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REQUIREMENTS FOR APPROVED TYPE CERTIFICATES

AIRPLANE STRUCTURES
AIRPLANE ENGINES
AIRPLANE PROPELLERS

TOGETHER WITH

RECOMMENDED PRACTICE

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PREFACE

The Air Commerce Regulations, effective as amended June 1, 1928, which establish the factors to be considered in determining the airworthiness of airplanes are as follows:

Sec. 10. Airworthiness Requirements.

(*A*) AIRWORTHINESS FACTORS.—In determining the airworthiness of airplanes the following factors are taken into consideration:

(1) The structural strength of wings, ailerons, tail surfaces, fuselage including engine mount, fittings, control system, and landing gear, as shown by stress analyses.

(2) Cockpit, cabin, and control arrangements.

(3) Power plant and power-plant installation.

(4) Equipment and instruments.

(5) Propellers.

(6) Design of fittings.

(7) Materials and workmanship.

(8) Flying characteristics and qualities.

Certain of these items may be demonstrated by theoretical analyses and drawings, others by visual inspection, and others by tests.

(*B*) LOAD-FACTOR REQUIREMENTS.—The load-factor requirements for the structural members of an airplane shall be determined as follows:

(1) *Flying conditions.*—Figure 1 hereof, and hereby made a part of these regulations, shall be used to determine the load factors for the high angle of attack condition of landplanes and float-type seaplanes. Reference shall be made to it and the factor obtained for the appropriate weight and horsepower loading, the factors for gross weights which lie between 2,500 and 12,500 pounds being determined by interpolation between the appropriate curves. For multi-engined airplanes the horsepower to be used for computing the loading in pounds per horsepower shall be that necessary to maintain level flight with full load at sea level. For flying boats and amphibians of 5,000 pounds gross weight or less the factor for 5,000 pounds shall be used; for gross weights in excess of 5,000 pounds the factors shall be those indicated by the actual weight and power loading.

In the low angle of attack condition the load factor shall be 65 per cent of the high angle of attack factor, but in no case less than 3.

For the inverted-flight and nose-dive conditions the load factor shall be 40 per cent of the high angle of attack factor, but in no case less than 2.

(2) *Landing conditions.*—The load-factor requirements for landing conditions shall be obtained from Table 1. If the landing gear is designed to absorb landing shocks by the flow of a liquid through an orifice (the oleo or oleo-pneumatic type), the design load factors

'or the landing gear may be reduced by not more than 25 per cent ›m those shown in Table 1, but suitable provision shall be made to carry taxiing shocks after the absorber has been forced to the full extent of its travel.

For float seaplanes the floats and the members to which they are attached, including the "carry-through" tubes in the fuselage, shall be designed for a factor in landing of 8.0.

The shock absorber in the chassis of a landplane or amphibian shall be designed to absorb the energy generated by a free drop of the air-

