically unstable. Similarly, the figure may be applied to directional and lateral dynamic stability if the airplane is viewed from above or from the rear respectively.

e. The minimum acceptable evidence that dynamic stability exists is that the amplitude of the oscillation shall decrease within not more than three complete cycles following the initial disturbance. In the case of longitudinal dynamic stability, for example, the third maximum or minimum of velocity shall depart less from the initial speed than did those which preceded it. In the case of directional and lateral stability, there is frequently no oscillatory motion involved following a disturbance of the airplane attitude. In such case, if the airplane returns to its original attitude, the motion is said to be "dead beat." The degree of stability present is indicated by the rapidity with which the airplane recovers its original attitude. The minimum acceptable evidence that directional stability exists for such cases will be that the airplane shall cease to skid following a yawing disturbance. For lateral stability it will be that the angle of bank shall decrease following a rolling displacement of the airplane.

f. It may not be possible to balance the airplane in level flight using METO power (or full throttle). In such a case, it must be balanceable in a climb using METO power, but at a speed which is 90 percent of the top level flight speed attained above. It may be impossible to balance directionally in level flight, in which case the heading may be maintained by use of the rudder during the investigation of the stability. If balanceable in level flight, but unstable when so balanced, the airplane must be stable when balanced in climbing flight using METO power (or full throttle) at 90 percent of the level flight speed with METO power, and the range of the longitudinal trimming device must be limited to the position required for such balance. This limitation should be imposed by stops which positively prevent the trimming device (elevator tab, for example) exceeding the limiting deflection.

g. For the current conventionally designed airplanes, the following loading and flight conditions are likely to be critical with respect to stability: (1) Maximum load at the forward *CG* with METO power in level flight for high wing aircraft. (2) Maximum load at the rearward *CG* with METO power in a climb at 120% V_s.

2. Acceptable Methods

a. Longitudinal.—Trim the airplane for the configuration, flight and power conditions being investigated (see 04.704). When the airplane has been so trimmed, the longitudinal dynamic *stability* should be investigated. This is done by pulling the control column back until the speed has been reduced by approximately 20 percent of the trim speed, releasing the control column, and observing the subsequent behavior of the airplane. If necessary, following the release of the control, the rudder should be used to maintain the original heading. If during any part of the test the speed increases so that it is apparent that it will exceed the placard V_g, the airplane should be pulled out very gradually and the test re-run with a somewhat smaller decrease in speed during the pull up. This procedure should be continued until it is ascertained that the placard V_g will not be exceeded or that the airplane is unstable. To be acceptable, the amplitude of the oscillations shall show a decrease within three complete periods.

b. Directional.—Directional stability may be investigated in two ways, namely: (1) Starting from balanced level flight, deflect the rudder at a fairly rapid rate the amount required to maintain a steady skid with the airplane yawed approximately 20° from its flight path, the wings maintained level by use of the ailerons, and the speed held constant by means of the elevator control or, if the rudder pedal force required to accomplish this exceeds approximately 100 pounds, the angle of yaw should correspond with the maintenance of 100 pounds rudder pedal force. When the steady condition has been established, the rudder control should be released and, if the airplane be directionally stable, it will cease to skid, i. e., the yaw will decrease to approximately zero and, if also laterally stable, the aileron deflection and force required to hold the wings level will also approach zero. The test should be conducted by executing skids both to right and left, recording in each case the time required from the release of the rudder controls and the number of oscillations, if any, involved to recover to steady level flight. (2) Starting from steady curvilinear flight in a turn that employs approximately a 20° bank, the feet should be removed from the rudder controls and the wings leveled by use of the ailerons. If stable, the airplane will cease to execute a skidding turn, and aileron forces will decrease to approximately zero.

c. Lateral.—Lateral stability may also be investigated in two ways, namely: (1) Starting from balanced flight, by banking the airplane approximately 20° by means of the ailerons, maintaining the original heading by means of the rudder, and the original speed by means of the longitudinal trimming device. When the steady slipping condition has been established, the aileron control should be released or, in the case that the control system friction is so great that the ailerons are not free to assume their trail position, the ailerons should be slowly returned to their neutral position. During such tests if it becomes necessary to retrim in order to maintain indicated airspeed at which the test was started, use should be made of the horizontal trim control and not the elevators directly. If the airplane is laterally stable, it will cease to slip, i. e., the wings will return to an approximately level attitude and the rudder deflection and pedal force required to maintain the heading will approach zero. The test should be conducted by executing slips both to right and left, recording in each case the time required from the release of the aileron control and the number of oscillations, if any, involved to recover to steady level flight. (2) Starting from steady curvilinear flight in a turn that employs a 20° bank, the aileron control should be released (aided to neutral if friction involved) and rudder applied to hold heading on a linear flight path while using trim device to maintain trimmed airspeed. If stable, the airplane will cease to execute a slip and wings will level out with forces on rudder decreasing to approximately zero.

d. Take-off.—If the airplane has been found to be dynamically longitudinally unstable when trimmed at 120 percent of the power-off stalling speed and the applicant attempts to show, in lieu thereof, satisfactory static and dynamic longitudinal stability by the alternate method described immediately below in paragraph 3, consideration should also be given to the status of stability during the take-off. While there are no specific requirements concerning stability during this accelerated flight condition, it is suggested that in cases such as these, particular attention be paid to any behavior which might indicate the presence of static instability during this maneuver. Sharp reversals in elevator control force not associated with acceleration through a trim speed, application of power or retraction of landing gear or flaps are considered unsatisfactory if the forces are of such magnitude to dangerously affect controllability during the take-off run and initial acceleration in flight thereafter.

3. Alternate Method—Climb Only

Longitudinal stability in a climb with METO power at 120 percent V_s. (1) In the event that the airplane is dynamically longitudinally unstable in this flight condition (i. e., trimmed at 120 percent of the power-off stalling speed), it may nevertheless be certificated as airworthy if it can be shown by an alternate method to be statically and dynamically stable when balanced at the best rate of climb speed and with METO power. Acceptable evidence of the existence of static longitudinal stability will consist of measured stick forces required to maintain various speeds between the speed for best rate of climb at which the airplane must be trimmed and the stalling speed. The magnitude of these measured forces must be such as to increase steadily with departure from the trimmed speed at any speed between the trim speed and 120 percent of the measured power-off stalling speed, rapidly enough to render easily perceptible to the pilot any change in speed through a change in the control forces. There shall also be no reversal (i. e., require forward push) in the control forces at any speed below 120 percent V_s until the stalling speed is reached; i. e., the control force shall in no case fall below zero before the stall is reached. (2) If the airplane has been found dynamically longitudinally unstable for this particular flight condition using the method described above in paragraph 2*a*, the *static and dynamic longitudinal stability* should be investigated by the alternate method of balancing the airplane longitudinally at the speed for best rate of climb, conducting dynamic stability tests, and thereafter measuring the elevator control force required to maintain various lower speeds while climbing steadily through a chosen altitude. The number of speeds to be investigated for static longitudinal stability should in no case be less than four between the trimmed speed and the stalling speed and preferably more. A control force may be measured with an ordinary spring scale and care should be taken before measuring these forces to eliminate any possible effect of friction in the elevator control system by moving the control back and forth slightly through the nominal position associated with the speed which is being maintained. A satisfactory set of data for static longitudinal stability will, when plotted as measured force against airspeed, produce a smooth curve indicating a maximum at or below 120 percent V_s.

04.706 Spinning. (Not applicable to airplanes certificated in the transport category.) At any permissible combination of weight and center of gravity position obtainable with all or part of the design useful load, there shall be no excessive reversal of control forces during any possible spinning up to 6 turns. It shall be possible promptly to recover at any point in the spinning described above by using the controls in a normal manner for that purpose and without exceeding either the limiting airspeed or the limit design normal acceleration for the airplane. It shall not be possible to obtain uncontrollable spins by means of any possible use of the controls: *Provided*, That compliance with the foregoing requirements with respect to spinning shall not be required for those airplanes—(*a*) permanently placarded "intentional spinning prohibited"; or (*b*) demonstrated to the satisfaction of the Administrator to be characteristically incapable of spinning.

1. General

a. Spinning is a very complicated motion wherein the airplane describes a helical flight path about a vertical axis. The airplane is usually stalled, its wings approximately level, and the longitudinal axis down by the nose by an appreciable angle. The spinning characteristics of an airplane manifest themselves in the behavior of the airplane upon entry into, during, and upon recovery from this steady spinning motion.

b. Systematic investigations indicate that with increasing the weight of a particular airplane, the spins tend to become flatter, the rate of descent higher, the recovery slower, and the loss of altitude greater during recovery. Also as the center of gravity is moved to the rear, the spins tend to become flatter and recoveries slower.

c. As mentioned above, the flight characteristics are influenced by the weight and the *CG* position at which the airplane is flown and, for a given combination of these, by the speed and the amount of power being drawn. It is the intent of the above regulations that the flight characteristics be satisfactory throughout the range of these variables, which may be expected to be used during the life of the airplane. The magnitude of static longitudinal stability, for example, tends to decrease with rearward movement of the *CG*, with reduction in speed, and with increase in power.

d. The fundamental purpose of the spinning requirements is to insure that any airplane likely during its life to be spun, shall have characteristics such that, in order to spin steadily following a normal entry into the spin, the elevator and rudder controls must be held in the full pro-spin position. In order that this be so, it is obviously necessary that no "back pressure" on either rudder or elevator control may exist and that, if these controls be released, they must move toward their trimmed position and the steady spinning motion of the airplane stop fairly promptly thereafter.

e. Another purpose of these requirements is to insure that for any such airplane (i. e., one likely to be spun), if it be possible to obtain an "uncontrollable" spin or a steady spin with an "abnormal" use of any of the controls, it is nevertheless possible to recover safely from such spin. A still further purpose is to insure that in any case, recovery to normal flight following stopping the spinning motion is possible without attaining dangerously high speeds or high accelerations and without excessive loss of altitude.

f. Spin strips or their equivalent are sometimes employed to obtain the desired spinning characteristics. Service records indicate that in such cases the aircraft may have its spinning tendency increased from a stalled position.

g. Highly tapered wings have a tendency to stall first near their tips, the stall at one tip usually preceding that at the other, which usually produces roll just prior to or during the stall. This is, of course, conducive to the development of a spin and for this reason almost any airplane having such wings should be investigated for any undue tendency to spin. The taper ratio is approximately the ratio of the wing chord at the root to the chord of the tip and any value of this ratio in excess of three may be considered "high."

h. For the current conventionally designed airplanes, the following loading and flight conditions are likely to be critical: (1) Forward *CG* with respect to excessive recovery speeds and accelerations. (2) Rearward *CG* (maximum load) with respect to spin recovery.

2. Tests

All spin tests should be conducted with the greatest possible caution. No spin test should be conducted until the applicant has satisfied himself that the airplane complies with the requirements governing all the other flight characteristics.

For official flight tests, the inspector is authorized to require the applicant to install such safety devices as releasable load aft of the center of gravity, spin chutes, means of rapid exit from the airplane, etc., prior to conducting any spin tests.

The spin testing should begin with a process of familiarization during which the operator should proceed cautiously to acquaint himself with the spinning characteristics of the airplane by entering the spin repeatedly and allowing its duration to increase each successive time by as little as one-half turn if he considers this to be necessary. During this period each individual control should be investigated for the existence of back pressure and if it be found, it should be interpreted as a warning to proceed with even greater caution.

The following tests shall be conducted to demonstrate compliance with the requirements. All such tests shall be conducted at the maximum weight obtainable at the center of gravity positions hereinafter specified unless experience during these tests indicates the probability that some other combination of weight and center of gravity position may be critical, in which event the applicant shall investigate such of these as are necessary to determine which is in fact critical. All tests with flaps retracted shall be conducted with the airplane trimmed for level flight at 90 percent of the maximum level flight speed obtained with maximum-except-take-off power at the critical altitude of the engine. With flaps extended, the tests shall be conducted with the airplane trimmed for steady gliding flight with throttles closed at a speed equal to 130 percent of the measured stalling speed with the airplane in the same condition.

Normal spins

Entry.—Entry into normal spins may be made with any necessary use of the controls and power but, once started, the spin shall thereafter be maintained with the rudder and elevator in the full pro-spin position and with the ailerons neutral, power-off.

Recovery With Free Controls.—As acceptable evidence that there is no reversal of control force, the airplane shall, with rearmost center of gravity, be so spun for six turns and, upon releasing the elevator and rudder from the full spinning position, the spinning motion shall stop in not more than four turns thereafter.

Recovery by Normal Use of Controls.—As acceptable evidence of the ability to recover by normal use of the controls:

a. Flaps and landing gear retracted.—The airplane shall, without rearmost center of gravity, be so spun for 1, 2, 4, and 6 turns and, with foremost center of gravity for six turns and, upon moving the elevator and rudder controls from the full spinning position toward but not beyond the neutral or "trimmed" position, the spinning motion shall stop and normal control be regained in not more than one and one-half turns after starting this movement of controls and the complete recovery to level flight at the trimmed speed shall be made without having at any time exceeded either the maximum permissible airspeed or the limit normal acceleration.

b. Flaps and landing gear extended.—The airplane shall also, with rearmost center of gravity and with flaps and landing gear fully extended, be so spun for one turn and, upon moving the elevator and rudder controls from the full spinning position toward but not beyond the neutral or "trimmed" position, the spinning motion shall stop and normal control be regained in not more than one and one-half turns after starting this movement of the controls.

Uncontrollable Spin.—As acceptable evidence that an uncontrollable spin cannot be obtained by any use of the controls the airplane shall, with rearmost center of gravity, be so spun for six turns at the end of which the elevator shall be completely reversed (if necessary) following which the airplane shall recover or recovery shall be possible with the elevator maintained in this position by reversal of the rudder. Recovery in two to three turns is normally considered satisfactory.

Abnormal spins

Entry.—Entry into abnormal spins may be made in any manner but, once started, the spin shall thereafter be maintained with the elevator and rudder in the full pro-spin position and with the ailerons either full with or full against the spin as hereinafter specified.

Recovery.—The airplane shall, with rearmost center of gravity, be so spun for six turns with ailerons full with the spin and also with ailerons full against the spin and, upon neutralizing the aileron control and making full use of the elevator and rudder controls for the purpose of recovery, the spinning motion shall stop and normal control be regained in not more than two turns thereafter.

NOTE: If in any of the normal or abnormal tests above with landing gear retracted the applicant has any reason to believe that the spin recovery would be more critical with the landing gear extended, the entire process should be repeated with the landing gear in the extended position. After the most critical test has been determined for the airplane with the landing gear retracted, the test should be repeated with the landing gear extended as a check to insure that the latter condition is not more critical.

Non-spinning airpla

Most airplanes claimed to be incapable of being spun depend upon some limitation of the control surface movement or the center of gravity location to insure this inability to obtain a spin. For this reason, in such cases, the spinning characteristics of the airplane should be investigated with the elevator and rudder control systems so arranged for a movement in excess of that subsequently to be available on the airplane and at a center of gravity location aft of the rearmost limit for which certification is requested.

On the basis of the cases which have been encountered thus far, it is recommended that this investigation be made with the elevator and rudder movement not less than 5° in excess of the intended limit and with the center of gravity not less than 2 percent of the MAC aft of the rearmost limit. Also, the exact margin which should be required should depend to some extent upon the reliability of the means provided to limit the control and center of gravity movement. An attempt should then be made to obtain a spin by any means whatever and this must be found to be impossible and further to be most unlikely under any reasonably probable condition likely to exist during the life of the airplane.

04.707 Flutter and vibration. Wings, tail surfaces, control surfaces and primary structural parts shall be free from flutter or objectionable vibration in all normal attitudes or conditions of flight between the minimum flying speed and the maximum indicated airspeed attained in official flight tests. (See § 04.722.)

1. General

a. Buffeting is the condition in which repeated aerodynamic forces are experienced by any part of an airplane which have been caused and maintained by unsteady flow arising from a disturbance set up by any other part of the airplane. *Flutter* is an oscillation of definite period but unstable character set up in any part of an airplane by a momentary disturbance and maintained by a combination of aerodynamic, inertia and elastic characteristics of the member itself. Once set up, it is likely to destroy the parts involved unless quickly stopped. Flutter usually is possible only above a certain airplane speed and, for this reason, a reduction in speed is strongly indicated in the event it is encountered. *Vibration* is somewhat the same as flutter except that the oscillations are damped so that the amplitude remains the same or decreases.

b. 04.722 (Maximum Airspeed) requires that the airplane be flown steadily at the design V_g or at $1.33V_L$, whichever is less, and also with flaps extended at the design V_f, or $2V_{sf}$, whichever is less. The purpose of this requirement is to provide a demonstration that flight is practicable and that the airplane may be satisfactorily operated under what are believed to be critical conditions at two critical speeds within the range of all speeds permitted by the operation limitations. A further purpose which this requirement serves is, in those cases in which it is impossible or unsafe to attain V_g, to establish the placard V_g at 90 percent of the highest speed which is actually obtained. It should be noted, however, that in no such case in which the design V_g has not been attained and in which the design V_g is in excess of $1.333V_L$ may a placard V_g be based upon an actual speed less than $1.33V_L$.

c. There are a number of factors involved in this demonstration of practicability and satisfactory operation, the more important of which are:

(1) The test should show that the secondary structure, i. e., such items as engine cowling, windshields, inspection and lubricating openings, etc., most of which are not amenable to analytical treatment in the process of design, will probably remain intact during the normal service life of the airplane.

(2) The test should show that the airplane is controllable and that no dangerous characteristics may be expected at the placard speeds selected during steady flight.

(3) The test provides a means by which to determine that a fixed or adjustable pitch propeller has been so selected or that the stops have been so adjusted for a controllable pitch propeller that it will not idle at an engine speed in excess of 110 percent METO rpm. (See "Propellers", page 123).

(4) The test will provide a means by which to determine whether or not the occurrence of flutter or objectionable vibration of any part of the airplane is probable within the placard airspeed limitations.

d. 04.743 (Airpseed Limitations) requires that the airspeed operating limitations (placard speeds) for gliding flight and for flight with flaps extended be not more than 90 percent of the corresponding speeds actually obtained in accordance with 04.722. The purpose of this requirement is to provide a margin of safety of 10 percent of the speeds actually obtained to allow for inaccuracies in instruments or imprecision on the part of the future users of the airplane thus insuring, with reasonable certainty, that if the user attempts to observe the operating limitations the speeds will not in fact exceed those which have actually been investigated in accordance with 04.722.

e. Flutter can be caused by the combined effects of several undesirable features, any one of which might not be dangerous in itself. Freedom from flutter as observed in flight tests is not necessarily complete assurance, for there may exist a particular combination of speed, acceleration, weight or other features at which a dangerous flutter may occur. Detailed principles of flutter prevention are outlined in CAA Engineering Reports, Nos. 22, 23, and 24, entitled "Flexure-Torsion Binary Flutter," "Perpendicular Axes Control Surface Binary Flutter," and "Parallel Axes Control-Surface Binary Flutter," respectively. (Also see discussions under 04.404 and 04.424.)

f. In any control surface or similar surface attached to the trailing edge of an airfoil, the use of a spring device as an essential part of the control system greatly increases the probability of flutter. Such installations are not considered satisfactory unless used in conjunction with other counteracting features, such as oleo damping devices or over-balancing of control surfaces. It should be noted that when spring devices are used in the control system, the vibration and flutter characteristics must be investigated both with and without the use of the spring device in accordance with 04.436.

g. During performance and flight characteristic tests, care should be exercised to have the control system as free of friction as possible. Friction that is present in a new aircraft may deter detection of flutter tendencies until later in service use when control movements are more free.

h. Tail buffeting is usually associated with relatively high angles of attack and is often caused by wing-fuselage interference. It will frequently occur during attempts to spin an airplane that tends to spiral or in flight conditions such that a heavy load factor is imposed on the wing with resultant breakdown of airflow over the wing. A practical discussion of preventive measures is contained in NACA Technical Note No. 460, "Full-Scale Wind-Tunnel Research on Tail Buffeting and Wing-Fuselage Interference of a Low-Wing Airplane". Any buffeting which might result in injury to the structure or difficulty in control is unsatisfactory.

i. The airspeed indicator must be marked for the placard values of V_g, V_L, and V_f. It should be noted that V_L is obtained directly from design data, but that V_g and V_f are limited to not more than 90 percent of the highest speed actually attained during the type inspection tests or 90 percent of the design values, whichever are lower.

2. Acceptable Methods

The procedure given below should be followed:

a. The test should be conducted at the maximum weight for which certification is sought.

b. The test should be conducted at the rearmost *CG* location attainable with maximum weight, unless there is reason to believe that some other *CG* location may be critical in respect of one or more of the characteristics involved, in which case the test will be conducted at the rearmost *CG* location and at any other deemed necessary by the Inspector during official tests.

c. The airplane should be trimmed for level flight at METO power, the speed very gradually increased until the desired value of true indicated airspeed from calibration curve is reached, which value should be maintained for such time as is necessary to make all of the observations required, and thereafter level flight should be regained by a very gentle pull-out. It is recommended that the above trim setting remain unchanged unless the control forces required to maintain the airplane at the desired speeds become impossibly large. In this event, it is recommended that the trim setting be changed as little as is possible in order to relieve these control forces.

d. It is suggested that in the case of V_g the process described in item *c* be repeated at least three and preferably a greater number of times, each succeeding time attaining successively higher speeds until V_g be reached; that all of the characteristics under investigation be very carefully observed and all required observations be made during each such trial; and that the test be discontinued if at any stage in the entire process it appears that difficulty will be encountered with any further increase in speed.

e. Dives to design gliding speed should be made in increments increasing the speed 10 mph each time.. These dives should all be started with ample altitude to permit adequate time for egress should any failure occur. In this connection, it should be remembered that if flutter develops, disintegration of the affected members may occur in a very short time, and that the resulting aircraft maneuvers make it imperative that all such dives be made with at least 5,000 feet altitude at completion of levelling out slowly. It is desirable, if the engine will not over-rev, to use some throttle during the dive, since this will not only aid in preventing torching (fire from unburned gas extending outside exhausts), but will also make it possible at the first sign of flutter to close the throttle and reduce the airspeed (AVOID SHARP PULL-OUTS). It is usually necessary to drop well below the airspeed at which the flutter or vibration started before it will die out. Although no definite rule can be given, it is known that if a flutter tendency is present during any dive to an increasing speed increment, it may be incited by small sharp movements of the control surface, care being taken not to alter the flight path or attitude of the airplane during these movements.

The above procedure, if started at relatively low speeds, is believed best for the purpose of discovering potentially destructive flutter before it can reach dangerous proportions since the violence and magnitude of flutter usually increases with speed.

04.707–T Flutter and Vibration. All parts of transport category airplanes shall be free from flutter or excessive vibration under all speed and power conditions appropriate to the operation of the airplane during take-off, climb, level flight, and landing, and during glide at speeds up to the maximum indicated airspeed attained during official flight tests. (See § 04.722). There shall be no appreciable buffeting for any flap position at any speed in excess of 10 miles per hour above stalling speed for such position nor shall buffeting at lower speeds be so violent as to interfere with the pilot's control of the airplane or cause discomfort to its occupants.

The requirements of this section are considered to be clearly stated by its text. It is not anticipated that any special flight testing will be necessary to show compliance beyond the usual dive test required of all airplanes unless difficulty be encountered during this or any other necessary testing.

04.708 Ground and Water characteristics. Land planes shall be maneuverable on the ground and shall be free from dangerous ground looping tendencies and objectionable taxiing characteristics. The seaworthiness and handling characteristics of seaplanes and amphibians shall be demonstrated by tests deemed appropriate by the Administrator. (See § 04.452 for water stability requirements.)

1. Ground Characteristics

Taxiing tests should be conducted on smooth and rough ground as may likely be encountered under normal operating conditions. Speeds should be used which should vary up to approximately 70 percent of stalling speed. Particular attention should be paid to the following:

Taxiing over rough ground.—There is some evidence to indicate that critical loads can be built up in taxiing over rough ground, even when the shock-absorbing system is entirely satisfactory with respect to capacity for landing purposes.

Brakes.—Their adequacy in maneuvering on the ground and their tendency to cause nosing-over should be investigated. Any bad tendency will normally be exaggerated when taxiing in a strong side or tail wind.

2. Water Characteristics

In order to check water stability, taxiing tests should be made cross wind in a fairly strong breeze.

Sailing ability should be investigated by actually sailing the aircraft with appropriate use of the engine.

Porpoising tendencies should be investigated and reported upon for extreme loading combinations.

Ability to maneuver up to and while on the step should be investigated and reported on.

Effectiveness of the water rudders when provided should be checked.

3. Ski Handling—General

In the case of skis, the interpretation is that satisfactory landings and take-offs are required, as well as satisfactory ski trim in normal flight.

During landings, take-offs, taxiing, etc., particular attention should be paid to the following:

a. No undue directional instability.

b. Satisfactory ability to make normal speed taxiing turns without undue skidding tendency.

c. No undue tendency to have the nose of each ski digging in during landings or take-offs.

d. Trimming gear adequate with no danger of damage occurring during normal ground handling in taxiing or maneuvering over rough snow or drifts.

e. Ski trim in flight stable and satisfactory for all normal flight attitudes including slips and skids.

f. If braking devices are employed, attention should be given to adequacy, effectiveness, and protection during normal operation on the ground. Positive action should be required.

04.71 MODIFIED PERFORMANCE REQUIREMENTS FOR AIR CARRIER AIRPLANES NOT CERTIFICATED IN THE TRANSPORT CATEGORY.

The weight of any multi-engine air carrier aircraft manufactured pursuant to a type certificate issued prior to January 1, 1941, and which aircraft is being operated in accordance with the requirements of Part 61, may be increased beyond the values corresponding to the landing speed specified in § 04.700 and take-off requirements of § 04.701, subject to the following conditions:

04.710 The increased weight shall be known as the *provisional* weight (§ 04.103). The *standard* weight (§ 04.102) shall be the maximum permissible weight for all operations other than those in accordance with the requirements of Part 61. The *provisional* weight shall be the maximum permissible weight for any operation.

04.711 Compliance with all the airworthiness requirements except landing speed and take-off is required at the *provisional* weight, except that the *provisional* weight may exceed the *design* weight on which the structural loads for the landing conditions are based by an amount not greater than 15 percent, provided that the airplane is shown to be capable of safely withstanding the ground or water shock loads incident to taking-off at the *provisional* weight.

04.712 The aircraft shall be provided with suitable means for the rapid and safe discharge of a quantity of fuel sufficient to reduce its weight from the *provisional* weight to the *standard* weight.

04.713 In no case shall the *provisional* weight exceed a value corresponding to a landing speed of 5 miles per hour in excess of that specified in § 04.700, a take-off distance of 1,500 feet in the case of landplanes, or a take-off time of 60 seconds in the case of seaplanes.

04.714 Aircraft engaged in operations in accordance with the requirements of Part 61 shall be certificated for the weight at which they comply with the take-off and other performance provisions of those regulations for the particular operation involved provided that such certified weight shall not exceed the provisional weight. It may, however, be less than the provisional or *standard* weights, dependent upon the ground or water facilities and the nature of the route flown.

04.72 PERFORMANCE TESTS.

04.720 General. Compliance with the foregoing performance requirements shall be demonstrated by means of suitable flight tests of the type airplane. Computations may be used to estimate the effects of minor changes. Additional information concerning the performance characteristics of air carrier airplanes is specified in § 04.73. Such characteristics shall be determined by direct flight testing, or by methods combining basic flight tests and calculations. All performance characteristics shall be corrected to standard atmospheric conditions and zero wind. Methods of performance calculation and correction employed shall be subject to the approval of the Administrator.

As indicated in 04.720, the airplane presented for type approval will be subjected to such official flight tests as are necessary to prove compliance with the requirements, and to supply the pertinent information required upon a form to be provided by the CAA.

The applicant shall, at his own expense and risk, conduct such official flight tests as required by the inspector to demonstrate compliance with the minimum requirements.

Ballast shall be carried to simulate pay load when necessary during flight tests. Consideration must be given to the vertical as well as horizontal location of ballast where such might have an appreciable effect on performance or flight and handling characteristics. Careful attention must also be given to supporting structures to determine that failure will not result from loads imposed during tests.

04.7200 The applicant shall provide a person holding an appropriate commercial pilot certificate to make the flight tests, but a designated inspector of the Administrator may pilot the airplane during such parts of the tests as he may deem advisable.

The applicant must provide the personnel necessary, in the opinion of the CAA, for the conduct of the official flight tests, including a competent commercial pilot with an appropriate rating as required by CAR 20. No persons other than the crew necessary to conduct the tests shall be carried during the flight tests.

04.7201 In the event that the applicant's test pilot is unable or unwilling to conduct any of the required flight tests, the tests shall be discontinued until the applicant furnishes a competent pilot.

04.7202 Parachutes shall be worn by members of the crew during the flight tests.

04.7203 The applicant shall submit to the inspector of the Administrator a report covering all computations and tests required in connection with calibration of flight instruments and correction of test results to standard atmospheric conditions. The inspector will conduct any flight tests which appear to him to be necessary in order to check the calibration and correction report or to determine the airworthiness of the airplane.

The calibration report of powerplant instruments, loading schedule, and proposed flight test program shall be submitted by the applicant in addition to those specified in 04.7203.

04.721 Loading conditions. The loading conditions used in performance tests shall be such as to cover the range of loads and center of gravity positions for which the airplane is to be certificated.

04.7210 Use of ballast. Ballast may be used to enable airplanes to comply with the flight requirements as to longitudinal stability, balance and landing, in accordance with the following provisions:

04.72100 (*a*) Ballast shall not be used for this purpose in airplanes having a gross weight of less than 5,000 pounds nor in airplanes with a total seating capacity of less than 7 persons.

04.72101 (*b*) The place or places for carrying ballast shall be properly designed and installed and plainly marked.

04.72102 (*c*) The loading schedule which will accompany each certificate issued for an airplane requiring special loading of this type shall be conspicuously posted in either the pilot's compartment or in or adjacent to the ballast compartments and strict compliance therewith will be required of the airplane operator.

Ballast should not be used, for the purpose of enabling compliance with airworthiness requirements, in airplanes having a gross weight of less than 5,000 pounds nor in airplanes with a total seating capacity of less than 7 persons. This provision is not intended to include fixed or permanent ballast where the attachment and supporting structure have been approved by the CAA Aircraft Engineering Division and a suitable placard provided, nor is it intended to prohibit the practice of placarding baggage compartments or fuel tanks for reduced capacities. Where removable ballast is permitted in aircraft with a gross load of 5,000 pounds or over, the place or places for carrying such should be properly designed and installed and plainly marked.

04.7211 Fuel to be carried. When low fuel adversely affects balance or stability, the airplane shall be so tested as to simulate the condition existing when the amount of fuel on board does not exceed one gallon for every 12 maximum (except take-off) horsepower of the engine or engines installed thereon. When the engine is limited to a lower power, the latter shall be used in computing low fuel.

This is interpreted to mean that tests with low fuel should be made only for the purpose of demonstrating compliance with requirements pertaining to flight and handling characteristics or performance and is not intended for use in tests to demonstrate the efficiency of the fuel system or for certification.

04.722 Maximum airspeed. The flight tests shall include steady flight in relatively smooth air at the design gliding speed (V_g) for which compliance with the structural loading requirements (§ 04.21) has been proved, except that they need not involve speeds in excess of 1.33 V_L (§ 04.111), provided that the operation limits are correspondingly fixed (see § 04.743). When high-lift devices having nonautomatic operation are employed, the tests shall also include steady flight at the design flap speed V_f (§ 04.114), except that they need not involve speeds in excess of 2 V_{sf} (see § 04.113). In cases where the high-lift devices are automatically operated, the tests shall cover the range of speeds within which the devices are operative.

04.723 One engine inoperative performance. Multi-engine airplanes shall be flight tested at such altitudes and weights as are necessary, in the opinion of the Administrator, to prepare accurate data to show climbing performance within the range of weight for which certification is sought, with the critical engine inoperative and each other engine operating at not more than maximum except take-off power. Such data when approved by the Administrator shall be kept in the airplane at all times during flight in a place conveniently accessible to the pilot.

1. Critical Inoperative Engine

The "critical" inoperative engine is that engine which results in the lowest best rate of climb for any multi-engine airplane when such engine is inoperative. This engine will not necessarily be critical for controllability and trim. (See Section V, D, form ACA 283, "Type Inspection Report".) Experience has indicated that in cases of two- and four-engine airplanes the left outboard engine inoperative is usually critical for both conditions.

Acceptable method.—To eliminate the necessity of conducting sawtooth climbs at various speeds with each engine inoperative, it is considered adequate to determine the "critical" engine, before establishing the one engine inoperative performance, by means of check climbs conducted as follows:

a. The airplane weight may be any weight except that the loading should not be such as to be more unfavorable to one engine than to the other(s). In cases where some uncertainty exists regarding this matter the standard gross weight should be used.

b. Check climbs should be conducted on the same day, and, if possible, on the same flight; in any event, under as nearly identical conditions as is practicable regarding the weight of the airplane during each climb, altitude, airspeed, and the condition on the operative and inoperative engines in order that the minimum amount of correction to the observed data will be necessary to establish the identity of the critical engine.

c. It is considered desirable in all cases to conduct the check climbs at airplane weights so that the weight correction will not be necessary. It should be noted that an airplane weight change of 1 percent may affect the rate of climb by an increment of approximately 15 feet per minute. Therefore, in cases where the difference in rates of climb with the various engines inoperative is of small magnitude, it becomes apparent that the weights should be carefully controlled, and particular care should be taken to obtain accurate results in the determination of the critical inoperative engine on airplanes for which the usable ceiling is below the critical altitude of the engine(s), because of the pronounced effect of small increments of climb on the usable ceiling in these cases.

d. When the usable ceiling is above the critical altitude of the engine(s), reasonable care should be exercised in selecting the critical inoperative engine, but, since slight changes in the rate of climb will usually affect the usable ceiling only by an amount acceptable without further correction for weight, it will usually be considered satisfactory in such cases to select the critical inoperative engine on the basis of the check climb data without correcting for weight if the weights during such tests are maintained within 1 percent.

e. Usually the above may be accomplished as follows:

(1) All climbs to start at an altitude sufficiently below or above the pressure altitude at which timing is to start to permit attaining a steady flight condition. Timings with each engine inoperative are to start at the same pressure altitude.

(2) The duration of the timed portion of each climb should be at least five minutes.

(3) All climbs are to be made at the same indicated airspeed, which should preferably be approximately the speed for the best rate of climb.

(4) The operating engine(s) is (are) to be held at as nearly identical rpm's and manifold pressure readings as possible. The carburetor heater, cowl flaps, and mixture setting are also to be in the same position in all climbs.

(5) The inoperative engine(s) is (are) to be in the same condition in all cases regarding propeller pitch setting, cowl flap position, throttle setting, and ignition switch position.

(6) If the manufacturer has already conducted tests to indicate which engine is critical, that engine should be cut out last in the series of the check climbs in order that this test be made at the reduced aircraft weight, thereby resulting in more conclusive results in cases where the weight reduction may obscure the true result. It is to be noted that it will generally be desirable to conduct these tests in succession with no intervening tests and with as little delay as practicable in order that the effect of changing weight may be kept to a minimum.

Corrections.—In case it is considered necessary to use the weight or other correction in obtaining the true rate of climb in these check climbs, it will be acceptable to make these corrections as outlined in Flight Engineering Report No. 3.

2. One Engine Inoperative Performance Tests

The primary purpose of the requirements of 04.723 is to obtain the information necessary to inform the pilot of the one engine inoperative performance of the airplane under any condition likely to be encountered following an engine failure during the life of the airplane.

Unless some special reason is involved, "such weights as are necessary" is to be the maximum weight for which certification is sought. In order to accomplish this purpose it is necessary that sawtooth climbs be conducted. The nature and number of these sawtooth climbs should be the same as those described under 04.702 for the third purpose involved, except that the critical engine as determined immediately above is to be inoperative and the airplane is to be otherwise in the condition most favorable to climb, i. e., flaps retracted, inoperative propeller feathered or windmilling in high pitch. Cowl flaps, if present, are to be in the position that will meet the cooling requirement for standard air temperature at the altitude at which the tests are being conducted.

In the past, the information which has been placed in the hands of the pilot concerning the one engine inoperative performance has been limited to a usable ceiling, i. e., the altitude in standard air at which the best rate of climb is 50 feet per minute. In order to accomplish the above purpose, however, it is obviously necessary to go beyond this and it is requested that the applicant furnish a chart showing the one engine inoperative best rate of climb and the corresponding true indicated airspeed against altitude at various weights covering the range of weight at which the airplane is likely to be operated. Provision should be made for keeping the chart, once it is approved, in the airplane at all times in a place conveniently accessible to the pilot.

Acceptable Method.—Same as described under 04.702 except additional data will need to be recorded.

Corrections.—The corrections necessary to determine the actual rate of climb are described in Flight Engineering Report No. 3.

04.724 Airspeed indicator calibration. In accordance with § 04.5800, the airspeed indicator of the type airplane shall be calibrated in flight. The method of calibration used shall be subject to the approval of the Administrator.

04.725 Check of fuel system. The operation of the fuel system shall be checked in flight to determine its effectiveness under low fuel conditions and after changing from one supply tank to another. (See § 04.620.) For such tests low fuel is defined as approximately 15 minutes' supply in each tank tested at the maximum (except take-off) power certified.

1. General

a. The fuel system should be checked in flight to determine that it will feed satisfactorily in climbs at the best angle of climb speed and at the best gliding angle. Moderate rolls, slips and skids as might be made accidentally in the above climbs and glides with *low fuel* should not cause the engine(s) to cut out. The system should also feed promptly after one tank has run dry and the other tank is turned on.

b. Low fuel for the purpose of fuel system tests in flight is defined as approximately (METO HP ÷ 40) gallons in the tank tested. Each tank will be tested unless the arrangement of the system indicates identical results would be obtained. Special rulings will apply in cases of fuel systems with several small tanks which would result in an unreasonably large amount of residual fuel. If the fuel system has only one tank, the engineering inspector may at his discretion during official tests request the installation of a temporary auxiliary fuel tank for low fuel flight tests in order to avoid the possibility of a forced landing due to lack of fuel.

c. Fuel systems of single engine aircraft shall not be considered satisfactory if more than 10 seconds elapse after changing over from a tank which has run dry in flight, before the engine resumes full power operation.

d. Systems with tank outlets and vents interconnected and so arranged that it is impossible to feed from each tank individually should be treated as single tank systems.

2. Acceptable Method

Single or individual tank tests.—(1) Flight tests should be conducted with fuel arranged in the tanks so that each tank can be tested separately with low fuel but always with a safe amount of fuel in another tank. (2) If the aircraft has only one tank, the take-off will be made with a safe amount of fuel and the tests conducted with *low fuel* as defined above. (3) The tests should be conducted by changing over to the tank with *low fuel* after a safe altitude is reached. Each position or attitude should be maintained for a period sufficient to interrupt flow at the carburetor should the feed ports be uncovered. (4) Tests should include steady climbs with maximum permissible power in the attitude of best angle of climb and in glides at approximately the best gliding angle. Flaps, if installed, should be extended in glide. Banks involving moderate slipping and skidding should also be executed in the climbs and glides. The system should also be investigated for the effect of acceleration as would occur during take-offs and flight maneuvers. If tests are considered necessary, the take-off condition can be checked on the ground by accelerating rapidly to take-off speed with *low fuel* in the tank. The take-off should not continue beyond this point.

Multi-tank system.—In the case of a multi-tank system, the tests should be continued until the engine cuts out or the tank runs dry. (Tests of single tank systems should not be continued beyond the point where the *low fuel* supply is reached.) The time required to regain normal power, with and without the use of the auxiliary pump, *after* the fuel valve has been changed to another tank containing ample fuel must be recorded. The time should not exceed 10 seconds for single engine airplanes.

Residual fuel.—The amount of fuel (exclusive of sump capacity) in the tank after the engine has cut out should be recorded. Immediately upon landing, after conducting this test, the gas remaining in the tank that has been tested should be drained and the amount recorded in pounds or gallons or fraction thereof. Adjustments should be made on the individual gasoline gauge of this tank so that the amount drained will not register as reserve gas. The amount of gas remaining in the tested tank should be included in the empty weight of the aircraft as far as certification purposes are concerned.

Take-off and landing tank only.—In the case of aircraft using more than one fuel tank, where that aircraft is placarded for take-offs and landings on one (or more) tanks only, the auxiliary tanks may be considered acceptable when tested for the level flight, climb, and glide conditions only.

04.73 PERFORMANCE CHARACTERISTICS OF AIR CARRIER AIRCRAFT.

No air carrier shall operate aircraft in scheduled air transportation unless data shall have been submitted to and approved by the Administrator, covering the determination of such performance characteristics, in addition to those specified in §§ 04.70, 04.71, and 04.72, as are, in the opinion of the Administrator, necessary to determine the ability of such aircraft to safely perform the type of operation which the air carrier proposes to conduct. The method used for the determination of such ability shall be subject to the approval of the Administrator.

04.74 OPERATION LIMITATIONS.

04.740 Weight. Non-air-carrier airplanes may be certificated at a maximum authorized weight which is not sufficient to permit carrying simultaneously the full fuel and full pay load, provided that such weight shall be sufficient to provide a gasoline load of at least 0.15 gallon per certified maximum (except take-off) horsepower, with all seats occupied and with sufficient oil for this amount of fuel.

04.741 Provisional weight (Air carrier airplanes). (See § 04.71.)

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

1. General (Also see "Performance", 04.70.)

a. The *CG* limits will usually be established during the course of official type inspection tests and are based upon compliance with one or more of the requirements for flight characteristics. The aft limit is usually the most rearward with which compliance with the longitudinal stability, stalling, or spinning requirements is demonstrated. The forward limit is usually the most forward with which satisfactory take-off or landing or compliance with the balance requirements in a glide or climb are demonstrated. (Also see discussions under 04.704, 04.705, 04.706.)

b. The applicant should select the desired limits with which he proposes to demonstrate compliance. In case such compliance is shown without difficulty, no information is gained as to the absolute limits at which compliance is barely possible. Since in the future, the applicant may wish to alter the approved limits, and unless the absolute limits have been determined at the time of the original official type inspection, approval of altered limits will require further testing. For these reasons, the applicant should determine these absolute limits on all new designs.

2. Acceptable Method

a. To provide "means . . . by which the operator is suitably informed . . . " is interpreted as a placard or system of placards either self-contained or referring to a loading schedule which defines all possible permissible loading conditions. Such is required when it is possible to so load the airplane with useful load and/or equipment for which provision has been made in the design as to exceed the *CG* limits for which the airplane is certificated.

b. Very little can be written in the way of specific instructions because the conditions actually encountered may vary enormously. In general, every effort should be made to eliminate the necessity for the placard(s) but, failing this, the placards should be made as simple as possible without impairing their effectiveness. In some cases the placards required by 04.465 can be made to accomplish the purpose. In cases where a loading schedule is required which is too involved to become a placard, it is recommended that those points on or in the airplane such as fuel tanks, seats, baggage or cargo compartments, etc., which are involved in the schedule, bear placards referring to the schedule.

c. Although no standard form for loading schedule is specified, the following must be included: (1) The date of issuance. (2) The serial number of the airplane. (3) The "NC" number, if known. (4) The empty weight and the equipment included therein. (5) The location of all items of removable equipment. (6) The Note: "When baggage or cargo is not available, ballast equal to the minimum specified must be carried." (7) The Note: "Before adding, removing, or relocating any item of equipment, contact a CAA inspector." (8) The signature of the engineer responsible for the preparation of the schedule. (9) Space for the signature of Chief, Aircraft Engineering Branch. (10) Consideration of maximum possible and minimum required fuel and the appropriate oil capacities.

d. A resumé of the preliminary work involved in the preparation of the schedule must accompany the weight and balance report at the time of official tests.

04.743 Airspeed limitations. Maximum operation limitations will be incorporated in the aircraft certificate and will specify the indicated airspeeds which shall not be exceeded in level and climbing flight (§ 04.111), in gliding and diving flight, and with flaps extended. The values in gliding flight and with flaps extended will be 10 percent less than the corresponding maximum airspeeds attained in flight tests in accordance with § 04.722.

04.744 Powerplant limitations. Maximum operational limitations will be incorporated in the aircraft certificate and will specify powerplant outputs on take-off (§ 04.260), in climbing flight, and for all operations other than take-off and climbing flight (§ 04.105). The output, except for take-off, shall not exceed that corresponding to the maximum (except take-off) rating of the engine installed. For the above purposes no specified output will be in excess of that corresponding to the limits imposed by either the pertinent engine or propeller certification. (See §§ 04.60 and 04.61.)

04.75–T PERFORMANCE REQUIREMENTS FOR TRANSPORT CATEGORY AIRPLANES.

The following requirements shall apply in place of §§ 04.700 to 04.706, inclusive:

As a result of discussion during the development of the transport category requirements, it has been conceded that any airplane suitable for "transport" purposes should be a multi-engine airplane, and that airplanes with the high landing speeds that would be permitted by the transport regulations should be able to lose the power from one engine and still reach a proposed landing area. The regulations are therefore such that single engine airplanes are ineligible for certification in the category. This aspect of the regulations is reflected by various rates of climb required with one engine inoperative. CAR 40.2 requires that any new type airplane introduced into scheduled air carrier passenger operation must have been certificated in the transport category and CAR 61.712 imposes certain operating limitations upon such airplanes. These operating limitations are applicable to any part of any route flown by the airplane and take the form of specifying, in terms of the weight at which the airplane may be operated, a relation which must exist between certain items of performance measured as a part of the process of type certification and the dimensions of the route over which the airplane is operated.

The use of aircraft certificated under this category is not, however, confined to air carriers. For this reason the regulations include a set of minimum requirements for certification in the category. In the general case in which the applicant for certification anticipates the use of the airplane in scheduled passenger operation and subject to the operating requirements contained in CAR 61.712, it will be necessary to consider all or the extreme values of weight, altitude, and such other variables as are involved in the application of these requirements. The performance information obtained in demonstrating compliance with the minimum requirements for certification in the category would limit the use of the airplane in scheduled passenger operation to sea level airports and to routes covering no terrain of greater altitude than 4,000 feet, and would require the imposition of the operating limitations of CAR 61.712 at the maximum weights selected for that purpose. If the airplane is not used for scheduled passenger operation the minimum performance information required is necessary in order that the pilot may intelligently operate the airplane. It is for this reason, as well as for the reason that the information is necessary in order to comply with the operating requirements, that an operating manual must be furnished for each airplane which must contain certain of the performance information required for, and determined during, certification.

The fundamental distinction between the requirements of the transport category and those applicable to all airplanes, may perhaps be most aptly identified in terms of certain options, which must be exercised by the applicant for certification in the transport category prior to or at the time of his application. These options follow.

1. *The selection of the range of weight and altitude to be covered by the flight testing required for certification.*—This selection must be based upon the extent to which the applicant for certification is concerned with the operating limitations which will be imposed upon the airplane as a result. If the applicant is not concerned with this point, he may elect to conduct only the flight test required to demonstrate compliance with the minimum performance requirements contained in 04.750–T together with those required to demonstrate compliance with the flight characteristics and other requirements. If the airplane is not to be used for scheduled air carrier passenger operation, or if it is practicable to limit the operation of the airplane by a scheduled air carrier to sea level airports containing runways of ample length and to terrain altitude not in excess of 4,000 feet, this procedure appears satisfactory. It should be noted that this latter case could quite easily apply to a flying boat in scheduled operation. If, the applicant wishes to provide for the greatest possible flexibility in the matter of compliance with the operation limitations contained in CAR 61.712, considerably more performance measurement will be necessary. It may be entirely practicable, for example, for operation over routes involving appreciable differences in the altitude of airports along it, to take advantage of the improvement in performance which is possible by means of reducing the weight at which the airplane is operated. It may also be desirable to alter the various flap settings in order to improve the climbing performance at a given weight while not exceeding the limitations upon stalling speed at that weight. In cases such as these, it will be necessary to determine by flight testing and calculation, the effect of weight, altitude, and flap setting, throughout the range of each for which it is desired to provide, upon the take-off, landing, and climbing performance and to include this information in the operation manual. This selection must obviously be left to the applicant since, even though he may find it difficult to anticipate the uses to which the airplane may subsequently be put, he is nevertheless in better position to forecast this than anyone else.

2. *The selection of the weight range to be covered by the terms of the certification.*—This selection is closely related to the option above and must be based upon essentially the same considerations. The simplest possible selection of weights is a single maximum weight to be used both for take-off and landing and as a basis for the imposition of the operating limitations. The next simplest choice would appear to be a maximum take-off weight and a maximum landing weight differing from take-off weight. This requires the installation of fuel dumping equipment of sufficient capacity to reduce the weight of the airplane from the maximum take-off weight to the maximum landing weight in not more than ten minutes. The operating limitations may then be based upon the assumption that these two weights exist throughout each flight. The most flexible possible arrangement in the matter of weights is provided by selecting a range of weights for take-off and a range for landing, and determining the performance as functions of these weights in order that, for the purpose of showing compliance with the operating limitations, any weight within these ranges may be selected to fit the requirements of a particular route. It is obvious that this selection should be left with the applicant.

3. *The selection of the range of altitude to be covered by the terms of the certification.*—This is also closely related to option 1 and is analogous in its nature to option 2. The simplest possible selection is that indicated by the minimum performance requirements contained in 04.750–T; namely, sea level for the purposes of the determination of the take-off and landing distances and certain of the rates of climb and 5,000 feet for the purpose of determining the "en route" one engine inoperative rate of climb. The selection providing the greatest possible flexibility is to determine these items of performance over a range of altitude great enough to cover all anticipated routes over which the airplane may be operated.

4. *The selection of the wing flap positions desired for certification.*—04.7512–T requires that the flap control indicate a "retracted," a "take-off," an "approach," and a "landing" position and it may be noted that it is required that various items of performance be determined at each of these flap positions. It is also required that the stalling speed with the flap in the "approach" position must not exceed 85 mph, and that the stalling speed with the flap in the "landing" position must not exceed 80 mph. Obviously, no plans for flight testing may be made until these positions are selected unless the applicant wishes to investigate systematically the effect of flap position upon each or several of the items of performance which must be determined at the nominal position to be selected. This selection may therefore be seen also to be related to option 1 and possibly to option 2.

5. *The selection of the critical speed to be used in the determination of the take-off distance required by 04.7532–T.*—This is discussed in the text associated with 04.7532–T.

The practical effect of these selections is that, by making them, the applicant defines, in terms of the elements involved therein, the area to be covered by the terms of the type certificate and thereby the limits of airplane weight and airport altitude within which the airplane may be operated by a scheduled air carrier in compliance with the requirements of CAR 61.712.

Inspection and tests.—In addition to the procedure outlined under 04.03, the applicant for type certification of a transport plane should give the CAA the following information:

Prior to commencement of construction of any part of the airplane, notice of the intention to begin such construction and of the approximate date upon which it is to be undertaken should be given to the CAA in order that arrangements may be made for the necessary inspection during the course of construction of the entire airplane.

Prior to the commencement of any flying of the airplane, the applicant should notify the CAA of the approximate date upon which such flights are to begin. If this is done, the CAA representatives may observe such of this preliminary flying as may be deemed expedient in order that such observed data as may appear to be adequate for reducing the extent of official flight testing and also the amount of flying which might otherwise be necessary in order that the applicant submit a complete flight test report prior to the submission of the airplane for official flight tests.

Prior to the presentation of the airplane for the official type inspection, the applicant should submit to the CAA a proposed program which will indicate at least the following:

a. The area, defined by the several selections (described under 04.75–T) to be covered by the terms of the type certification.

b. A flight test program which will clearly indicate all of the tests it is proposed to conduct; the order in which they are to be conducted; the purpose of each such test; and for each, the airplane weight, *CG* position, flap setting, power to be drawn, and, where appropriate, the altitude, the trim speed(s) and the speed(s) or speed range to be investigated. It is suggested that this program will be most useful if, for each section of the CAR requiring flight tests, it indicates the tests proposed and also indicates in order, each flight to be made and, for each, in order, the tests to be conducted.

c. A description of the method(s) which the applicant proposes to use in order to reduce the observed data to standard conditions.

d. A statement of any intention on the part of the applicant to resort to calculation in lieu of, or for the purpose of generalizing test data, together with a description of the data upon which these calculations are to be based and the methods to be used therein.

Since it will require time for the CAA to determine the adequacy of this entire program, it is strongly recommended that it be submitted as early as practicable. Otherwise, the commencement of the testing may be delayed.

During the type inspection the applicant must, of course, make available the airplane for that purpose as well as all of the personnel and equipment necessary to obtain the required data. The CAA possesses certain indicating instruments which may be used for this purpoes such as, for example, a trailing bomb, sensitive altimeters, stop watches, carbon monoxide indicators, etc., as well as photographic equipment for measuring take-off and landing flight paths. It is, therefore, recommended that the matter of instrumentation be discussed with the CAA prior to any decision as to the detailed flight test program.

Upon the completion of the type inspection, the applicant should prepare the information necessary to show compliance with the requirements and the airplane operating manual required by 04.755 and submit them to the CAA as promptly as possible. Promptness is necessary in order that these data may be examined and a decision concerning the eligibility of the airplane for, and the appropriate terms of, type certification may be taken within the 60 day period beyond which the airworthiness certificate is valid indefinitely unless revoked. Otherwise it may be necessary to suspend the certificate at the end of the 60 days until this process can be completed. No scheduled air carrier operation of the airplane will be authorized until the process of type certification has been completed.

04.750–T Minimum requirements for certification. An airplane may be certificated under the provisions of § 04.75–T, upon there having been established, in accordance with the terms of that section: (*a*) A maximum take-off weight at sea level; (*b*) A maximum landing weight at sea level; (*c*) A maximum one-engine—inoperative operating altitude (as defined in § 04.7513–T), which shall be at least 5,000 feet at a weight equal to the maximum sea level take-off weight; (*d*) Take-off characteristics at maximum sea level take-off weight, and landing characteristics at maximum sea level landing weight, in accordance with the provisions of §§ 04.7532–T and 04.7533–T; and (*e*) Compliance with the requirements of all other applicable parts of the Regulations.

If a certificate is issued under these conditions, it may be amended from time to time to include landing and take-off weights over an increased range of altitudes, and other pertinent performance data, including additional landing and take-off characteristics obtained in accordance with the provisions of §§ 04.7532–T and 04.7533–T.

The primary significance of this requirement is that it provides means by which an airplane may be certificated in the Transport Category even though it may not subsequently be placed in scheduled passenger carrying operation, and, as a result of such, be subjected to the operating rules of CAR 61.712. Items (*a*), (*b*), (*c*), and (*d*), simply define a minimum number of weights and altitudes at which the performance must be determined. Item (*e*) means simply that the airplane must, at these weights and altitudes, comply with all the applicable requirements (see 04.01, 04.70, and 04.75–T).

The amendment to the certificate to which the last paragraph of this requirement refers will in practice be the revision of the Airplane Operating Manual required by 04.755–T to include the performance information corresponding with other weights or altitudes.

04.751–T Definitions.

04.7511–T Stalling speeds. In the following sub-sections of § 04.75–T:

V_{s0} denotes the true indicated stalling speed of the airplane in miles per hour with engines idling, throttles closed, propellers in low pitch, landing gear extended, flaps in the "landing position," as defined in § 04.7512–T, cowl flaps closed, center of gravity in the most unfavorable position within the allowable landing range, and the weight of the airplane equal to the weight in connection with which V_{s0} is being used as a factor to determine a required performance.

V_{s1} denotes the true indicated stalling speed in miles per hour with engines idling, throttles closed, propellers in low pitch, and with the airplane in all other respects (flaps, landing gear; etc.) in the condition existing in the particular test in connection with which V_{s1} is being used.

The text of this requirement is believed to be quite clear. The speeds are so formally defined because they become the basis for the specification in the following regulations of other speeds as multiples of these and required rates of climb as multiples of their squares. These stalling speeds are defined as "throttles closed" because greater consistency may be expected among the results of successive attempts to measure them than from attempts when appreciable power is drawn and they are, therefore, more useful for their basic purpose. Since the effect of power is ordinarily to decrease the stalling speed, they are also conservative for these purposes.

All stalling speeds required to be investigated are measured by means of flight tests by first calibrating the airplane air speed indicator by flying over a measured course at various indicator readings or using a trailing bomb, then stalling the airplane, following the procedure of 04.7543–T. The stalling speed is the lowest indicator reading observed during steady controlled flight following this process, corrected by the results of the calibration.

04.7512–T Flap positions. The flap positions denoted respectively as the landing position, approach position, and take-off position are those provided for in § 04.434–T, and may be made variable with weight and altitude in accordance with that section.

The flap positions corresponding with the flap control settings required by 04.434–T are open to selection by the applicant. The only regulatory restrictions upon this selection are that, once made, the weight at which the airplane may be operated must not exceed that at which the maximum stalling speeds (04.7530–T) and the minimum rates of climb (04.7531–T) are equalled, or, in scheduled operation, that at which the take-off (CAR 61.7122) and landing (CAR 61.7123) limitations are equalled.

The simplest possible choice would appear to be the retracted flap position for "take-off" and "approach" and the fully extended position for "landing". This conforms to most of the past practice in the use of flaps and requires a comparatively simple control mechanism. If, however, the airplane has been so designed that the maximum take-off and landing weights are limited by the required rates of climb at sea level, these weights will be reduced more rapidly with increasing altitude than would otherwise be required since the effect of a given small change in flap position upon the maximum weight permitted by the climb requirements is generally appreciably greater than its effect upon the maximum weight determined by the stalling speed requirements. Also, it is very possible that, for an airplane so designed, the distance required to accelerate during take-off to the minimum required speed and stop, may, at a given weight (less than the maximum), greatly exceed the distance required to reach a height of 50 feet. (See 04.75320–T and 04.75321–T.) Since the scheduled operation of the airplane will be limited to weights such that the greater of these two distances does not exceed the length of the take-off runway, it may be advantageous to use a "take-off" flap position such as to make these two distances more nearly equal. This could be done by using some flap extension since with increasing extension the speed to which the airplane must be accelerated may be reduced (see 04.75320–T) and the drag during the deceleration run will probably be increased both of which will, at a given weight, reduce the accelerate-stop distance. Flap extension would also decrease the stalling speed, and, therefore, the required rate of climb, as well as decrease the available rate of climb, but the net result would normally be an increase in the distance required to attain 50 feet of height. The optimum "take-off" flap position for a given weight; i. e., the position which would require a minimum runway length, would be a itio which would make these two distances equal.

The most complicated selection would be one permitting the airplane to carry the maximum possible weight allowed by the operating limitations of CAR 61.712 at any altitude for any possible runway length. Such flexibility in operation would, however, require secondary flap control systems of such complexity as to present a substantial design problem.

The selection of flap positions will also influence the nature and extent of the flight testing required for type certification. If, for example, a single flap position be used for each of the required control settings, it is only necessary to measure the stalling speeds at one weight and rates of climb at such weights and altitudes as may be necessary to cover the range of each for which the applicant wishes the airplane certificated. If, on the other hand, a number of flap positions are to be used for one or all of the required control settings, dependent upon altitude, runway length or other parameter, it will be necessary to measure stalling speed and climbing and other performance at each such position or at a sufficient number of these to permit the determination of the performance at all of them by interpolation or other suitable means.

04.7513–T Maximum one-engine-inoperative operating altitude (to be determined in complying with § 04.723) shall be the altitude in standard air at which the steady rate of climb in feet per minute is 0.02 $V_{s_0}^2$ with the critical engine inoperative, its propeller stopped, all other engines operating at the maximum-except-take-off power available at such altitude, the landing gear retracted, and the flaps in the most favorable position.

This requirement in its effect defines a minimum rate of climb which is acceptable as evidence of the ability of the airplane to maintain altitude in spite of the failure of an engine during cruising flight. The altitude, in standard air, at which this rate of climb exists under the conditions here specified becomes the basis for the operating rule specified in CAR 61.7125. The required rate of climb has been related to the stalling speed because the ability of the airplane to reach an intended point for landing is approximately proportional to the rate of climb; the failure to do so may result in a forced landing during which any damage done to the airplane or its contents is assumed to be related to the kinetic energy which must be absorbed; and this energy is proportional to $V_{s_0}^2$.

04.723 also applies to the Transport Category and the data obtained from the testing required by that requirement are adequate to establish the maximum one-engine-inoperative operating altitude and its variation with airplane weight. The process by means of which this may be accomplished is described in Flight Engineering Report No. 9. The testing required at each altitude and weight is outlined in 04.723. The nature of this testing and an acceptable method by means of which to reduce the observed data to standard conditions is described in Flight Engineering Report No. 3.

The number of weights and altitudes at which tests must be conducted depends upon the range of weight and altitude to be covered, the altitude characteristics of the powerplant, the consistency of the results of the testing actually conducted, etc. The number must, however, be great enough to establish at least the following: (1) The critical altitude(s) of the engines for METO power; (2) the apparent aerodynamic characteristics of the airplane; i. e., airplane efficiency factor "e" and parasite area "f" or their equivalent; (3) the rate of change with altitude of best rate of climb between all critical points in the altitude history of METO power.

The absolutely minimum amount of testing which appears likely to do this is to conduct at one intermediate weight a five speed sawtooth climb at not less than two altitudes in any full throttle range between adjacent pair of critical points in the altitude characteristics of the powerplant, independently to determine all critical altitudes, and thus to establish rate of climb versus altitude for that weight. Experience has indicated that it is necessary to carry out the climb at each speed during these sawtooth climbs for at least five minutes in order to obtain dependable results. The effect of weight may then be calculated, by the method of Flight Engineering Report No. 10 or its equivalent, at each of the critical points to establish the altitude history of climb at other weights.

It is recommended that compliance with this requirement be shown with the critical flap position required to meet the powerplant cooling requirements of 04.640. The applicant may, however, show compliance with the cowl flap position required for cooling in standard air. This will of course require additional cooling tests to establish this latter cowl flap position.

04.752–T Weights. The maximum take-off weight and maximum landing weight shall be established by the applicant and may be made variable with altitude. The maximum take-off weight for any altitude shall not exceed the maximum design weight used in the structural loading conditions for flight loads (§ 04.21), and shall not exceed the design weight used in the structural loading conditions for ground or waterloads (§§ 04.24 and 04.25, respectively) by a ratio or more than 1.15. The maximum landing weight for any altitude shall not exceed the design weight used in the structural loading conditions for ground or water loads.

The regulatory limitations upon the weight at which Transport Category airplanes may be certificated are conveniently summarized beginning on page 2 of Flight Engineering Report No. 9, which indicates all the factors to be considered by the applicant prior to selecting the design weights. This section requires the design landing weight to be not less than 87 percent of the design take-off weight and that 04.7520–T requires the installation of dump valves if the take-off weight exceeds the landing weight. Although not explicitly stated, the regulations as well as practical considerations, indicate the necessity, in all cases where the take-off weight exceeds the landing weight, of providing fuel capacity for a weight of fuel in excess of the maximum spread between any take-off and landing weight to be authorized for any scheduled flight. Where it is desired to use a take-off weight that exceeds the maximum allowable landing weight by more than 15 percent, this may be accomplished by designing the airplane for landing loads corresponding with a weight not less than 87 percent of the take-off weight, and by providing fuel dumping facilities adequate to reduce the weight from the take-off to the landing weight in ten minutes.

04.7520–T Fuel dumping provisions. If the maximum take-off weight for any altitude exceeds the maximum landing weight for the same altitude, adequate provision shall be made, in accordance with § 04.6, for the rapid and safe dumping during flight of a quantity of fuel sufficient to reduce the weight of the airplane from such maximum take-off weight to such maximum landing weight. Compliance with this requirement shall be shown by dumping suitable colored fluids and fuel in flight tests in the following conditions: (*a*) Level flight at a speed of 2.0 V_{s_1}; (*b*) climb at a speed of 1.4 V_{s1} with 75 percent of maximum-except-take-off power; (*c*) glide with power off at a speed of 1.4 V_{s_1}.

In conditions (*a*) and (*b*), the time required to dump the necessary amount of fuel shall not exceed 10 minutes. During such tests, the dumped fluid shall not come in contact with any portion of the aircraft or adversely affect its control; nor shall any fumes from such fluid enter any portion of the aircraft.

The basic purpose of these tests is to determine that the required amount of fuel may be safely jettisoned under reasonably anticipated operating conditions within the prescribed time limit without danger from fire, explosion or adverse effects on flight characteristics.

To demonstrate compliance with the fuel dumping time limit, the tanks should be so selected and loaded that the result will be the most critical condition permissible within the contemplated fuel loading limitations. An appropriate placard will be required in the event fuel dumping with the flaps extended results in any adverse effect on the flow pattern or the flap structure. In the event that the fuel flow pattern may be adversely affected when the critical engine is inoperative and its propeller feathered or stopped, additional tests shall be conducted in this latter configuration at a speed of $1.4V/_{s1}$ with METO horsepower on the remaining engine(s). The time limit as specified above will also be applicable in this case. During the tests, reasonably anticipated rough air conditions will be simulated.

04.753–T Required performance and performance determinations. Performance data shall be corrected to standard atmosphere and still air where such corrections are applicable. Performance data may be determined by calculation from basic flight tests if the results of such calculation are substantially equal in accuracy to the results of direct tests.

The items of performance which are governed by minima or otherwise required to be determined hereunder, all involve one or more of the following basic regimes: (1) The airplane being accelerated along the ground from rest to some specified airspeed; (2) The airplane climbing or gliding steadily; (3) The airplane being decelerated along the ground from some specified airspeed, or a contact speed during landing to rest.

The fundamental nature of these operating regimes is indicated in Flight Engineering Reports Nos. 7, 3, and 1, respectively, which also contain or discuss acceptable methods by means of which to correct the observed data to standard conditions. Flight Engineering Reports Nos. 9 and 10 indicate methods by means of which the required performance may be calculated for various weights and altitudes once it is established for one weight and certain altitudes and the former of these two reports sheds considerable light upon the nature and extent of the testing required to establish the required performance.

The take-off flight path, specified hereunder, ignores the transition which, in any real take-off, must take place between linear acceleration along the runway to steady climbing flight because no significant error is ordinarily introduced by doing so. The landing flight path does not, however, ignore the corresponding transition from steady gliding flight to deceleration along the ground. The nature of these transitions is discussed in an article, "An Approximate Method to Predict the Transition or 'Flare' Flight Path in the Take-off or Landing of an Airplane" in the November 1941 issue of the Journal of the Aeronautical Sciences.

The only information available to the CAA concerning the accuracy of the results of direct tests undertaken to determine any one of these three basic items of performance is the consistency of the results of successive attempts to measure any of them. The results of such tests for the distance required to accelerate along the ground to some selected speed indicate very great consistency. Figure 19 of Flight Engineering Report No. 7 shows, for example, that when the results of five separate measurements are plotted as velocity against distance, the points for all practical purposes, define a single curve. This is believed to indicate comparatively great accuracy in establishing such a distance. The case of rate of climb is not so fortunate. The information available is by no means conclusive, but it does indicate the rate of climb cannot be established by direct testing within such close limits of precision, and that rather extensive flight test data are likely to be necessary in order to establish representative values of this item of performance. The case of deceleration along the ground has also proven to be troublesome. There are really two cases to be considered. The first of these is the deceleration following contact during a landing. Flight Engineering Report No. 1 contains the results of 38 landings made with one airplane, and the discussion beginning on page 39 of that report, plus the tabulation of corrected results on page 67 of the report, will indicate the order of consistency of the effective constant deceleration involved. The other case is deceleration following the abrupt stopping of the engines simulating the case of engine failure during the take-off. Flight Engineering Report No. 7 contains the results of several attempts to measure this distance from various speeds. These results appear in figure 24 of that report. There are a number of reasons for the dispersion of these results, the more important of which appear to be the condition of the brakes and the consistency of the pilot technique.

The variation of the probable accuracy of the measurement of these three basic items of performance produce a corresponding variation in the amount of direct testing which would be required to establish them. In the case of the distance required to accelerate to a speed, for example, it is believed the results of three separate attempts would probably be definitive. The amount of testing required to establish the rate of climb at one weight throughout the range of altitude has already been discussed under 04.7513–T above. It is difficult to be very specific concerning the amount of testing required to establish the distance required to decelerate from a given speed. It is considered unlikely, however, that a representative value could be established by means of less than five consecutive attempts to measure the distance required to decelerate from a given speed.

04.7530–T Stalling speed requirements. **(*a*) V_{s_0} at maximum landing weight shall not exceed 80 miles per hour; (*b*) V_{s_1} at maximum landing weight, flaps in the approach position, landing gear extended, and center of gravity in the most unfavorable position permitted for landing, shall not exceed 85 miles per hour.**

The limitations which are imposed by the above requirements upon the stalling speeds have been dictated primarily by the effect of the speeds at which it is necessary that the airplane be operated during an approach and landing under adverse weather conditions, upon the safety of that operation. All of the airplanes which had been used in civil operation had, at the time these regulations were written, been designed to comply with a maximum landing speed requirement which had in no case exceeded 70 mph where passengers were to be carried in the airplane. Even though there are contained in the entire set of Transport Category requirements other but less direct limitations upon these stalling speeds, it was considered unwise to abandon any direct limitation because the alternative limitations are not absolute and it is, therefore, possible by the installation of a sufficient amount of power in an airplane and by the provision of airports having runways long enough, to design an airplane having stalling speeds far in excess of those with which we have been familiar.

The purpose of (*a*) of this requirement is to prevent the necessity of making contact at very high speeds during landing. The purpose of (*b*) is basically to provide a margin of speed between the stalling speed and the maximum which it is believed may be safely used during the instrument let-down and approach procedure. This margin is such as to provide for an acceleration of 2_g, corresponding with a 60° banked turn at 120 mph without stalling. It should be noted that these are true indicated airspeeds as defined in 04.7511–T.

These stalling speeds must be established by flight tests at at least one airplane weight, ordinarily the maximum landing weight.

04.7531-T Climb requirements. In the climb tests required by this section, the engine cowl flaps, or other means of controlling the engine cooling air supply, shall be in a position which will provide adequate cooling with maximum-except-take-off power at best climbing speed under standard atmospheric conditions.

(a) *Flaps in landing position.*—The steady rate of climb in feet per minute, at any altitude within the range for which landing weight is to be specified in the certificate, with the weight equal to maximum landing weight for that altitude, all engines operating at the take-off power available at such altitude, landing gear extended, center of gravity in the most unfavorable position permitted for landing, and flaps in the landing position, shall be at least 0.07 V_{s0}^2.

(b) *Flaps in approach position.*—The steady rate of climb in feet per minute, at any altitude within the range for which landing weight is to be specified in the certificate, with the weight equal to maximum landing weight for that altitude, the critical engine inoperative, its propeller stopped, all other engines operating at the take-off power available at such altitude, the landing gear retracted, center of gravity in the most unfavorable position permitted for landing, and the flaps in the approach position, shall be at least 0.04 V_{s0}^2.

(c) *Flaps in take-off position.*—The steady rate of climb in feet per minute, at any altitude within the range for which take-off weight is to be specified in the certificate, with the weight equal to maximum take-off weight for that altitude, the speed equal to the minimum take-off climb speed permitted in §04.75320–T (b), the critical engine inoperative, its propeller windmilling with the propeller control in a positions which would allow the engine (if operating normally and within approved limits) to develop at least 50 percent of maximum-except-take-off engine speed, all other engines operating at the take-off power available at such altitude, the landing gear retracted, center of gravity in the most unfavorable position permitted for take-off, and the flaps in the take-off position, shall be at least 0.035 V_{s1}^2.

Stalling speed	Rate of climb		
	$0.02V_s^2$	$0.04V_s^2$	$0.07V_s^2$
50	50	100	175
55	61	121	212
60	72	144	252
65	85	169	296
70	98	196	343
75	113	225	394
80	128	256	448

This requirement specifies a minimum rate of climb for each of three airplanes' configurations, representative of three different operating conditions certain to be encountered in the use of the airplane. These rates have been specified with two basic purposes in mind. The first of these, is that they are believed to be a minimum which will guarantee the ability of the airplane to perform any necessary maneuver safely. The other is that they penalize the extreme use of wing flaps for the purpose either of complying with the stalling speed requirements, or of establishing the minimum possible take-off distance at the greatest possible weight. The use of a comparatively great amount of flap during the landing increases the difficulty in executing the landing: first, because it increases the rate of descent at a given approach speed, and second, because it results in a trim with the nose of the airplane lower than would be required without or with lesser flap such that the rotation of the airplane about its lateral axis, which is involved in the final stage of a landing, must take place over a greater angular range. These rates of climb are also related to stalling speed for essentially the same reason as has been stated in the discussion of 04.7513–T.

The nature and extent of flight testing required to establish these rates of climb depend upon the range of weight and altitude for which it is desired the airplane be certificated. If a moderate range of altitude (sea level to 5 or 6,000 feet) is involved, and if, also, the testing specified under 04.7513–T has been conducted, a five speed sawtooth climb at a single weight and altitude for each of these three configurations will probably be adequate to establish values of airplane efficiency factor e and airplane parasite area f. It should be noted, however, that the effect of the slipstream upon the apparent aerodynamic characteristics of the airplane will vary with altitude and if there is any great variation in horsepower available over the range of altitude involved it may be necessary to investigate more than one altitude.

Flaps in landing position.—The airplane configuration here specified is that ordinarily used in the final stages of an approach for landing and the purpose of the rate of climb here specified is to insure that the airplane be able to "go around" for another attempt at landing in the event conditions beyond the control of the pilot makes this necessary.

Flaps in approach position.—This airplane configuration is intended to represent conditions which would probably occur during an approach for landing with one engine inoperative and the rate of climb is intended to insure that the airplane be able to "go around" in the event this becomes necessary.

Flaps in take-off position.—This airplane configuration is intended to be representative of the conditions which might be expected to occur in the event of engine failure at about the time the airplane leaves the ground during take-off, and the rate of climb specified is believed to be a reasonable minimum which will insure the ability of the airplane to continue the take-off under such circumstances. The landing gear is specified as retracted on the assumption that the retraction will be started at practically the instant of leaving the ground. The inoperative propeller control is specified in a position which would allow the engine to develop at least 50 percent of maximum except take-off engine speed upon the application of power, upon the assumption that a pilot would almost automatically pull the propeller pitch control to a comparatively low rpm setting but not so low that, in the event the engine recovered, practically no thrust would be available due to the high pitch setting of the propeller.

The determination of the allowable propeller control setting for the purposes of compliance with this requirement will ordinarily require that the airplane be flown at the speed involved in order to determine whether the constant speed control will govern at 50 percent of maximum except take-off rpm while drawing power at the limiting manifold pressure, the maximum cruising BMEP recommended by the engine manufacturer, or full throttle whichever shall occur first as the throttle is opened. If so, this propeller control position must be noted and the tests to establish the rate of climb conducted with the control of the inoperative engine set to this same position. If the constant speed control will not govern while drawing power as described above; i. e., if, with the control in the "minimum rpm" position, 50 percent of METO rpm be exceeded, the climb testing will be conducted with the control in that position.

If the design or operation of the propeller control mechanism is such as to render the above procedure inapplicable, consideration will be given to any alternative which conforms with the basic purpose of the requirement, this purpose being to permit such betterment of the rate of climb as might result from a single and practically instantaneous operation of some single propeller control which would not, however, permit the pitch of the propeller to exceed that established above drawing power; i. e., that pitch corresponding with 50 percent METO rpm under the specified throttle setting.

04.7532-T Take-off determination. The following take-off data shall be determined over such range of weights and altitudes as the applicant may desire, with a constant take-off flap position for a particular weight and altitude, and with the operating engines at not more than the take-off power available at the particular altitude. These data shall be based on a level take-off surface with zero wind.

The purpose of this requirement is to establish the dimensions of a take-off flight path such that if the take-off runway have a length equal to the greater of two possible dimensions of that flight path, an engine failure may be suffered by the airplane at any point within the length of that runway and the airplane be able either to stop within the length of the runway or to continue and clear all obstructions to flight until a safe landing may be made. Such a flight path necessarily involves consideration of the level of skill of the pilot who happens to be flying the airplane at the time. It has been attempted in the specification of the conditions under which this flight path is to be established, to provide for a reasonable level of skill by specifying certain minimum speeds which must be attained, as well as a sequence and timing in which it is assumed various adjustments may be made to the airplane, each of which would have measurable effect upon the resulting dimensions. These particulars are discussed below.

04.75320-T Speeds. (*a*) Critical-engine-failure speed, denoted by V_1, is a true indicated airspeed, chosen by the applicant but in any case not less than the minimum speed at which the controllability is adequate to proceed safely with the take-off, using normal piloting skill, when the critical engine is suddenly made inoperative. (*b*) Minimum take-off climb speed, denoted by V_2, is a true indicated airspeed chosen by the applicant, which shall permit the rate of climb required in §04.7531-T (*c*) but which shall not be less than 1.20 V_{s_1} for two-engined airplanes, or 1.15 V_{s_1} for airplanes having more than two engines, or less than 1.10 times the minimum speed at which the airplane is fully controllable in flight using normal piloting skill when the critical engine is suddenly made inoperative.

This requirement, as its text indicates, contemplates a speed at which the engine may be assumed to fail which may be lower than a speed at which flight is possible. Since the operating requirements of CAR 61.712 limit the take-off operation of the airplane to a weight such that in the event of engine failure, at the critical engine failure speed, the airplane may either be brought to rest within the length of the field or continue the take-off and attain a height of 50 feet at the edge of the field. It follows that for any airplane at a particular weight there is an optimum value of this critical engine failure speed which will produce the minimum required runway length and further, that this optimum condition obtains when the two alternative distances are equal. In the case of an airplane having a comparatively high wing loading but low power loading, and particularly in the case of airplanes with four or more engines, this optimum may be appreciably below a speed at which flight is possible. If such a speed is to be used for the purposes of establishing the dimensions of the flight path, it is obviously necessary that it be possible to continue the take-off acceleration after the failure of an engine until a minimum safe flying speed has been attained and it must be demonstrated by test that it is possible safely to do this. During the demonstration, the throttling of an opposite engine will not be permitted unless: (1) the engine is completely throttled, or (2) a satisfactory automatic means for such throttling is available. It appears open to some question whether any airplane equipped with a tail wheel landing gear arrangement can do this, although it does appear possible that an airplane equipped with a nose wheel landing gear might do so.

The second speed specified is a minimum speed at which it is considered safe to attempt to complete the take-off with one engine inoperative. The limitation upon this speed, based upon stalling speed, involves the power-off stalling speed and the 20 percent and 15 percent margins are arbitrarily specified as a reasonable minimum to insure against inadvertent stalling of the airplane. The difference between the two margins, based upon the number of engines installed in the airplane, is due to the fact that the application of power ordinarily appreciably reduced the stalling speed. In the case of the two engine airplane, at least half of this reduction is eliminated by the failure of an engine. In the case of a four engine airplane, certainly less than half and probably more nearly one-quarter only of the difference is eliminated by the failure of an engine. The difference in the required factors, therefore, provides approximately the same margin over the actual stalling speed under the power conditions which obtain after the loss of an engine no matter what the number of engines (in excess of one) may be.

This minimum take-off climb speed is also required to be at least 10 percent in excess of the minimum speed at which the airplane can be safely controlled when the critical engine is suddenly made inoperative. Flight tests are required to determine this latter speed and it is suggested that the most direct testing technique would consist of making the critical engine inoperative at a speed above the probable minimum and then slowly reducing the airplane speed until a minimum is reached. This minimum should then be checked by flying steadily at the speed so determined with all engines operating, closing the throttle on the critical engine, and verifying that control can be maintained. For the purposes of compliance with this requirement, the rudder pedal force required, without adjustment of the trim, to hold the airplane steadily upon a heading with zero yaw shall not exceed 180 pounds. This condition of trim will ordinarily involve two or three degrees of roll, the wing on the side of the operating engines being down. It should also be noted that during the test required to verify the minimum controllability speed, appreciably greater roll than this may be required momentarily in order to regain control. This is acceptable so long as control can be established with reasonable promptness at the same speed and without loss of altitude. Finally, it should be noted that 04.7540-T requires that this speed be not more than 1.20 V_{s_1}.

The selection by the applicant of the two speeds specified will influence the nature of the testing required to establishing the take-off flight path specified. If the critical engine failure speed, V_1, is equal to, or in excess of, the minimum take-off climb speed, V_2, the segment of the take-off flight path involving acceleration along the ground may be determined separately from those succeeding segments which involve steady climbing flight. If, however, V_1 is less than V_2, the testing must involve an actual take-off acceleration during which the critical engine is made inoperative at the speed V_1, but the acceleration continued at least until the speed V_2 is attained and the resulting flight path recorded photographically.

In either event, the critical speed thus determined should be included in the Operating Manual (see 04.755-T).

04.75321-T Take-off path. The lengths and slopes of segments of the take-off path, and the location of critical points on the complete path shall be determined in accordance with the following conditions and assumptions. The location of the points defined below shall be expressed in terms of the horizontal and vertical distances from the starting point.

(*a*) Starting point. The point from which a standing start is made with all engines operating.

(*b*) Critical-engine-failure point. The point at which the airplane attains speed V_1 (critical engine failure speed) when accelerated from point (*a*) with all engines operating.

(*c*) Accelerate-and-stop point. The point on the take-off surface at which the airplane can be brought safely to a stop if all engines are cut at point (*b*).

(*d*) Start-of-climb point. The point on or just clear of the take-off surface at which the airplane attains speed V_2 (take-off-climb-speed) when the critical engine is made inoperative with its propeller windmilling in low pitch at point (*b*).

The take-off acceleration segment, (*a*) to (*d*), shall be determined by making a continuous run up to speed V_2 with the critical engine cut at point (*b*).

(*e*) Retraction-completion point. The point at which landing gear retraction is completed when retraction is initiated not earlier than point (*d*).

The initial climb segment, (*d*) to (*e*), shall be assumed to correspond to the rate of climb at speed V_2 with landing gear extended and windmilling propeller in low pitch.

The second climb segment, beginning at point (*e*), shall be assumed to correspond to the rate of climb at speed V_2 with landing gear retracted and windmilling propeller in high pitch, as defined in § 04.7531-T (c). This segment may continue indefinitely or may end at point (*g*) in accordance with paragraph (*g*) following.

(*f*) 50-foot height point. The point at which the airplane attains a height of 50 feet (above the take-off surface) along the take-off flight path defined herein.

(*g*) Feathering-completion point. The point where feathering or stopping of the inoperative propeller is completed, if the applicant desires to include this step in the take-off determination. It shall be assumed that the decision to feather or stop is made not earlier than the instant of attaining point (*f*).

In the event that it is desired to include propeller feathering or stopping in the take-off path, the final climb segment, beginning at point (*g*), shall be assumed to correspond to the rate of climb at speed V_2 with landing gear retracted and the propeller of the inoperative engine feathered or stopped.

The take-off path specified above, involves the determination of the distances traversed by the airplane for two alternative sequences of events. The first of these is that the airplane be accelerated to the critical engine failure speed, at which speed all engines are made inoperative and the airplane decelerated to rest. The second sequence of events is that the airplane be again accelerated to the same speed but that at that speed, the critical engine only be made inoperative and the take-off continued under certain specified conditions. The distance required to accelerate to the speed V_1 is thus common to both sequences.

The first of these assumed sequences of events will be called hereafter the "accelerate-stop" distance. It is obvious that when the throttles are suddenly closed, a finite time will elapse before the propellers and the rotating parts of the engine are decelerated from the take-off rpm at which they are, prior to closing the throttle, being driven, to an idling rpm. During this period the propellers continue to exert thrust, until a certain zero thrust rpm is reached as a result of the deceleration. For this reason the speed of the airplane continues to increase beyond that which exists at the moment the throttles are closed before it begins to decrease again. The period of time covered by this deceleration of the engine rpm is also a very critical period for the application of brakes since there usually results a change in trim of the airplane and certain adjustments in the position of the controls must be made. For both of these reasons it appears necessary, in order to establish a representative dimension for the distance that would be required in the event of an actual failure of an engine during take-off and the election of the pilot to stay on the ground, to conduct tests involving a continuous run starting from rest and ending at rest rather than to determine separately the distances required to accelerate to the selected speed and the distance required to decelerate from this speed when this latter maneuver is performed as a part of a landing. This process ordinarily requires a comparatively long runway and in some cases it has been found impossible to attain the required speed and stop within the available length of a runway. If this cannot be avoided, it then becomes necessary to determine the total distance involved in accelerating to various lower speeds such as to permit extrapolation of the results to the required speed. In determining this distance, the wing flap shall be in the take-off position at least until the engines have been made inoperative but they may thereafter be extended to aid the deceleration if it is demonstrated by the applicant that this may be done with reasonable ease and with safety.

The second sequence of events identified above produces what will hereafter be called the take-off flight path. This is made up of the following segments:

(1) Acceleration along the ground. This has been discussed in the foregoing.

(2) Steady climbing flight with critical engine inoperative. The inoperative propeller idling with its pitch control in the take-off position and the landing gear extended for the length of time required to retract the landing gear.

(3) Steady climbing flight with the landing gear retracted and the inoperative propeller pitch control in the position permitted by 04.7531–T (*c*) for the length of time required to attain a height of 50 feet above the take-off surface and thereafter, for the length of time required to feather the propeller.

(4) Steady climbing flight with the inoperative propeller feathered for the length of time which may remain of the time limit upon the use of take-off power. Cases have been encountered in which propeller feathering is incomplete at the end of one minute from the start of the take-off and in such cases the flight path has been based upon the assumption that take-off power would continue to be drawn during an emergency such as would exist following the failure of an engine during the take-off, until the propeller is in fact feathered.

(5) Steady climbing flight with the inoperative propeller feathered drawing maximum except take-off power upon the operating engine(s) indefinitely thereafter.

It may be noted that items (*e*), (*f*), and (*g*), of this requirement offer to the applicant the alternative of basing the remainder of the flight path upon the conditions specified in the third segment discussed above.

The various conditions which have been specified in this requirement to govern the configuration of the airplane assumed to exist in each of the various climb segments have been intended to reflect as closely as possible, the probable order in which a pilot would make changes in the airplane configuration in the actual case of engine failure or, are conservative in their nature in an effort to simplify the testing required to establish the flight path. Thus, for example, it is assumed that the pilot will do nothing except raise the landing gear for the length of time required for the gear fully to retract and it is also assumed that insofar as its effect upon the climbing performance is concerned the gear is down throughout this period. It is assumed that the pilot would make some adjustment of the propeller pitch control but that he would not initiate the operation of feathering prior to attaining a height of 50 feet which, in the limiting case permitted by the operating requirements, would also be the far end of the takeoff runway; and, further, it is assumed that performance of the airplane remains that with the propeller idling until the feathering is complete.

The take-off flap setting must be maintained throughout the determination of the take-off flight path.

Subject only to the restriction involving a value of V_1 less than V_2 which is covered by item (*d*) of this requirement and has been discussed in 04.75320–T, each of these segments of the take-off flight path may be separately established, one of them (segment No. 3) has already been established in order to show compliance with 04.7531–T (*c*). The extent and nature of the testing required to establish the others is substantially identical with that required for the third segment which has been discussed.

04.7533–T Landing determination. The horizontal distance required to land and come to a complete stop from a point at a height of 50 feet above the landing surface shall be determined for such range of weights and altitudes as the applicant may desire. In making this determination:

(*a*) Immediately prior to reaching the 50-foot altitude, a steady gliding approach shall have been maintained, with a true indicated airspeed of at least 1.3 V_{s_0}.

(*b*) The nose of the airplane shall not be depressed, nor the power increased, after reaching the 50-foot altitude. At all times during and immediately prior to the landing, the flaps shall be in the landing position, except that after the airplane is on the landing surface and the true indicated airspeed has been reduced to not more than 0.9 V_{s_0} the flap position may be changed.

(*c*) The operating pressures on the braking system shall not be in excess of those approved by the manufacturer of the brakes.

(*d*) The brakes shall not be used in such manner as to produce excessive wear of brakes or tires.

(*e*) The landing shall be made in such manner that there is no excessive vertical acceleration, no tendency to bounce, nose over, porpoise, ground loop, or water loop, and in such manner that its reproduction shall not require any exceptional degree of skill on the part of the pilot, or exceptionally favorable conditions. If this last condition (with respect to exceptional skill or favorable conditions) is not met, the distance to be determined shall be that considered to correspond to a piloting technique normally usable.

The purpose of this requirement is to specify a distance which is required from a point 50 feet above the take-off surface to land and bring the airplane to rest, which is at once representative of actual operating technique and may serve as the basis for the specification of a landing runway length within which a pilot of average skill may reasonably be expected to be able to land the airplane safely under the most adverse weather or other operating conditions likely to be encountered in the actual operation.

The minimum approach speed of 1.3 V_{s_0}, contained in item (*a*), is intended to provide a reasonable margin above the stalling speed. Items (*b*) and (*e*) are concerned primarily with the prevention of an attempt to place the airplane in contact with the runway surface at a very high speed in order to take advantage of the greater deceleration provided by most wheel brake installations than is available from the drag of the airplane while still airborne. Flight Engineering Report No. 1 covers an investigation undertaken to determine the effect of various factors which were considered in drafting this requirement and indicates the critical dependence of the landing distance here defined upon the contact speed. Obviously it will defeat the purposes of this requirement if a distance be obtained by making contact at so high a speed as to require an exceptional degree of skill on the part of the pilot, or to base a distance upon exceptionally favorable conditions such as wind or the nature of the surface of the runway. Item (*e*) forbids it for this reason.

Items (*c*) and (*d*) are concerned primarily with the extent to which wheel brakes may be appropriately used in establishing this distance. "Excessive wear" is considered (1) skidding of a tire, (2) excessive heating of the brakes which requires replacement during a series of five official test landings, and (3) nose-over to the extent of causing a tail wheel to leave the ground surface after its contact.

It should be noted also that compliance with item (*c*) requires the establishment of a maximum recommended pressure by the manufacturer of the brakes and the installation of a measuring device by means of which the pressure actually used may be observed. It is recommended that a statement of the approved operating pressure be obtained from the manufacturer of the brakes and submitted as a part of the test program. The measurement of this landing distance is most satisfactorily done by photographic means and the CAA is equipped with such means. The field set-up for the use of this equipment and the equipment itself, as well as the method of reading and analyzing results, are covered by Flight Engineering Reports Nos. 4 and 8.

Tests to establish the landing distance must be conducted with the most forward *CG* position to be approved by the terms of the type certificate.

04.754–T Flight characteristics. There shall be no flight characteristic which makes the airplane unairworthy The airplane shall also meet the following requirements under all critical loading conditions within the range of center of gravity, and, except as provided in 04.7541–T (*d*), at the maximum weight for which certification is sought

The relation between flight characteristics and safety involves the level of skill and the scope, character, and intensity of attention required on the part of the pilot to fly the airplane at all. It is theoretically possible to design an airplane which cannot be flown by a single pilot because, for example, he may have not enough strength, or alternatively a sufficiently delicate touch, to operate the controls or there may be so many necessary operations that he cannot, within the required time, perform them all. The purpose of the requirements contained in this and succeeding sections is to insure that a pilot, or any appropriate crew, may operate the airplane with enough margin of comfort that the performance of the airplane which has been covered by the preceding requirements may be realized and a reasonable measure of safety maintained.

The succeeding material covers only the more important of the flight characteristics under what have been agreed to be critical or representative flight regimes. It is the purpose of the first sentence of 04.754-T to cover all those which have not been and probably cannot be anticipated for all airplanes and therefore covered by specific requirements. The succeeding requirements will in most cases indicate what are considered to be the critical loading conditions, and, assuming the airplane to be found in compliance with those requirements at those loading conditions, no investigation of other loading conditions will ordinarily be required.

04.7540-T Controllability and maneuverability. The airplane shall be controllable and maneuverable during take-off, climb, level flight, glide, and landing, and it shall be possible to make a smooth transition from one flight condition to another, without requiring an exceptional degree of skill, alertness, or strength on the part of the pilot, under all conditions of operation probable for the type, including those conditions normally encountered in the event of sudden failure of any engine. It shall be possible, with power off, with flaps either retracted or in the landing position, with the center of gravity in the most unfavorable location within the certificated range, and with the airplane trimmed for a speed of 1.4 V_{s_1}, to change the flap position to the opposite extreme, to make a sudden application of take-off power on all engines, or to change the speed to any value between 1.10 V_{s_1} and 1.70 V_{s_1}, without requiring a change in the trim control or the exertion of more control force than can readily be applied with one hand for a short period. It shall not be necessary to use exceptional piloting skill in order to prevent loss of altitude when flap retraction from any position is initiated during steady horizontal flight at 1.1 V_{s_1}, with simultaneous application of not more than maximum-except-take-off power.

The purposes of the first sentence of this requirement are considered to be obvious from a reading of the text. Ordinarily no specific flight tests will be required to demonstrate compliance with its terms except for the following: (1) Those conditions normally encountered in the event of the sudden failure of any engine; (2) the establishment of the maximum tolerable cross component of wind velocity during take-off or during approach and landing.

The former of these must be investigated with the airplane in the take-off configuration and at the maximum take-off weight authorized for sea level (see 04.750-T (*a*), (*b*), and 04.752-T), and will require the following specific demonstrations:

a. A demonstration that it is possible to recover to straight flight at the same speed with the wings level after any one engine is rendered suddenly inoperative during steady flight at best angle of climb speed or 120 percent of V_{s_1} (see 04.7511-T) whichever the applicant shall select, while drawing take-off or maximum available power on the operating engine(s) with the landing gear retracted and the airplane loaded at the rearmost center of gravity. The rudder pedal force required to maintain straight flight shall not exceed 180 pounds.

b. A demonstration that it is possible to execute 15° banked turns with or against the inoperative engine starting from steady flight at 140 percent of V_{s_1} with one engine inoperative, the inoperative propeller operating in low pitch, maximum except take-off power on the operating engine(s), landing gear extended, wing flaps in the landing position, and with the airplane loaded at rearmost center of gravity.

c. In order to demonstrate that the rudder will not lock or become heavily over-balanced in the fully deflected position at moderate angles of yaw, the airplane shall also be investigated for the effect of executing sudden changes in heading, as read by means of the directional gyro while holding the wings level, in gradually increasing increments until a maximum of 15° has been reached or dangerous airplane characteristics encountered. This investigation shall be conducted at 140 percent of V_{s_1} with one engine inoperative, the inoperative propeller feathered, maximum except take-off power on the operating engine(s), landing gear retracted, and the wing flaps in the condition used in the determination of the maximum one engine inoperative operating altitude (see 04.7513-T).

The requirements of CAR 61.7122 (*c*) and CAR 61.7123 (*b*) make necessary the establishment of a maximum tolerable cross component of wind velocity beyond which take-off or landing upon a given runway becomes unsafe or impossible. The magnitude of this maximum cross-wind velocity depends greatly upon the controllability of the airplane either in the air or on the ground or both. It will, therefore, be required that the applicant state a maximum tolerable velocity for the particular airplane and demonstrate by means of at least three take-offs and three landings in cross-wind velocities equal to or in excess of that selected that the airplane is safely controllable. The velocity so selected will then become a part of the operating limitations for the airplane.

The second sentence of this requirement concerns itself with those changes in flap position and/or power which are likely to be encountered during an approach for landing when it becomes necessary to go around for another attempt at landing. Its purpose is to make any of these changes possible, assuming the pilot to find it necessary to devote at least one hand to the initiation of the desired operation, without being over-powered by the primary airplane controls. It aims at a design such that no excessive change in trim results from the application or removal of power or the extension or retraction of wing flaps. Compliance with its terms also requires that the relation of control force to speed be such that reasonable changes in speed may be made without encountering very high control forces. Compliance must be demonstrated starting with the landing gear in a position appropriate to the initial flap position; i. e., when flaps are up initially, the landing gear must be up and with flaps initially down, the landing gear must be down.

The third sentence is also concerned with the eventuality of going around during an approach for landing in which event it would be obviously desirable to be able to retract the wing flaps quickly and automatically at such a rate that if power be applied simultaneously with the initiation of flap retraction no altitude would be lost. The design feature involved with this requirement is obviously the rate of flap retraction. Compliance must be demonstrated with the landing gear extended.

04.7541–T Trim. The means used for trimming the airplane shall be such that after being trimmed and without further pressure upon or movement of either the primary control or its corresponding trim control by the pilot or the automatic pilot, the airplane will maintain:

(*a*) Lateral and directional trim under all conditions of operation consistent with the intended use of the airplane, including operation at any speed from best rate of climb speed to high speed and operation in which there is greatest lateral variation in the distribution of the useful load;

(*b*) Longitudinal trim, under the following conditions: (1) During climb at the best rate of climb speed with maximum-except-take-off power; (2) During a glide with power off at a speed not in excess of 1.4 V_{s_1}; and (3) During level flight at any speed from 90 percent of high speed to the sum of V_{s_1} and 20 percent of the difference between high speed and V_{s_1};

(*c*) Rectilinear climbing flight with the critical engine inoperative, each other engine operating at maximum-except-take-off power and the best rate of climb speed under such conditions;

(*d*) Rectilinear flight with any two engines inoperative and each other engine operating at maximum except-take-off power under the following conditions: (1) With the weight of the airplane not more than that at which there is a speed range in level flight of not less than 10 miles per hour; (2) With the speed of the airplane not more than the high speed obtained under the conditions specified in (1) less 10 miles per hour.

It is the purpose of this regulation to require that it be possible to trim the airplane completely for any flight condition which it is reasonable to assume will be maintained steadily for any appreciable time. Flight tests will, of course, also be required to demonstrate compliance for an unsymmetrical power. Yaw will be limited to that with "wings level" or "zero" when yawmeter is installed as a part of the required equipment.

"All conditions of operation consistent with the intended use of the airplane," under which item (*a*) of this requirement requires laterial and directional trim and which were foreseen at the time the regulation was originally written, are contained in items (*b*), (*c*), and (*d*). As a convenience in testing, it will be acceptable to interpret "best rate of climb speed" in items (*a*), (*b*) (1), and (*c*) as $1.4V_{s1}$. There is also no apparent need that lateral and directional trim exist at the high speed when longitudinal trim is required by item (*b*) at 90 percent of this speed only. Therefore, lateral and directional trim at 90 percent of the high speed will also be considered satisfactory.

Although not specifically stated in item (*c*), the demonstration required may be made with the inoperative propeller feathered. Further, the trim speed required in item (*c*) shall be that at which the enroute climb and cooling is demonstrated when the speed selected by the applicant is greater than the best rate of climb speed.

04.7542–T Stability. The airplane shall be longitudinally, directionally, and laterally stable in accordance with the following provisions. Suitable stability and control "feel" may be required in other conditions normally encountered in service if flight tests show such stability to be necessary for safe operation.

Stability is closely related to trim in that if stability is absent, trimming is impossible. It will be noted that in the succeeding requirements, a great deal more attention is devoted to longitudinal stability than to the lateral stability. This is regarded as merely a reflection of the fact that the longitudinal or elevator control is intimately involved in the establishment of center of gravity limits, which is always necessary, while the lateral characteristics ordinarily have negligible effect upon these. It will also be noted that, concerning longitudinal stability, the static stability is defined primarily in terms of the way the longitudinal control feels to the pilot, whereas the dynamic stability is specified in terms of the behavior which the airplane itself shall exhibit when certain specific things are done with the elevator control.

It has been attempted, in the succeeding requirements, to cover those specific flight regimes in which stability is considered essential. It is believed that these are critical in the sense that if the required stability is obtained in these conditions it will probably also be obtained in any other flight condition likely to be encountered with the airplane. If an airplane design be encountered for which this is not true it may be necessary to investigate other flight conditions.

04.75420–T Static longitudinal stability. In the flight conditions described in the following sub-section 04.754200–T:

(*a*) At any speed which can be obtained without excessive control force and which is more than 10 miles per hour above or below the specified trim speed, but not greater than the appropriate maximum permissible speed or less than the minimum speed in steady unstalled flight the characteristics of the elevator control forces and friction shall be such that; (*1*) A pull is required to maintain speeds below the specified trim speed and a push to maintain speeds above the specified trim speed; (*2*) The control will, when unrestrained by the pilot, move continuously toward its original trim position.

(*b*) Where a stable slope of the stick force versus speed curve is specified, any decrease in speed below trim speed shall require an increase in the steady pull on the elevator control and any increase in speed above trim speed shall require an increase in the steady push on the control. Such slope shall be between such limits that any substantial change in speed is clearly preceptible to the pilot through a resulting change in stick force, and that the stick force required to produce necessary changes in speed does not reach excessive values.

The two basic purposes in requiring the static longitudinal stability are defined by this requirement.

The first of these is to require that, once the airplane has been trimmed, it will tend to maintain the trim speed in such manner as to require a conscious effort on the part of the pilot to depart from that speed, using forward pressure on the control column for an increase in speed and the reverse for a decrease. These forces must be such that departures in speed in either direction from the trim speed will require control column pressure that increase approximately proportionately as the speed departs from the trim speed. Static stability is specified in terms of "stick" forces because it is believed necessary and desirable to provide "feel" of the airplane for the pilot through this medium. Thus it may be seen that elimination of friction from the control system is an important factor which must be considered in connection with static stability. The accomplishment of this purpose is intended to provide safeguards against inadvertent stalls or inadvertent elevator control operation at excessive speeds, easy handling qualities during instrument flight, and generally to hold to a minimum the amount of attention and skill required of a pilot during landings, take-offs, and the other normal operating conditions.

The second basic purpose is to make it possible to make such changes in speed as may be required to perform a maneuver without it being necessary to readjust the trim in order to relieve very high control forces which would otherwise be necessary.

The ideal relation of elevator control force to speed would be such that the control force required to maintain any speed should increase proportionally with the departure of speed from the trim speed and the magnitude of the force, at the extremes of possible or desirable departures, would be easily within the ability of the pilot to exert. Such ideal relationship is, however, very difficult to obtain in an actual airplane design, particularly as the size of an airplane increases and the relative amount of power installed increases. Another difficulty is that there is friction in most airplane control systems which tends to blanket out the control forces involved when these are below a certain magnitude.

The requirement contained here is, therefore, a compromise between what is desirable and what may readily be provided. Thus it requires not that the control forces involved be proportional to departures from trim speed throughout the range of attainable airplane speed, but only throughout a limited range of speed either side of the trim speed, and beyond this range that there be no reversal of the control forces. It also requires that the control system friction remain low enough that if the elevator control be released at any speed more than 10 mph away from the trimmed speed the control must move toward the trimmed position and the speed of the airplane must return when equilibrium is established to within 10 mph of the trimmed speed. This, in effect, requires that the control force at any point within the range of speeds which may be attained with the airplane must exceed the force required to overcome the friction in the control system except within the range 10 mph above or below the trim speed.

The greatest difficulty likely to be encountered in attempting to demonstrate compliance with this requirement appears to be that for any condition involving the use of an appreciable amount of power the control force required to make substantial changes in speed tends to become vanishingly small, so small that appreciable changes in speed may occur without the change in the control force being perceptible to the pilot. This seems especially likely to be true if the control surfaces have been so designed that without power the control forces required to produce necessary changes in speed do not become excessive. Figure 55 has been prepared to indicate what is required in the way of static longitudinal stability characteristics.

Although not specifically covered by the regulations, it is also desirable that motion of the control column exhibit substantially the same characteristics as are required of the control forces. That is, it should require forward displacement of the column to produce a speed in excess of the trimmed speed and vice versa; the required displacement should be approximately proportional to the departure from the trimmed speed produced; and there should be no reversal of the sense of the required displacement of the control.

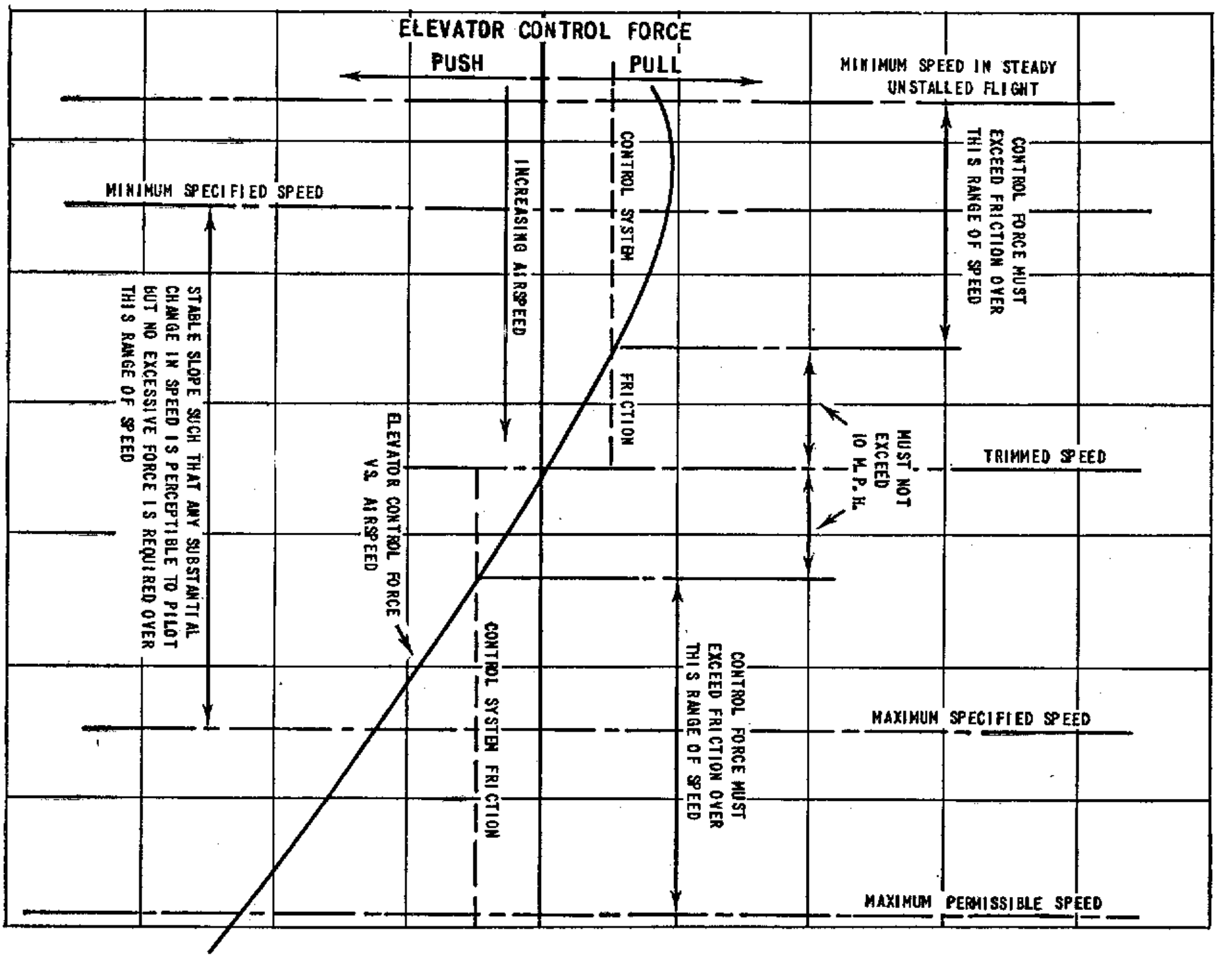


Figure 55.—Static longitudinal stability.

04.754200-T Specific stability conditions. (*a*) *Landing.*—With flaps in the sea level landing position, the landing gear extended, maximum sea level landing weight, the airplane trimmed at 1.4 V_{s_1} and throttles closed, the stick force curve shall have a stable slope at all speeds between 1.1 V_{s_1} and 1.8 V_{s_1}.

(*b*) *Approach.*—With flaps in sea level approach position, landing gear retracted, maximum sea level landing weight, the airplane trimmed at 1.4 V_{s_1} and with power sufficient to maintain level flight at this speed, the stick force curve shall have a stable slope at all speeds between 1.1 V_{s_1} and 1.8 V_{s_1}.

(*c*) *Climb.*—With flaps retracted, landing gear retracted, maximum sea level take-off weight, 75 percent of maximum-except-take-off power and with the airplane trimmed at 1.4 V_{s_1}, the stick force curve shall have a stable slope at all speeds between 1.2 V_{s_1} and 1.6 V_{s_1}.

(*d*) *Cruising*.—With flaps retracted, maximum sea level take-off weight, 75 percent of maximum-except-take-off power, and with the airplane trimmed for level flight, the stick force curve shall have a stable slope at all speeds obtainable with resonable stick forces between: (1) 1.2 V_{s_1} and the maximum permissible speed, when the landing gear is retracted; (2) 1.2 V_{s_1} and the level flight speed, when the landing gear is extended.

The four specific flight conditions covered above are those in which it is believed that static longitudinal stability is important. The trim speed ranges specified are those which are considered essential for the pilot to be able to rely upon for the approximate proportionality of control force to departure from the trim speed. The flight tests required to demonstrate compliance will involve the actual measurement of the control forces. These forces must be low enough to permit attaining the full range of speeds under conditions (*a*), (*b*), and (*c*) only.

04.75421–T Dynamic longitudinal stability. The airplane shall not be dynamically unstable longitudinally, as shown by the damping of the normal long period oscillation, under any flight condition that is likely to be maintained for more than 10 minutes in ordinary service. Compliance with this requirement shall be demonstrated under at least the following conditions: (*a*) During level flight with 75 percent of maximum-except-take-off power; (*b*) During a climb with 75 percent of maximum-except-take-off power at a speed equal to 75 percent of that obtained in item (*a*) above.

Any short period oscillation occurring between stalling speed and maximum permissible speed shall be heavily damped with the primary controls in a fixed position.

During the discussion which preceded the adoption of the above provisions, it seemed generally conceded by most that dynamic longitudinal instability was somewhat more an annoyance than a positive hazard. It was, however, considered to be of sufficient annoyance to warrant eliminating it for any flight condition likely to be maintained for any appreciable time. Since the removal of power and the extension of flaps and/or landing gear ordinarily improves stability, no gliding flight condition is specified. If stability be found during the two flight conditions here specified, it will ordinarily be unnecessary to investigate any other.

The testing required to demonstrate compliance with this requirement must ordinarily be done at both extremes of the range of center of gravity and involves disturbing the airplane by means of the longitudinal control and thereafter observing the oscillation of the airspeed.

It should be noted that the regulation requires that, "The airplane shall not be dynamically unstable." In order to show compliance, therefore, it will be necessary that the test involve not less than three complete cycles, unless the motion be completely damped in fewer, in order to be sure that the amplitude of the oscillation is not increasing.

No specific tests are required for the short period oscillation to which the last sentence of this regulation refers unless it be encountered in the course of the flying which must otherwise be done with the airplane.

04.75422–T Directional and lateral static stability. The static directional stability, as shown by the tendency to recover from a skid with rudder free, shall be positive for all flap positions and symmetrical power conditions, and for all speeds from 1.2 V_{s_1} up to the maximum permissible speed. The static lateral stability, as shown by the tendency to raise the low wing in a sideslip, shall be positive within the same limits.

The purpose of this requirement is to insure that the airplane shall behave normally if disturbed in roll or yaw. Although no real motion of the airplane involving roll is possible without yaw also being involved, and vice versa, it is convenient to investigate the rolling and yawing stability separately and this is done in the testing required to show compliance with this requirement.

In the case of directional stability, the heading of the airplane is altered to approximately 20° (as read on the directional gyro) by means of the rudder, the wings held level by means of the ailerons and the rudder released. The airplane is considered to comply if the resulting skid diminishes. In the case of lateral stability the airplane is rolled to approximately 20° (as read on the gyro horizon) by means of the ailerons, the heading maintained by means of the rudder and elevator and the ailerons released. The airplane is considered to comply if the angle of bank diminishes.

04.7543–T Stalling. With power off, and with that power necessary to maintain level flight with flaps in approach position at a speed of 1.6 V_{s_1} maximum landing weight, flaps and landing gear in any position, and center of gravity in the least favorable position for recovery, it shall be possible to produce and to correct roll and yaw by unreversed use of the aileron and rudder controls up to the time when the airplane pitches in the manuever described below. During the pitching and recovery portions of the maneuver it shall be possible to prevent appreciable rolling or yawing by normal use of the controls.

In demonstrating this quality, the order of events shall be: (*a*) With trim controls adjusted for straight flight at a speed of 1.4 V_{s_1} reduce speed by means of the elevator control until the speed is steady at slightly above stalling speed; then (*b*) Pull elevator control back at a normal rate until a stall is produced as evidenced by an uncontrollable downward pitching motion of the airplane, or until the control reaches the stop. Normal use of the elevator control for recovery may be made after such pitching motion is unmistakably developed.

In any case, the airplane shall not pitch excessively before recovery is completed.

The airplane shall be recoverable without difficulty or the use of power from the inoperative engine when it is stalled with the critical engine inoperative and the remaining engines operating at 75 per cent of maximum-except-take-off power.

The basic purpose of this regulation is to require that, in any stall likely to be encountered inadvertently, the first response of the airplane shall be a downward pitching motion and not a rolling motion; i. e., it is intended to prohibit tip stalling. In demonstrating compliance, the rate of approach to the stall should be not greater than one mile/hour/second. Zero thrust is permissible. An average of five stall tests with one flap setting (or three for each setting when the complete flap angle range is investigated) will be used to determine the stall speed as indicated at the beginning of the pitch or the minimum speed in steadily controlled flight.

The primary purpose of the flight test to demonstrate compliance with the last sentence in this regulation is to insure that the airplane does not become uncontrollable or lose an excessive amount of altitude when so stalled (flaps and landing gear retracted and inoperative propeller in low pitch at $1.4V_{s_1}$).

04.755–T Airplane operating manual. There shall be furnished with each airplane a copy of a manual which shall contain such information regarding the operation of the airplane as the Administrator may require, including, but not limited to, the following: (*a*) All performance data secured under § 04.7513–T to 04.7533–T, inclusive, together with any pertinent descriptions of the conditions, airspeeds, etc., under which such data were determined; (*b*) Adequate instructions for the use and adjustment of the flap controls under § 04.434–T; (*c*) The indicated airspeeds corresponding to those determined in § 04.75320–T, together with pertinent discussion of procedures to be followed if the critical engine becomes inoperative on take-off; (*d*) A discussion of any significant or unusual flying or ground-handling characteristics, knowledge of which would be useful to a pilot not previously having flown the airplane.

The primary purpose of this manual is to provide for the crew who will operate the airplane any information concerning the airplane considered by the CAA essential to or likely to promote safety during such operation. This will ordinarily require a certain amount of descriptive material concerning those parts of the airplane directly operated or otherwise used by the crew, an understanding by them of the nature, location, and functioning of which is therefore essential. The manual should also contain, in order to serve this purpose, a description and chronological outline of the procedure to be followed by the crew during various phases of the operation, both "normal" and "emergency" in which special attention and emphasis should be given to any precaution which should be observed therein in the interest of safety. "Check lists" which list in chronological sequence the operations to be performed by each active crew member during each phase of the operation of the airplane appear likely to be very useful in this connection. Finally, to accomplish this purpose of the manual, it is ordinarily necessary to include all instructions covering loading the airplane necessary to insure that the operating limitations upon weight and *CG* position may be readily observed.

Another important purpose of the manual is to implement the operating requirements of CAR 61.712; i. e., to furnish a source for all the airplane information necessary to establish the limitations specified by those requirements as well as that necessary to enable the crew readily to operate the airplane within the limitations so established. This purpose requires the inclusion in the manual of all operating limitations peculiar to the airplane under any circumstance likely to be encountered during its life as well as information concerning each of the items of performance involved by CAR 61.712 as functions of weight, altitude, wind velocity, flap setting, etc., throughout the range of these variables for which it is desired by the applicant to provide; the point being that the scheduled operation of the airplane by an air carrier will be limited to values of all such variables within the range(s) covered by information available in the manual. This situation requires that the applicant consider the extent to which he wishes to limit the usefulness of the airplane subsequent to its certification as a type.

It may be noted, concerning the material to be included in the manual, that two types are involved. The first of these is the operating limitations which are, in effect, a partial statement of the terms upon which the airworthiness certificate is issued. Compliance with these operating limitations is therefore required by law (see Section 610 (a) of the Civil Aeronautics Act of 1938.) The second type of material is the performance information, recommended operating procedures, general arrangement, and loading instructions, the observance or use of which is not legally required of the operator of the airplane. This second type of material is intended to convey information believed likely to promote or contribute to safety in operation.

It is believed that the usefulness of the manual will bear some inverse relation to its physical bulk and to the extent of its complexity. It is, therefore, strongly recommended that great care be taken to prepare it in the simplest most compact form consistent with the completeness and clarity of presentation of the necessary information. This will probably require careful editing of text and consideration of details of arrangement. It is believed that an 8″ x 10½″ size will probably be found most convenient and this size is recommended. It is also recommended that durability be considered in selecting the materials involved in its reproduction and binding. Finally, it is suggested that consideration also be given to the likelihood of revisions and the manner in and ease with which this may be accomplished.

Upon receipt of the manual as a part of the process of type certification (see "Procedure to be Followed by Applicant" in Introduction) it will be examined by appropriate CAA divisions involved and, when found to comply satisfactorily with the regulation, each page will be embossed with the seal of the Civil Aeronautics Administration and a copy so sealed returned to the applicant. The aircraft specification for the type will list the manual as an item of required equipment and the applicant's sealed copy must be made available upon request to any CAA inspector issuing an original airworthiness certificate under the type certificate in order that he may verify that the manual furnished with that individual airplane conforms with the approved manual.

Following is an outline indicating the scope and arrangement of the manual and the treatment of the necessary material which, it is believed, will best accomplish all of the purposes considered above. Its use is recommended.

TITLE PAGE

This page should form the front cover and should contain the name of the airplane manufacturer, the model designation, and the serial number of the airplane to which the manual applies.

LOG OF REVISIONS

This page should take the form of a table in which to record for each revision an identifying symbol, a date, and the page numbers involved.

I.—OPERATING LIMITATIONS

The purpose of this section is merely to state the limitations without any unnecessary explanation of what they are. The manual should point out that observance of these limitations is required by law.

A. Weight. (Indicate the range of maximum take-off and landing weight approved by means of a table or suitable diagram showing these weights at various altitudes throughout the range for which performance information is contained in the manual. State that airplane weight in excess of maximum landing weight must be disposable fuel. State any other limitations on weight and if appropriate refer to "Loading Information," Section V.)

B. Center of Gravity. (State all authorized *CG* limits. Refer to "Loading Information.")

C. Powerplant. (State all powerplant limitations; i. e., mp, rpm, maximum time for use of take-off power, cylinder head and barrel and oil temperatures, minimum fuel octane No., etc. Any limitation on rpm due to roughness, vibration, tip speed, etc.; propeller pitch, cowl flap position(s), etc.)

D. Speed. (State "never exceed" speed for all flap settings, "level flight or climb" speed, minimum controllability speed with one engine inoperatiive, minimum speed at which airplane may be climbed with one engine inoperative, critical speed during take-off beyond which flight may be continued and below which power should be cut and the airplane stopped, minimum "approach" speed, and any other limiting speed—all in true indicated airspeed.)

E. Crew. (State number and identify members of minimum crew necessary to safe operation.)

F. Flaps. (State maximum flap extension approved for take-off, approach, landing, as functions of weight or altitude if appropriate.)

G. Taxiing. (State any limitations on speed, power, wind direction, velocity during turns, cowl flap postion, etc.)

H. Wave Height. Seaplanes and flying boats only. (State maximum wave height approved for take-off and for landing.)

I. Critical Cross-wind Velocity. (State critical or limiting value beyond which take-off or landing become dangerous.)

II. PERFORMANCE INFORMATION

This section should contain all the performance information necessary to implement the operating requirements of CAR 61.712 and to safely operate the airplane.

A. Engine Power Curve. (A copy of the engine manufacturer's standard chart of bhp vs. mp @ rpm and bhp vs. Altitude @ rpm and @ mp.)

B. Airspeed Calibration. (A plot of TIAS vs. IAS @ various flap positions, preferably on one page.

C. Stalling Speeds. (A table or diagram of true indicated stalling speeds at various weights at all authorized flap settings, power-off.)

D. Climb

1. *Take-off*
2. *Enroute*
3. *Approach*
4. *Landing*

(A diagram of Climb *vs.* Altitude at Weight plus the Climbing Air Speed *vs* Altitude for each airplane configuration in standard air throughout the range of weight and altitude for which the airplane is certificated.)

E. Take-off

1. *Accelerate-stop distance.* (That is, the distance required to accelerate to the minimum speed for control with one engine inoperative and stop. A diagram showing Distance *vs.* Altitude at Weight throughout the range of altitude h and weight W for which the airplane is certificated for take-off.)

2. *Flight path.* (That is, the take-off flight path specified by 04.75321–T (*a*), (*b*), (*d*), (*e*), (*f*), and (*g*), as a function of weight and altitude throughout the range of each for which the airplane is certificated for take-off. It is suggested that this can probably be done most conveniently by preparing a diagram to scale on rectangular graph paper showing for each of several altitudes the flight path for each of several weights from the minimum to the maximum for the particular altitude.)

F. Landing. (That is, the distance required by 04.7533–T. A diagram showing Distance *vs.* Altitude at Weight and Wind Velocity throughout the range of altitude and weight for which the airplane is certificated for landing.) (See figure 9 of Flight Engineering Report No. 13.)

G. Minimum Take-off Runway Length. (Present by means of a diagram such, for example, as figure 5 of Flight Engineering Report No. 9 or a suitable table, the minimum take-off runway length permitted by CAR 61.7122 in the case where no obstacle exists at the end of the runway and there is no wind at various altitudes throughout the range for which performance information is provided.)

H. Minimum Effective Landing Runway Length. (Present, by means of a diagram such, for example, as figure 6 of Flight Engineering Report No. 9 or a suitable table, the minimum effective landing runway length, as defined by CAR 61.7124, permitted by CAR 61.7123 in the case of aero wind at various altitudes throughout the range for which performance information is provided.)

I. Altitude of Terrain. (Present, by means of a diagram such, for example, as figure 4 of Flight Engineering Report No. 9 or a suitable table, the altitude which may be cleared in accordance with the requirements of CAR 61.7125, at various airplane weights throughout the range of those likely to be encountered in actual operation.)

III. RECOMMENDED OPERATING PROCEDURES

This section of the manual should contain information, peculiar to the airplane, concerning normal and emergency procedures, knowledge of which might enhance the safety of operation of the airplane. The manual should state that these procedures are not mandatory.

A. Start and Warm Up Engines
B. Taxi
C. Take-off
D. Approach
E. Land

Outline normal procedure for each, noting any special precautions in the interest of safety. Include check list for each crew member for each operation. Describe or refer to procedure in any emergency likely to occur in each.

F. One Engine Inoperative. (Outline procedure to be used in event of engine failure during (*a*) take-off and (*b*) cruising flight including recommended speeds, trims, operation of remaining engine(s), propeller feathering, etc.)

G. Propeller Feathering. (Outline any necessary or desirable procedure to be followed.)

H. Fuel Dumping. (Outline procedure including speeds, power, etc.)

I. Control System Locks

J. Flap Control

K. Auxiliary Power Plant. (Describe procedure for starting and operating.)

L. Fire Extinguishing Equipment. (Outline procedure to be used in event of fire in various parts of airplane.)

M. De-icers. (Wing, tail, propeller, and carburetor.)

N. Windshield Cleaners and De-icers.

O. Landing Gear
P. Automatic Pilot
Q. Emergencies. (Outline any methods by means of which wing flaps, landing gears, etc., may be operated in event of failure of primary systems.)

Note: Add any other item(s) necessary.

IV. DESIGN FEATURES

This section should contain all descriptive material covering those parts of the airplane with which operating personnel should be especially familiar for safe operation or in the event of emergency.

A. Fuel System
B. Oil System
C. Carburetor Heating or Fluid De-icing System
D. Electrical System

(A–D: A brief description of the arrangement and operation of each.)

E. Hydraulic System
F. Control System
G. Auxiliary Power Plant
H. Emergency Exits. (A description, diagram showing location, and instruction for use.)
I. Hull Compartmentation. (Flying Boats Only.)
J. Emergency Equipment. (Description, location, instructions for use.)

V. LOADING INFORMATION

This section should include the information necessary to load and keep the airplane within the operating limitations governing weight and *CG* location.

A. Cabin Arrangement. (A diagram, approximately to scale, identifying and showing the location of each item of disposable load.)
B. Loading Schedule. (If required.)
C. Instructions. (Adequate instructions for the use of the loading schedule or other equivalent means.)

04.9 MISCELLANEOUS REQUIREMENTS

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

Gasoline	6 lbs. per gallon.
Lubricating Oil	7.5 lbs. per gallon.
Crew and Passengers	170 lbs. per person, unless otherwise specified by the Administrator.
Parachutes	20 lbs. each.

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

"Adequate means" is interpreted as being independent of tire inflation pressure and shock absorber travel. This normally requires that the means be attached to, or a part of, the fuselage or hull structure. They must also be readily identifiable and usable.

APPENDIX I

AN INTERPRETATION OF 04.003 FOR LARGE AIRPLANES

A. General

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

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

B. Structural Loading Conditions

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

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

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

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

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

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

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

C. Proof of Structure

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

Wing analysis.—In preparing the program mentioned above, the following points should be considered:

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

b. Determination of allowable compressive loads in wing covering.

c. Allowable shear loads in webs.

d. Combined loadings.

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

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

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

D. Detail Design

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

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

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

APPENDIX II

SAMPLE WEIGHT AND BALANCE REPORT

NAME OF MANUFACTURER

REPORT NO. ----------

WEIGHTS AND BALANCE OF MODEL ----------

SERIAL NO ----------

IDENTIFICATION MARK ----------

Date ----------------------

Prepared By ----------------------------------

Checked By ----------------------------------

Witnessed By ----------------------------------
(Signature of Civil Aeronautics Administration Representative

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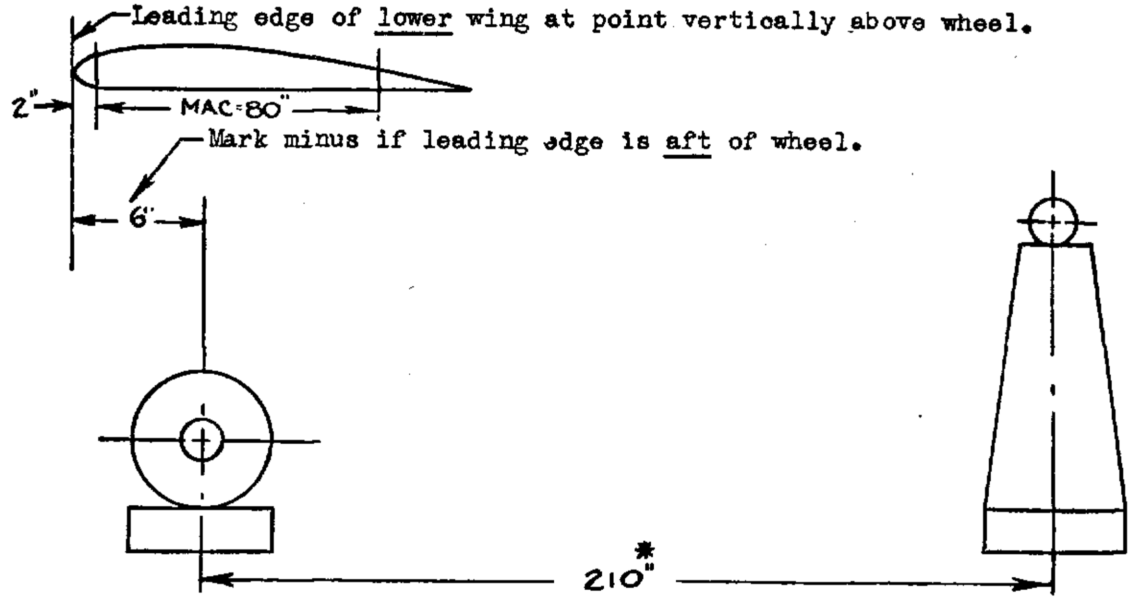
SECTION 1. Aircraft Empty Weight

(A) Empty weight as weighed (in level landing position [1])



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

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

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

CG Empty (as weighed) is aft of wheel centerlines $\frac{250 \times 210}{2250} =$ 23. 3″

CG Empty (as weighed) is aft of lower wing leading edge 23.3+6 = 29. 3″
Lower wing leading edge is aft of datum 100. 0″

CG Empty (as weighed) is aft of datum 100+29.3 = 129. 3″
Datum to MAC leading edge = 102″ (see page 2 of report 981)

[1] Level by means provided in accordance with 04.91.

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SECTION 1—*Continued*

(B) Empty weight as weighed includes the following:

1. *Required equipment*[2]

Item No.[3]	*Name*	*Weight*
10	Starter	21
11	Battery	40
12	Heater	2
13	Ventilator	4
14	Generator	20
15	Position lights	--
16	8.50-10 wheels (mfr. and model) and 8.50-10 6-ply tires	--
17	10½ inches streamline tail wheel	--

2. *Optional equipment*

Item No.	*Name*	*Weight*[4] *(net increase)*	*Horizontal arm from datum*	*Horizontal moment*
7	Wheel streamlines	24	71	1704
19	Flares (type)	17	175	2975
4	Adj. metal prop. 70 lbs.	24	13	312
6	Optional instruments not required (list)	15	60	900
20	Optional fuel capacity 70 gals. (2 tanks at 35 gals.)	15	90	1350
21	Radio:			
	Receiver (type) and antenna	30	60	1800
	Shielding (type)	10	16	160
	Bonding	10	50	500
5	Ballast container and straps, etc	20	138	2760
	Total optional	165	-------	12461
3.	*Empty weight as weighed*	2250	129. 3	290925
	Optional Equipment	−165	-------	−12461
	Basic empty weight	2085	X_E	278464

$X_E = \frac{278464}{2085} =$ Distance from datum to CG of airplane empty with all required items only.

[2] "Required Equipment" (see page 5). List all such items even though weights are not included for some.
[3] Item numbers to correspond with numbers used in balance diagram.
[4] All weights of equipment are *installation* weights.

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SECTION 2. MOST FORWARD *CG* CONDITION.

(A) *Loading as actually flown:*

Item No.	*Name*	*Weight*	*Horizontal arm*	*Horizontal moment*
--	Empty weight as weighed	2250	129.3	290925
1	Oil 5 gals	38	51	1938
2	Fuel 20 gals	120	90	10800
3	Pilot + parachute	[5] 225	90	20250
4	Propeller (if other than noted in section 1 (B)).			
5a	Ballast (incl. containers, straps, etc.)	100	60	6000
	Total	2733	120.8	329913
	Datum to MAC leading edge	102	102	
	Percent of MAC	18.8	÷ 80(MAC)	=23.5%
	Inches aft of leading edge of wing	120.8	−100	=20.8 in.

(B) *Loading substantiated by 2 (A):*

Item No.	*Name*	*Weight*	*Horizontal arm*	*Horizontal moment*
	Basic empty weight	2085	X_s	278464
1	Oil 5 gals	38	51	1938
2	Fuel 70 gals.[6]	420	90	37800
3	Pilot [7]	170	90	15300
3	Passengers (in front seat)	170	90	15300
3	Parachutes in front seats (2 at 20 lbs.)	40	90	3600
4	Propeller (heaviest to be used)(70–46)	24	13	312
6	Optional instruments	15	60	900
7	Wheel streamlines	24	71	1704
21	Radio equipment forward of most forward *CG* limit.			
	Plus other items of optional equipment critical for most forward *CG* load condition.			
	Total	W_F	[8] X_F	M_F

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

[6] *Fuel substantiated* shall be as follows: (See 04.7211) (*a*) 1 gal. for every 12 maximum except take-off horsepower when minimum fuel is critical. (*b*) Full tanks when maximum fuel is critical.

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

[8] Shall not exceed limits in 2 (A) and 3 (A).

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SECTION 3. Most Rearward *CG* Condition

(A) *Loading as actually flown:*

Item No.	*Name*	*Weight*	*Horizontal arm*	*Horizontal moment*
	Empty weight as weighed	2250	129.3	290925
1	Oil 5 gals	38	51	1938
2	Fuel 20 gals	120	90	10800
3	Pilot and Parachute	[5] 225	90	20250
4	Propeller (if other than noted in Section 1 (B)).			
5b	Ballast (incl. containers, straps, etc.)	200	250	5000
	Total	2833	132	373913
	Datum to MAC leading edge		102	
	Percent of MAC		30÷80(MAC)=	37.5%
	Inches aft of leading edge lower wing		132−100=	32.0 in.

(B) *Loading substantiated by 3 (A):*

Item No.	*Name*	*Weight*	*Horizontal arm*	*Horizontal moment*
	Basic empty weight	2085	X_s	278464
1	Oil 5 gals	38	51	1938
2	Fuel $\frac{240}{12}$=20 gals.[6]	120	90	10800
3	Pilot [7]	170	90	15300
4	Propeller (lightest to be used) included in basic empty weigh	no net increase		
6	Flares (Type)	17	175	2975
7	Radio equipment aft of most rearward *CG* (Passengers in rear seat are at arm of 125. If aft of 132 the rear passengers and parachutes should be included here) Plus other items of optional equipment critical for most rearward *CG* condition.			
	Total	W_z	[8] X_z	M_z

See footnotes 5, 6, 7, and 8 on page 178.

SECTION 4. Full Load Condition

(A) *Loading as actually flown:*
(Same form as 2 (A) and 3 (A))

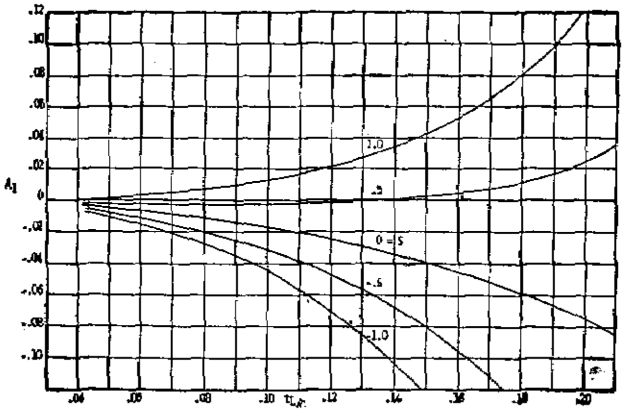

Figure 56.

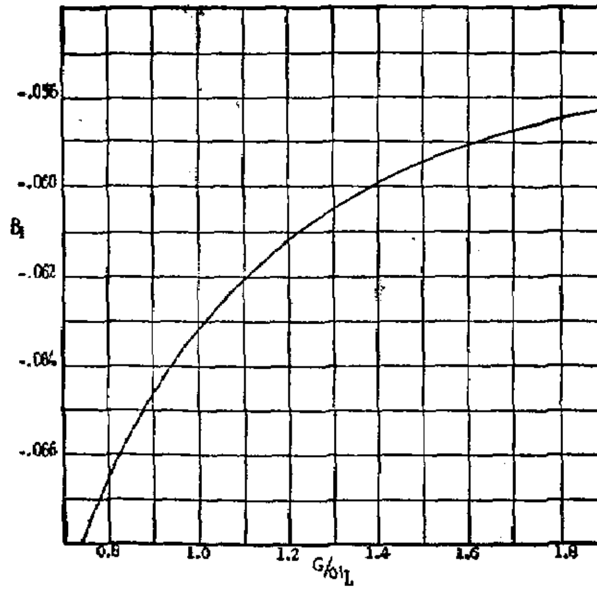

Figure 57.

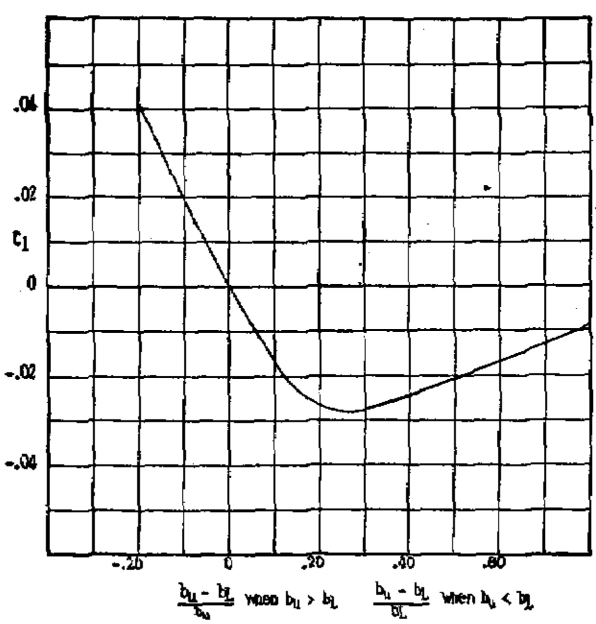

Figure 58.

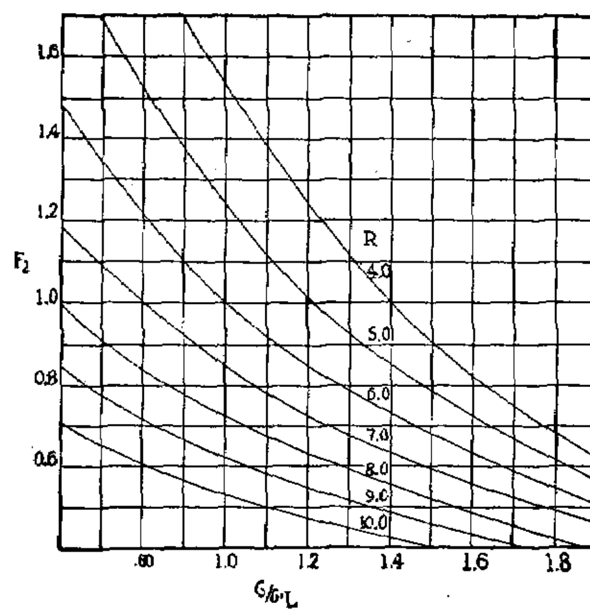

Figure 59.

APPENDIX III

BIPLANE WING LIFT COEFFICIENTS

Reprinted from Air Commerce Bulletin, November 15, 1934

Two NACA Technical Reports [1] embody a complete exposition of the latest available information as to the effects on the individual wing lift coefficients of stagger, wing thickness, gap, decalage, overhang, unequal chords, and unequal effective areas. The purpose of this paper is to present in summarized form a simplified practical solution for C_{L_u} and C_{L_L} based on the data of these reports, except for certain practical compromises and the elimination of an inconsistency as noted later.

First are listed the known cellule and wing characteristics, followed by computations and references to the figures of this paper in the order corresponding to the quickest solution. A sample computation parallels the general presentation.

Given:

$b_u=40$ ft. Overall span of upper wing.
$b_L=20$ ft. Overall span of lower wing.
$b'_u=40$ ft. Net span of upper wing (over-all less fuselage cut-out).
$b'_L=17.4$ ft. Net span of lower wing (over-all less fuselage cut-out).
$S_u=300$ sq. ft. Gross area of upper wing.[2]
$S_L=76$ sq. ft. Gross area of lower wing.[2]
$S'_u=300$ sq. ft. Net area of upper wing (gross less cut-outs).
$S'_L=64$ sq. ft. Net area of lower wing (gross less cut-outs).
$c'_u=7.5$ ft. $=S'_u/b'_u$. Mean geometric chord of upper wing.
$c'_L=44$ in. $=S'_L/b'_L$. Mean geometric chord of lower wing.
$G=66$ in. *Distance normal to zero lift direction.*[3]
Stagger $=44$ in. Distance parallel to zero lift direction.[3]
$t_L=6.6$ in. Maximum thickness of c'_L.
$\delta=3$. Decalage in degrees.

Solution:

$t_L/G=.10=6.6/66$
$s=1.0=\text{stagger}/c'_L=44/44$
$A_1=.012$ from figure 56, function of t_L/G and s.
$\frac{b_u-b_L}{b_u}=.50=\frac{40-20}{40}$
$F_1=.50=1-\frac{b_u-b_L}{b_u}$
$G/c'_L=1.5=66/44$
$B_1=-.0596$ from figure 57, function of G/c'_L.
$C_1=-.015$ from figure 58, function of $\frac{b_u-b_L}{b_u}$
$D=c'_L/c'_u=\frac{S'_L}{S'_u}\times\frac{b'_u}{b'_L}=\frac{64}{300}\times\frac{40}{17}=.50$
$K_1=[F_1(A_1+B_1\delta)+C_1]D=[.50(.012+3\times-.0596)+(-.015)].50=-.049$
$A_2=.050+0.17s=.050+0.17\times1=.22$
$R=\frac{1}{2}\left[\frac{b_u^2}{S_u}+\frac{b_L^2}{S_L}\right]=\frac{1}{2}\left[\frac{40^2}{300}+\frac{20^3}{76}\right]=5.3$
$F_2=.76$ from figure 59, function of R and G/c'_L.
$B_2=.0186$
$(A_2F_2+B_2\delta)=.22\times.76+.0186\times3=.223$
$C_2=-.013$ from figure 60, function of $\frac{b_u-b_L}{b_u}$ and $(A_2F_2+B_2\delta)$
$K_2=[(A_2F_2+B_2\delta+C_2)]D=[(.233-.013)].50=.105$
$E=4.68=S'_{u/S'_L}\pm300/64$
$C_{L_u}=(1+K_2)C_L+K_1=(1+.105)C_L-.049$
$C_{L_u}=1.105C_L-.049$
$C_{L_L}=(1-K_2E)C_L-K_1E=(1-.105\times4.68)C_L-(-.049\times4.68)$
$C_{L_L}=.508C_L+.23$

When $C_L=0, C_{L_u}=-.049 C_{L_L}=.23$
When $C_L=1.0, C_{L_u}=1.056 C_{L_L}=.738$

Plot straight lines through these values in figure 5.

REMARKS: (1) It should be noted that the methods in T. R. 501 of correcting for overhang in figures 59 and 61 are incorrect in that K_{10}, K_{11} and K_{12}, as well as $F_2\times K_{20}$ and K_{21}, should correspond to c_{L/c_u} equals unity, i. e., equal chords. The correction for unequal *chords should have been introduced* later by *multiplication* of the values of K_1 and K_2 for equal chords by the ratio of geometric chords of the lower to upper wing.

(2) Gross areas are used only for the determination of the average aspect ratio.

(3) For the case of deflected flaps an equivalent decalage should be introduced.

(4) In a correct solution the derived straight lines for C_{L_u} and C_{L_L} will intersect at the corresponding value of C_L of the cellule.

(5) The use of the mean aerodynamic centers makes this method of solution applicable also to those cases where the wings incorporate sweep back and/or taper in plan form.

(6) Wings incorporating twist are a special problem not directly amenable to the procedure of this paper.

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[1] Relative Loading on Biplane Wings, by Walter S. Diehl, NACA T. R. 458. Relative Loading on Biplane Wings of Unequal Chords, by Walter S. Diehl, NACA T. R. 501.
[2] Assuming wings *continuous from tip to tip.*
[3] Between mean aerodynamic centers of upper and lower wings as shown in figure 61. (See also 04.217-E.)

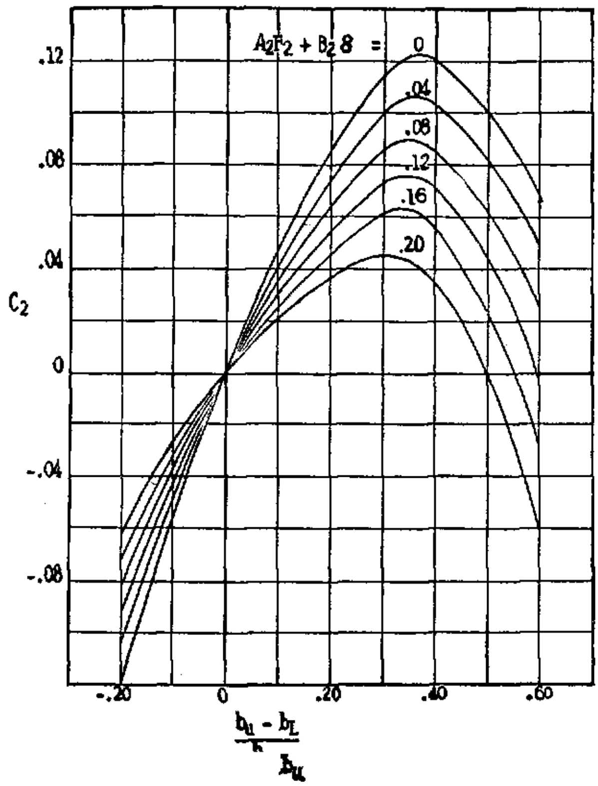
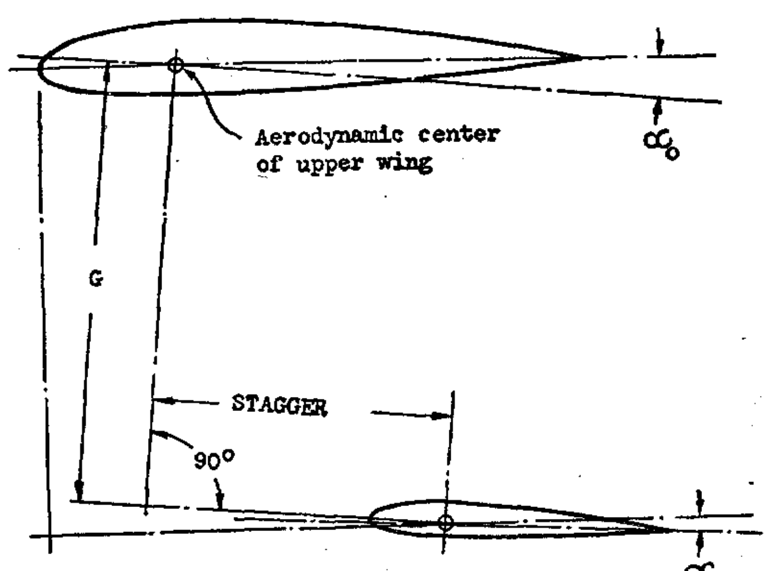


Figure 60.



α_0 is weighted average angle of zero lift

Figure 61.

APPENDIX IV

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

1. Summary

This appendix presents and describes a simple and rapid approximate method for the determination of the spanwise distribution of section lift coefficient c_l[1] on wings for use when a rational method is required. One completely worked example and three additional examples which compare the results obtained by this approximate method with available theoretical methods are included. Limitations of the method are given in section 6. The method described here incorporates a tabular form for use in making the necessary computations. In practice, it is necessary to enter only 7 basic columns in the table, and the remainder of the work is a simple routine procedure which can be carried out by personnel with no engineering knowledge of the principles involved.

2. Basic Considerations

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

$$c_l = C_L c_{la_1} + c_{lb}$$

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

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

$$c_{lb} = \frac{a_o}{2}(\alpha Ro + \beta)$$

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

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

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

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

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

[1] See nomenclature, p 187.

TABLE XXII.—Spanwise Air-Load Distribution

(The following tabulation is for wing shown in Fig. 63)

	1	2	3	4	5	6	7	8	9	10	11	12	13	14	Final $C_{l_{a1}}$ 15	16	17	18	19	20	21	22	Computed c_{lb} 23	Computed $c_{lb}c$ 24	Faired $c_{lb}c$ 25	Final c_{lb} 26
	y/b_2	c—ins.	Δ—ins.	Multiplier	②×③×④	a_o C_L per degree	[1] ⑥×④	⑥/$\bar{a}_o$ [2]	$\frac{4\bar{c}}{\pi}$/②	①²	1−⑩	√⑪	⑨×⑫	⑧+⑬	⑭/2	[3] i	α_{I_o}	⑯−⑰	[4] ⑤×⑱	[5] ⑦×⑱	⑥/2	⑱+α_{Ro}	[6] ㉑×㉒	②×㉓	[7] $c_{l_b}c$	㉕/②
Section from ℄ of airplane to end of flap	.000	102.00	23.04	.333	783.36	.100	------	1.000	.996	000	1.000	1.000	.996	1.997	.998	000	−8.00	8.00	6,266.88	------	.050	3.70	.185	18.87	18.87	.185
	.0960	98.70	23.04	1.333	3,032.06	.100	----	1.000	1.030	.009	.991	.995	1.025	2.025	1.012	000	−8.00	8.00	24,256.48	------	.050	3.70	.185	18.26	18.00	.182
	.1920	94.69	23.04	.667	1,454.44	.100	------	1.000	1.074	.037	.963	.981	1.054	2.054	1.027	000	−8.00	8.00	11,635.52	------	.050	3.70	.185	17.31	14.70	.155
	.2880	91.02	23.04	1.333	2,796.13	.100	----	1.000	1.117	.083	.917	.957	1.069	2.069	1.034	000	−8.00	8.00	22,369.04	------	.050	3.70	.185	16.83	9.70	.107
	.3840	87.80	23.04	.333	674.31	.100	------	1.000	1.158	.147	.853	.924	1.070	2.070	1.035	000	−8.00	8.00	5,394.48	------	.050	3.70	.185	16.24	1.25	.014
Section from end of flap to beginning of tip	.3840	87.80	21.96	.333	642.70	.100	------	1.000	1.158	.147	.853	.924	1.070	2.070	1.035	000	−1.20	1.20	171.24	------	.050	−3.10	−.155	−13.60	1.25	.014
	.4755	83.95	21.96	1.333	2,458.06	.100	------	1.000	1.211	.226	.774	.880	1.066	2.066	1.033	000	−1.20	1.20	2,949.67	------	.050	−3.10	−.155	−13.01	−5.15	.068
	.5670	80.40	21.96	.667	1,177.06	.100	------	1.000	1.265	.321	.679	.824	1.042	2.042	1.021	000	−1.20	1.20	1,412.47	------	.050	−3.10	−.155	−12.46	−9.25	−.115
	.6585	76.87	21.96	1.333	2,250.75	.100	------	1.000	1.323	.434	.566	.752	.995	1.995	.997	000	−1.20	1.20	2,700.90	------	.050	−3.10	−.155	−11.91	−11.40	−.148
	.7500	73.38	21.96	.333	537.14	.100	------	1.000	1.386	.563	.437	.661	.916	1.916	.958	000	−1.20	1.20	644.57	------	.050	−3.10	−.155	−11.37	−11.37	−.155
Section from beginning of tip to extreme tip of wing	.7500	73.38	15.00	.333	366.90	.100	----	1.000	1.386	.563	.437	.661	.916	1.916	.958	000	−1.20	1.20	440.28	------	.050	−3.10	−.155	−11.37	−11.37	−.155
	.8125	69.18	15.00	1.333	1,383.60	.100	------	1.000	1.470	.660	.340	.583	.857	1.857	.928	000	−1.20	1.20	1,660.32	------	.050	−3.10	−.155	−10.72	−10.72	−.155
	.8750	62.27	15.00	.667	622.70	.100	------	1.000	1.633	.766	.234	.484	.790	1.790	.895	000	−1.20	1.20	747.24	------	.050	−3.10	−.155	−9.65	−9.65	−.155
	.9375	49.51	15.00	1.333	990.20	.100	------	1.000	2.054	.879	.121	.348	.715	1.715	.857	000	−1.20	1.20	1,188.24	------	.050	−3.10	−.155	−7.67	−7.67	−.155
	1.000	000	15.00	.333	000	.100	------	1.000	------	1.000	000	000	------	------	------	000	−1.20	1.20	000	------	.050	−3.10	−.155	000	000	000

Σ⑤=19,169.41   Σ⑦=   Σ⑲=82,437.33   Σ⑳=

CALCULATIONS FOR CONSTANTS

$b/2$= 240 ins.

S (total wing area) = 2Σ⑤ = 2×19,169.41 38,398.82 sq. ins.

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

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

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

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

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

1 Column 7 is calculated only when (a=variable).
2 When (a=constant) the values in column 8 are 1.
3 See sketch page 185. The signs of the angles should be carefully observed.
4 $\alpha\ Ro$ as of column 19 is calculated only when (a=constant).
5 $\alpha\ Ro$ as of column 20 is calculated only when (a=variable).
6 Values taken from column 23 are final values of c_{lb} when no high-lift device is operating.
7 Values of $c_{lb}c$ for column 25 are taken from the faired curve of Column 24.

3. Use of Tabular Form

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

Column 1.—Before entering the values of $\frac{y}{b/2}$ in this column, it is necessary to divide the semi-span into a convenient number of sections, and then divide these sections into a convenient number of even parts. Examples of this are shown on figure 62. It is necessary to locate section divisions at the beginning of the tip region (see Fig. 62 (1)), at the ends of high lift devices (see Fig. 62 (2)), and at points where there is an abrupt change in plan form (see Fig. 62 (3)). These section divisions are shown as heavy lines on figure 62. The sections thus obtained are then divided into an *even* number of parts as indicated in figure 62. (An *even* number of parts is necessary in order to insure accuracy in the numerical integration which is automatically provided for in Table XXII.) The values of $\frac{y}{b/2}$ may now be entered in column 1, taking care to enter the $\frac{y}{b/2}$ values at the main section division *twice*, as shown.

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

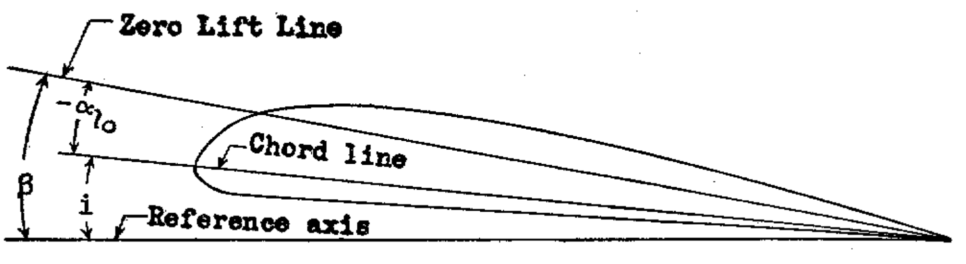
Column 3.—Enter here the actual width in inches of the small divisions of the semi-span within the section (see Table XXII).

Column 4.—Enter here a multiplier which, within a section, is a series of the following form (Simpson's rule for approximate integration). It will be noted that the first and last terms of this series are .333, and the intermediate terms are a repetition of 1.333 and .667. Examples of multiplier values follow:

Two divisions: .333, 1.333, .333
Four divisions: .333, 1.333, .667, 1.333, .333
Six divisions: .333, 1.333, .667, 1.333, .667, 1.333, .333
Eight divisions: .333, 1.333, .667, 1.333, .667, 1.333, .667, 1.333, .333
See also Table XXII for an example of this procedure.

Column 6.—Enter here the slope of the lift curve, a_o, for infinite aspect ratio in C_L per degree for the pertinent airfoil section (or airfoil-flap combination). Data for this purpose can be obtained from standard NACA reports.

Column 16.—Enter here the angle of incidence, i. This is the angle between the chord line (the line used as datum for airfoil ordinates and angles) and the reference axis. The reference axis can be chosen as any convenient axis in the plane of symmetry, such as, the fuselage axis or the chord line of the root chord. Care should be taken in using the proper sign for i, positive values being as measured in the sketch below. (The sign of α_{l_o} is shown as negative in the sketch to agree with NACA airfoil data where the reference line for angles of attack is always the chord line. Therefore, considering only the geometry of the particular sketch, β is obviously $i+\alpha_{l_o}$ or using the sign and expression for β given on the sketch, $\beta=i-(-\alpha_{l_o})=i+\alpha_{l_o}$).



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

Column 17.—Enter here the angle of attack for zero lift, α_{l_o} for the pertinent airfoil section (or airfoil-flap combination), *taking care to use the proper sign.* Data for this purpose can be obtained from standard NACA reports. Computations can now proceed in accordance with the instructions on Table XXII. In cases where high lift devices are not used, the final values of $c_{l_{a_1}}$ and c_{lb} for design purposes are given in columns 15 and 23, respectively.

When such devices are used, the c_{la_1} values of column 15 still apply, but it is necessary to fill out columns 24, 25, and 26 in order to obtain the final c_{lb} (column 26) values for design purposes. Instructions for filling out these columns are given in the following section.

4. Procedure for Obtaining c_{lb} When High Lift Devices are Used

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

This adjustment process is performed by computing column 24 to obtain $c_{lb}c$ and plotting the values thus obtained against the semispan. Examples of this are shown by the dashed lines on figures 63 and 65. These curves are then faired as shown by the solid lines taking particular care to fair in such a manner that the *total area under the faired curve is equal to zero.* The values of $c_{lb}c$ from the faired curve are then entered in column 25 and the final c_{lb} for design purposes is obtained in column 26.

5. Comparison Examples

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

Example 1.—The wing planform of this example is shown in figure 63. The wing has no aerodynamic twist, except that induced by the flap, which is deflected 30°. This example is taken from NACA Technical Report 585, page 3 (reference (2)). Table XXII shows the computation of c_{la_1} and final c_{lb}. Fairing of $c_{lb}c$ is shown on figure 63. Table XXIII gives the computation of c_l for a wing lift coefficient $C_L=1.72$, and a comparison of final values of c_l, c_{la} and c_{lb} with those obtained theoretically by reference 2 is shown in figure 64. It will be noted that the agreement of the c_l values is very satisfactory for design purposes.

TABLE XXIII.—Final c_l Distribution for a Wing C_L of 1.72

(Example from NACA, T. R. 585)

		Final c_{la}	Final c_{lb}	Final c_l
1	2	3	4	5
$\frac{y}{b/2}$	c_{la}	$C_L \times 2$	c_{lb}	3+4
0	0.998	1.72	0.185	1.905
0.0960	1.012	1.74	.182	1.922
.1920	1.027	1.77	.155	1.925
.2880	1.034	1.78	.107	1.887
.3840	1.035	1.78	.014	1.794
.4755	1.033	1.78	−.068	1.712
.5670	1.021	1.76	−.115	1.645
.6585	.997	1.71	−.148	1.562
.7500	.958	1.65	−.155	1.495
.8125	.928	1.60	−.155	1.445
.8750	.895	1.54	−.155	1.385
.9375	.857	1.47	−.155	1.315
1.000	000	000	000	000

Example 2.—The wing planform for this example and the comparison curves of c_l are shown on figure 66. This wing has no aerodynamic twist. It will be noted that the agreement of the approximate method with the theoretical results is satisfactory for design purposes. (The C_L value of 4.52 for this example is a theoretical value corresponding to an angle of attack beyond the stall. However, the c_l values for this wing at angles of attack below the stall would be directly proportioned to those shown on figure 66.)

Example 3.—The wing planform and comparison curves for this example are shown on figure 67 This wing has a straight center-section, a root to tip chord ratio of 4, and an aerodynamic washout of 2°. The lift coefficient of the wing, C_L, is 0.687. In this case, a comparison is made between the $\frac{c_l c}{b/2}$ values given by the approximate method and the theoretical curve from NACA Technical Note 732 (reference (3)). (It will be noted that the value $\frac{c_l c}{b/2}$ is directly proportional to the load per foot of span acting on the wing.)

Example 4.—This comparison example is shown on figure 68. The flap deflection is 60°, the ailerons are in neutral position, and the wing C_L is 1.716.

The theoretical curves for c_l, c_{la}, and c_{lb} are the same as those shown in figure 7–4, page 7–44, of ANC 1 (1). It will be noted that the agreement of the approximate method with the ANC 1 (1) theoretical method is entirely satisfactory for structural design purposes. Figure 68 also shows a comparison of the c_{lc} values of the approximate and theoretical methods. (The term c_{lc} is directly proportional to the load per foot of span acting on the wing.)

6. Conclusions

On the basis of these comparison examples and many other comparison examples which have been completed, it is concluded that the approximate method given here of computing spanwise distribution of c_l is satisfactory for structural design purposes, provided that: (1) The aspect ratio is within the normal range of values (say from 5 to 12), and, (2) The wing has reasonably rounded tips, if the taper ratio is greater than 0.5. (This restriction as to rounded tips does not apply for taper ratios less than .5)

In cases where the wing does not have reasonably rounded tips and, at the same time, the taper ratio is greater than 0.5, the approximate method can also be used provided that an empirical tip correction such as is outlined in paragraph 1.32 of ANC–1 (1), "Spanwise Air-Load Distribution" is employed. This method is considered satisfactory for any amount of aerodynamic twist that may be encountered in conventional design practice.

7. References

(1) Schrenk, O.: *A Simple Approximation Method for Obtaining the Spanwise Lift Distribution.* T. M. 948, NACA, 1940.

(2) Pearson, H. A.: *Span Load Distribution for Tapered Wings with Partial-Span Flaps.* T. R. 585, NACA, 1937.

(3) Sherman, Albert: *A Simple Method of Obtaining Span Load Distribution.* T. N. 732, NACA, 1939.

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

8. Nomenclature

- S Wing area, square inches
- b Span, inches
- c Chord, inches
- $\bar{c}$ Average chord, inches ($= S/b$)
- y Distance of a particular station from centerline of wing, inches
- C_L Wing lift coefficient
- c_{la_1} Additional lift coefficient for a section when wing $C_L = 1.0$
- c_{la} Additional lift coefficient for a section ($= C_L c_{la_1}$)
- c_{lb} Basic lift coefficient for a section due to aerodynamic twist, when wing is operating at zero lift
- c_l Section lift coefficient ($= c_{la} + c_{lb}$)
- α_{l_0} Angle of attack of a section for zero lift, degrees
- i Angle between the chord line and the reference axis, degrees (see sketch on page 185)
- β Angle between the zero lift line and the reference axis, degrees (see sketch on page 185; note that $\beta = i - \alpha_{l_0}$)
- α_R Angle between reference axis and the wind direction, degrees (positive when the reference axis is so inclined to the wind direction as to produce positive lift, assuming (for this purpose only) that the reference axis acts as a zero lift chord line on airfoil section)
- α_{R_0} Angle between the wind direction and the reference axis when the wing is operating at zero lift, degrees
- α_{a_s} Angle between the zero lift line of a section and the wind direction, degrees ($\alpha_{a_s} = \alpha_R + \beta$)
- a Lift curve slope, C_L/degree
- a_o Section lift curve slope, c_l/degree (slope of graph of c_l vs. α)

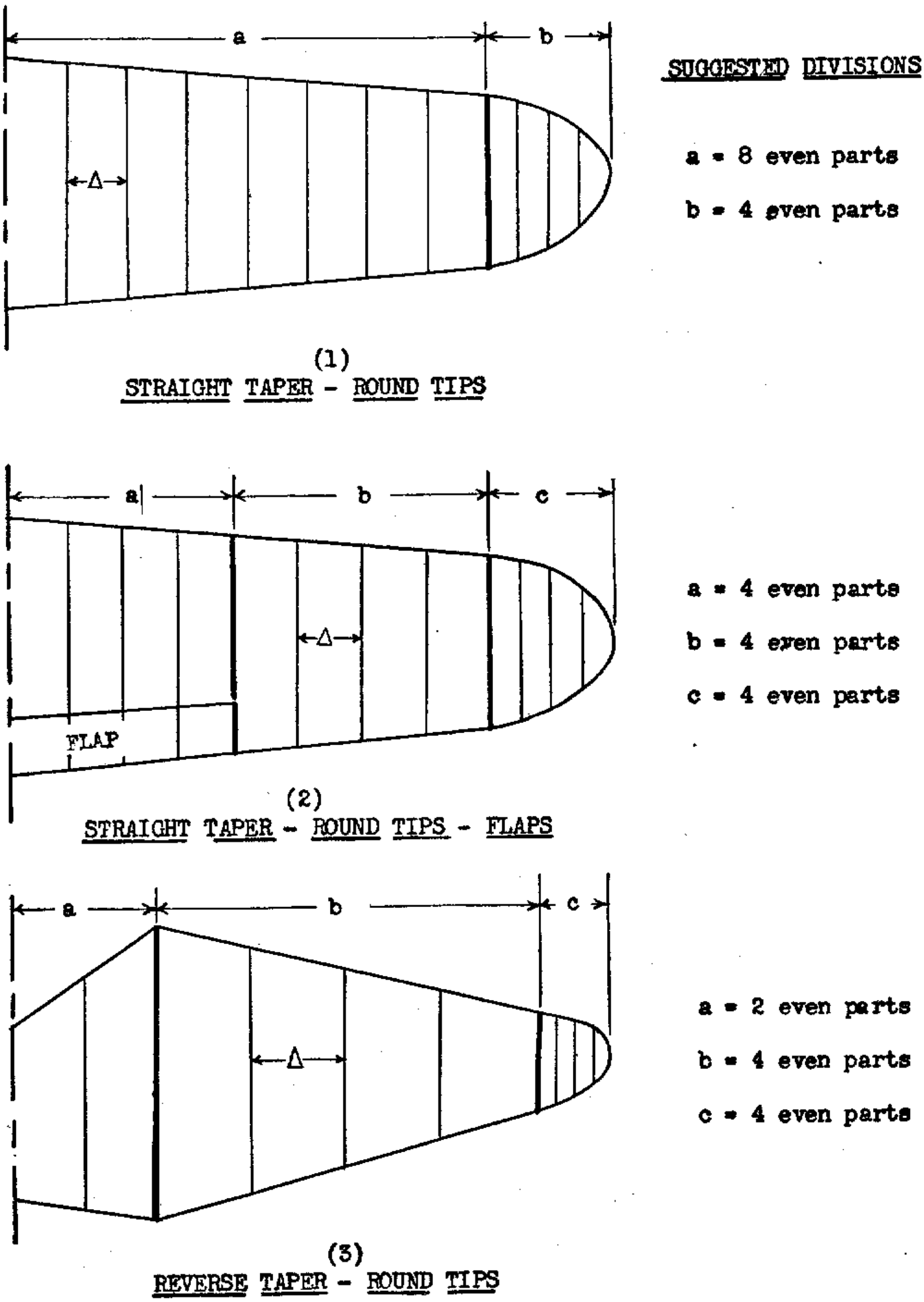
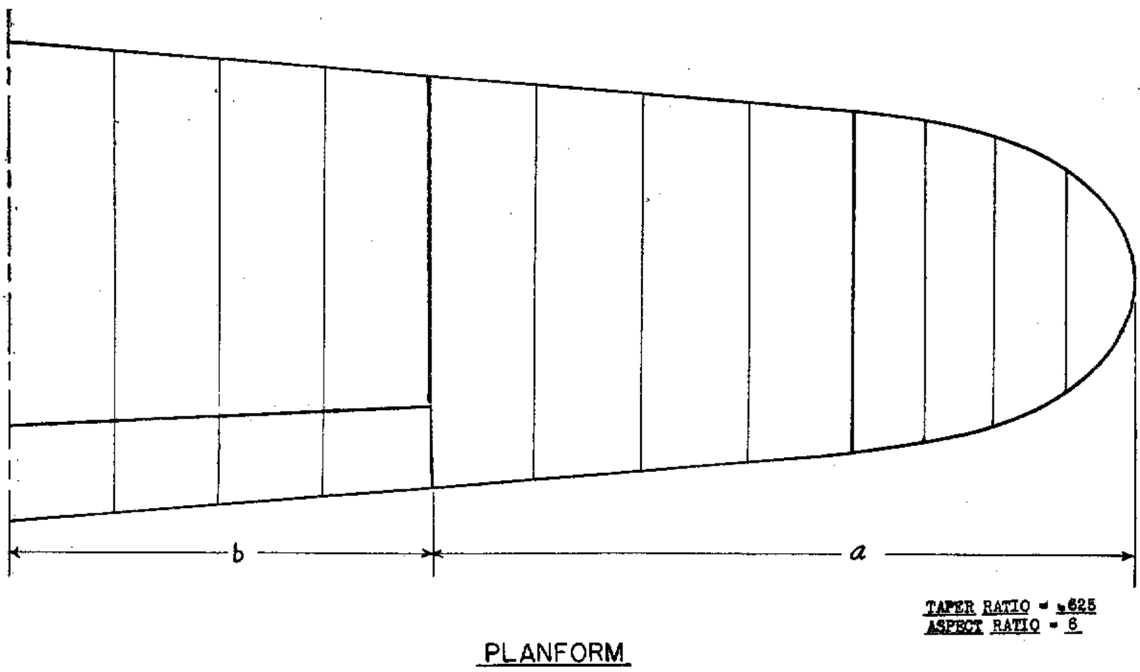


Figure 62.—Examples of wing planform division.



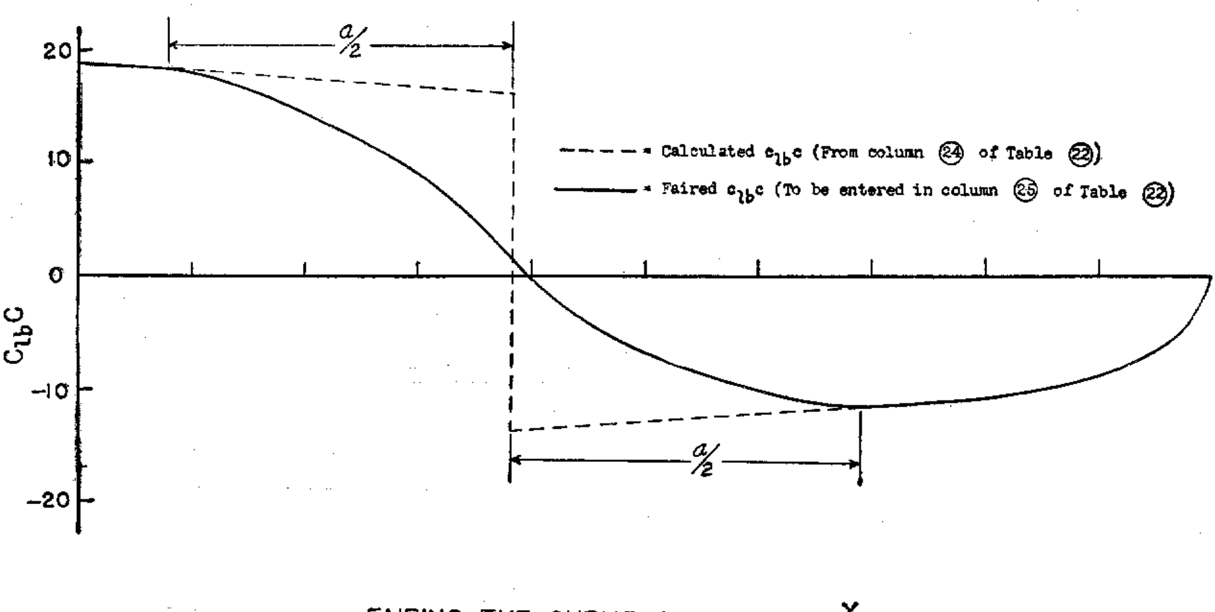


Figure 63.—Comparison example from NACA T. R. 585.

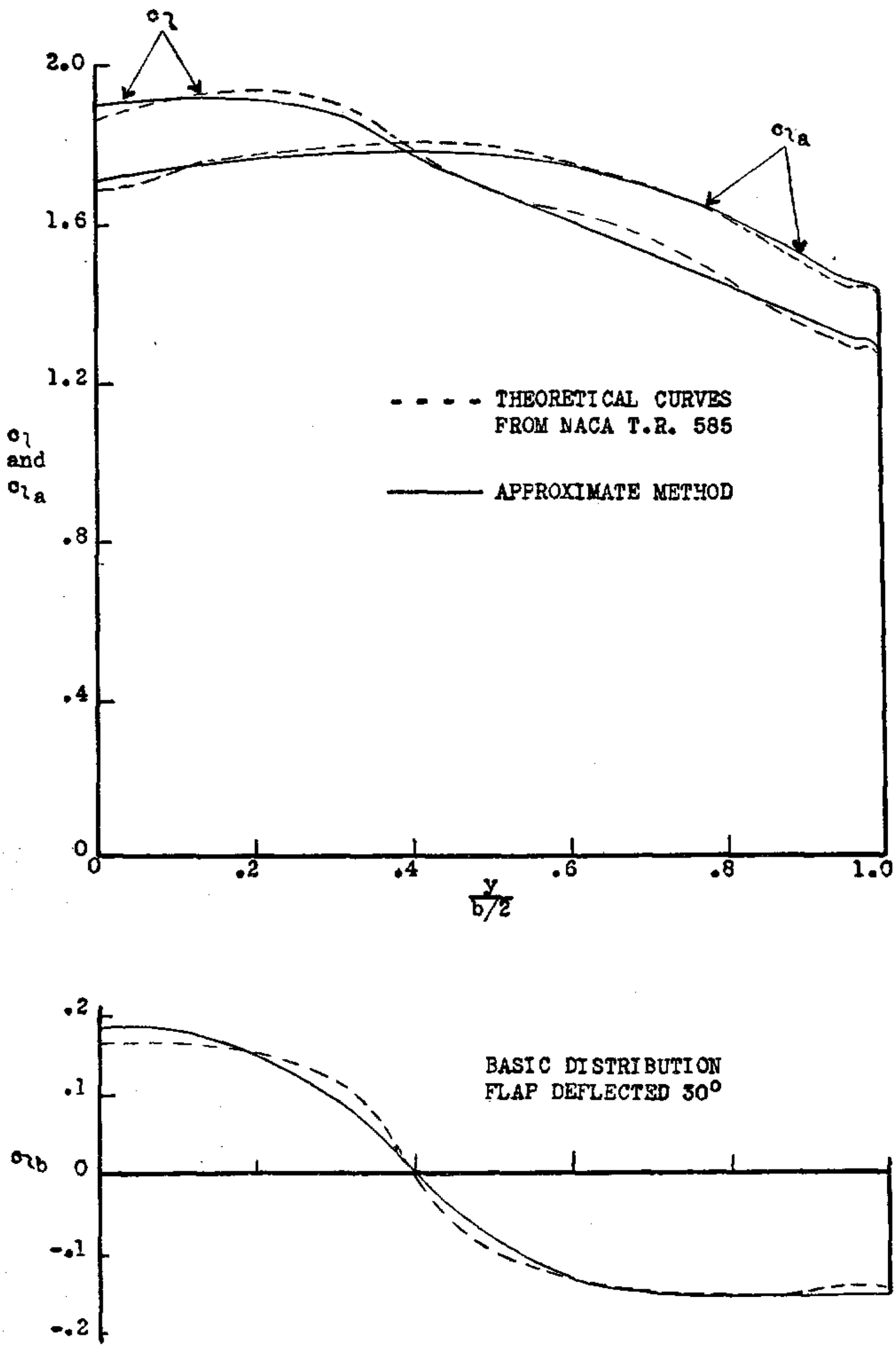


Figure 64.—Comparison example from NACA T. R. 585.

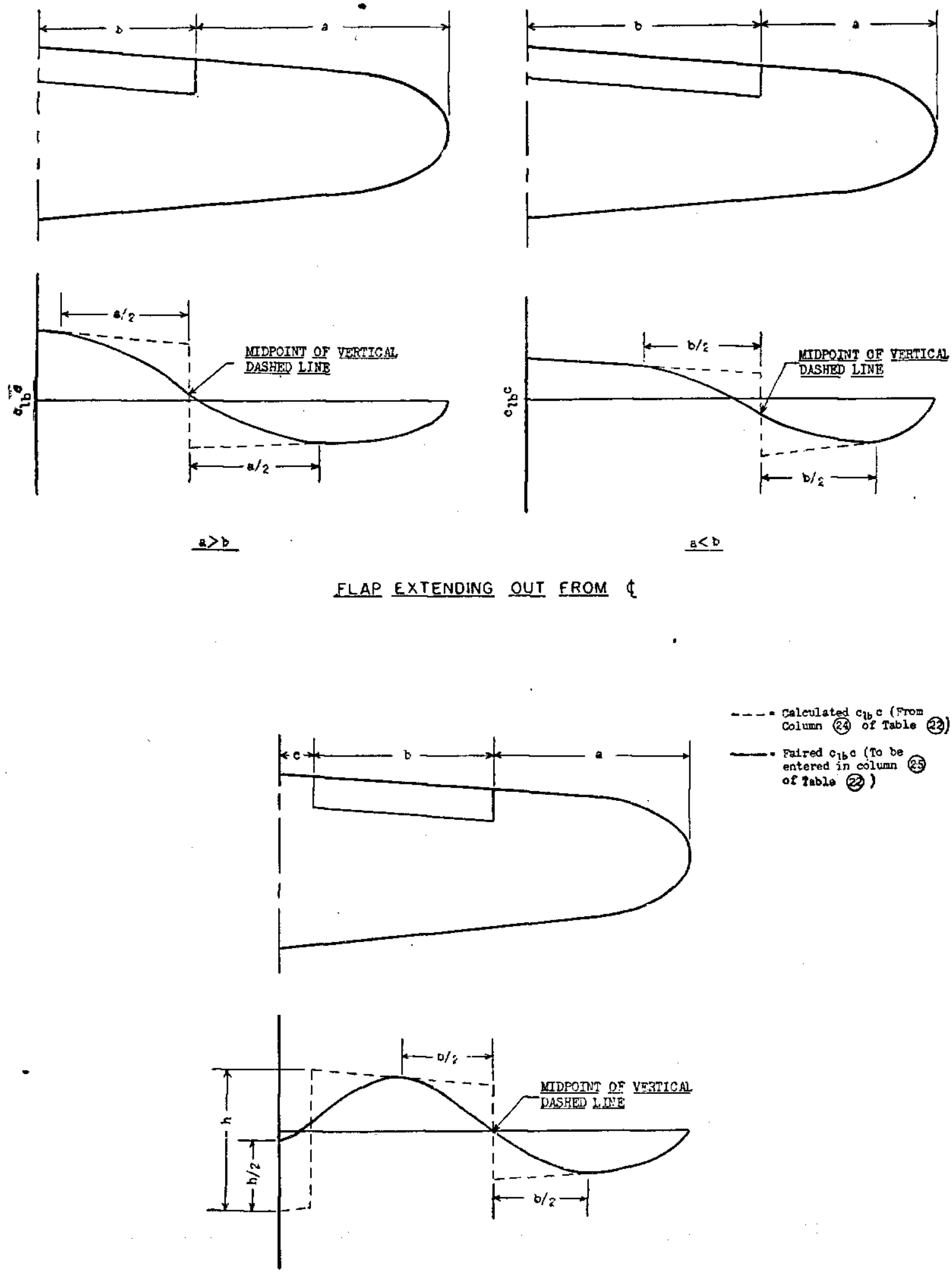


Figure 65.—Examples of fairing the curve of $c_{lb}c$ vs. $\frac{y}{b/2}$.

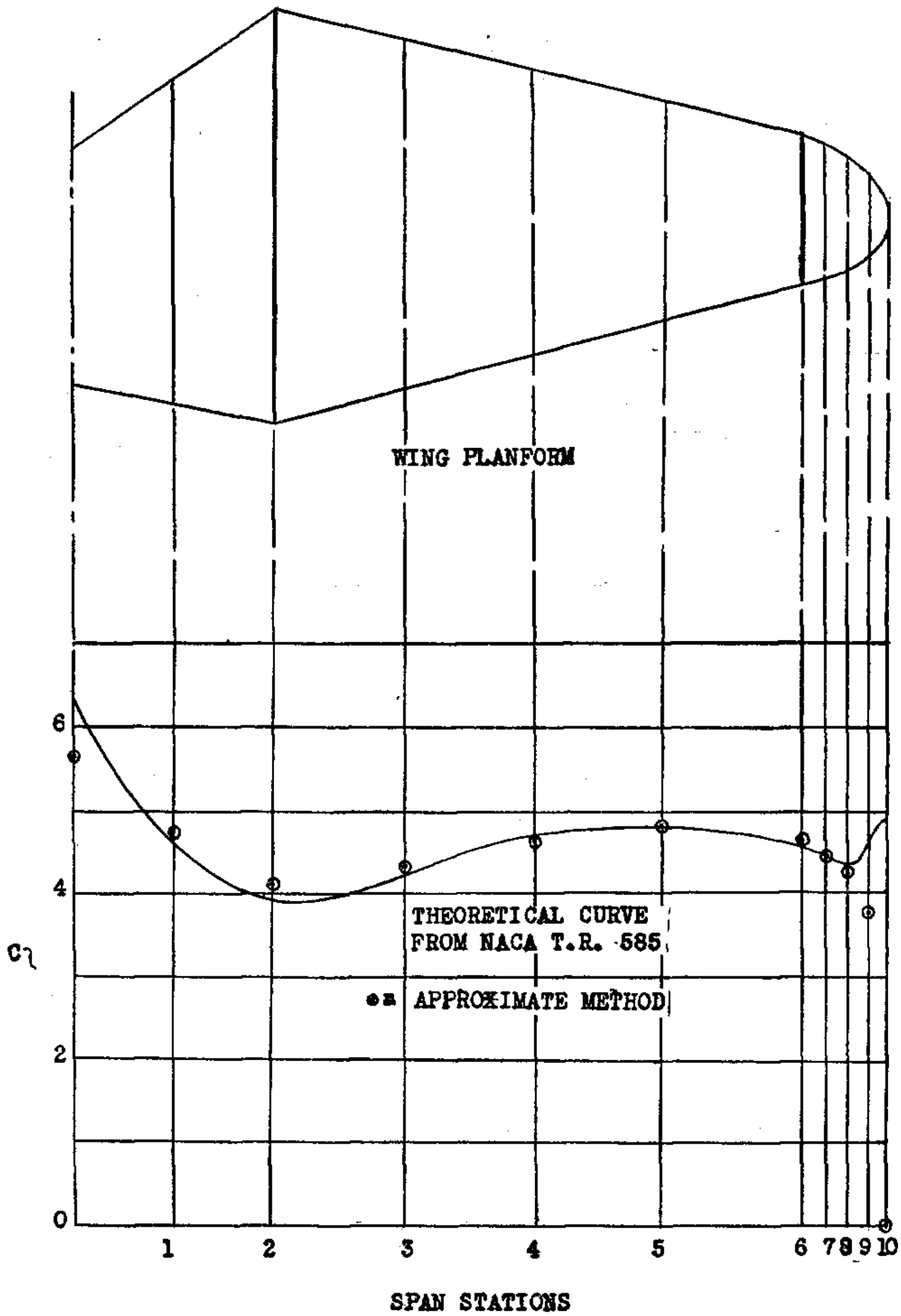


Figure 66.—Wing with reverse taper comparison example from T. R. 585.

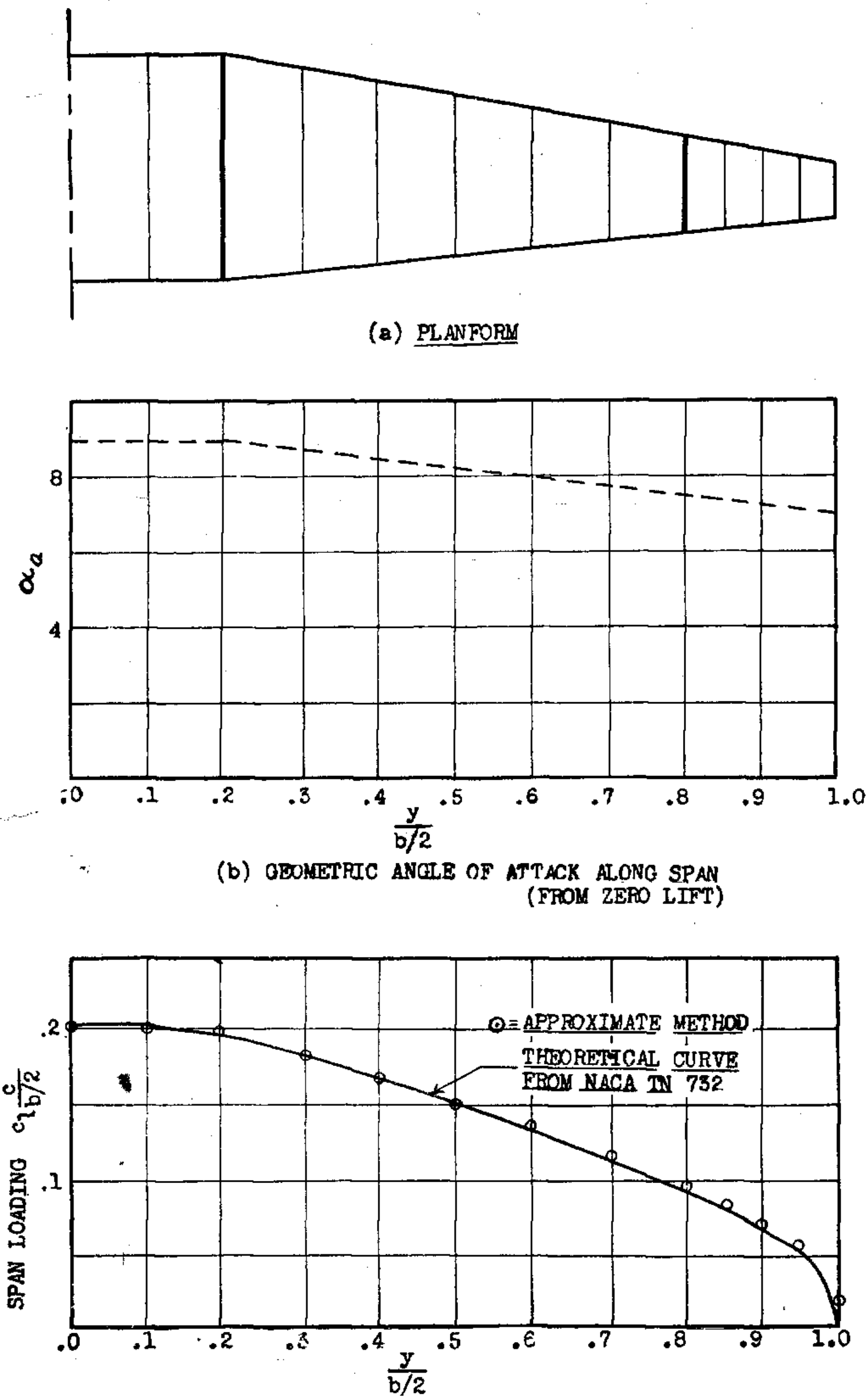


Figure 67.—Comparison example from NACA T. N. 732.

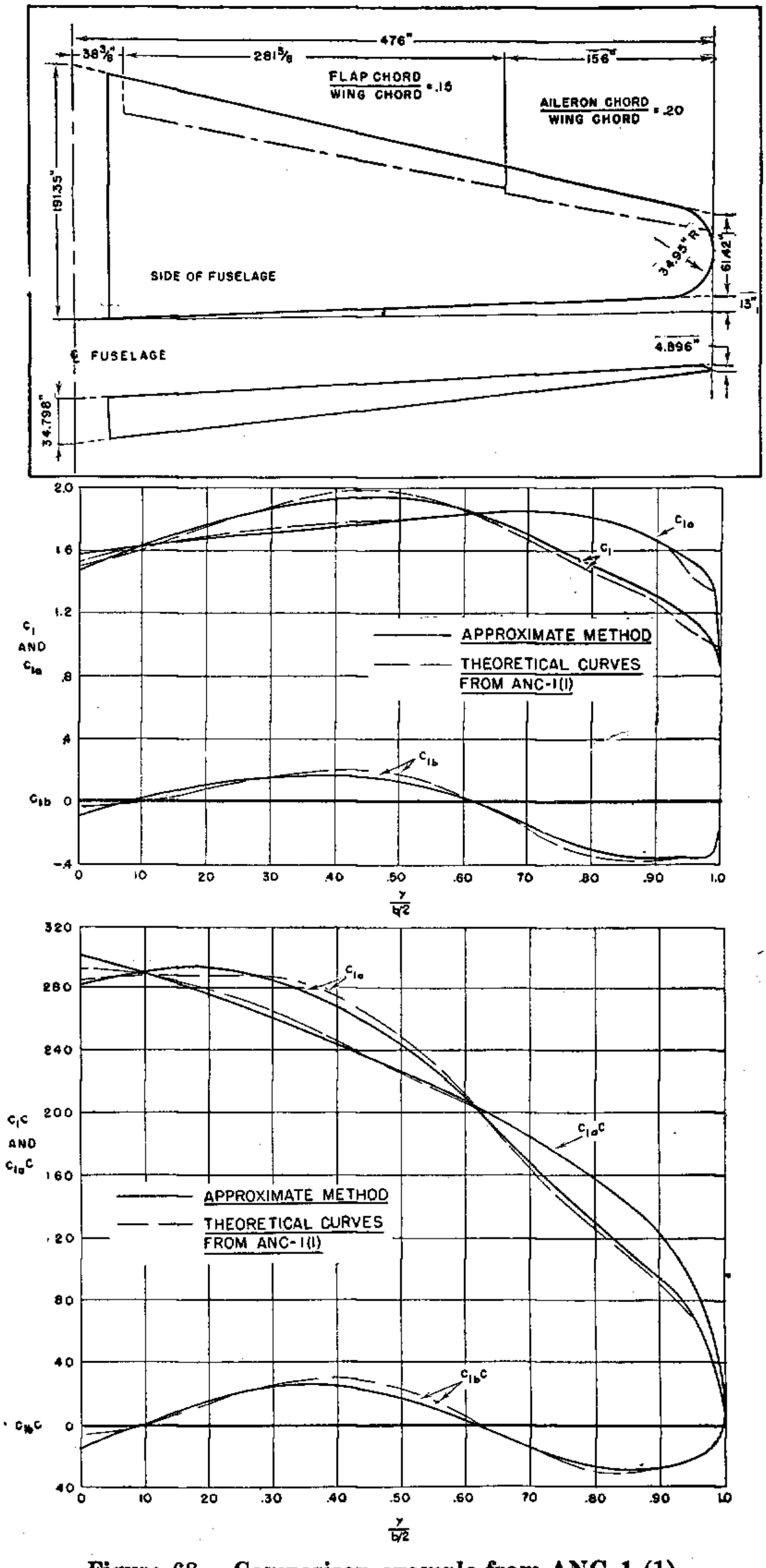


Figure 68.—Comparison example from ANC-1 (1).

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