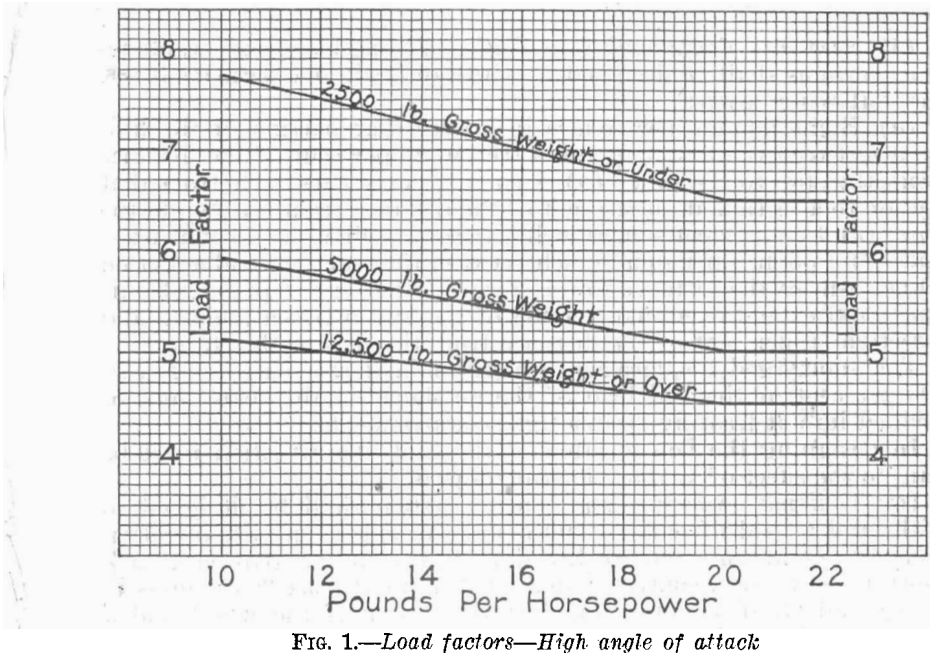

FIG. 1.—*Load factors—High angle of attack*

For flying boats and amphibians of 5,000 pounds gross weight or less, use the 5,000-pound gross weight factors. Horsepower loading shall be based on rated engine horsepower except that for multi-engined airplanes the horsepower shall be taken as that necessary to maintain level flight with full load.

NOTE.—It is to be noted that the load factors given in this chart are not sufficiently high to conform to the international requirements throughout the entire range of horsepower loading. Manufacturers designing airplanes for export should take into account the load factors and design requirements of the country to which the airplane is to be exported.

plane from the height specified in Table 1 without subjecting the airplane or landing gear to forces greater than those corresponding to the load factors for landing.

The tail-skid shock absorber shall be designed to resist a free drop in the three-point landing attitude equal to that shown in Table 1 without imposing loads on the tail skid or fuselage greater than those corresponding to the load factors for landing.

(3) *Tail surfaces and ailerons.*—Tail surfaces and ailerons shall be designed to withstand the loads specified in Table 1.

TABLE 1

Class	Factor for landing conditions	Height of free drop for design of shock absorber (inches)	Loads per square foot on ailerons and horizontal tail surfaces	Vertical tail surfaces
2,500-pound or less	6.5	24	30	75 per cent of load required on horizontal surfaces.
5,000-pound	5.5	22	25	
12,500-pound	5.0	20	20	
12,500-pound or over	4.5	18	20	

(4) *Seaplane floats and boat hulls.*—Flying-boat and seaplane-float bottoms shall be designed to withstand, without permanent set, the following loads:

(*a*) Eight pounds per square inch over that portion of the hull lying between the first step and a section 25 per cent of the distance from the step to the bow; (*b*) 4 pounds per square inch from that section to a section at 75 per cent of the distance from the step to the bow; (*c*) 4 pounds per square inch between the first and second steps; and (*d*) 4 pounds per square inch between the rear step and a section 50 per cent of the distance to the stern. If but one step is used, this load shall extend over that portion of the hull between the step and a section 50 per cent of the distance from the step to the stern.

The submerged displacement of the main floats shall be at least 190 per cent of the full load of the airplane. Main floats shall be divided into at least five water-tight compartments.

In designing the bow portion of the hull due attention must be paid to the effect of striking floating objects.

(*C*) CONTROL SYSTEM.—The control system shall be designed to withstand a load of at least 300 pounds in a fore-and-aft direction and 150 pounds in a lateral direction applied at the grip on a stick control; on a wheel control it shall withstand at least 300 pounds in a fore-and-aft direction applied at the center of the wheel and a couple equal to 125 pounds times the diameter of the wheel applied tangent to the rim of the wheel. The rudder controls shall be designed to withstand a load of not less than 300 pounds applied at the point of contact with the foot. On airplanes having a gross weight of 1,000 pounds or less, two-thirds of the above values may be used. In any case, the strength of the control system shall be sufficient to withstand loads on the control surfaces 25 per cent greater than those specified in Table 1 for the design of the surfaces. The entire control system shall be so designed that the rudder, elevators, ailerons, and other movable parts thereof will not jam, and, if necessary, control stops shall be provided to prevent it.

Dual controls on airplanes carrying passengers for hire shall be so constructed or arranged as to prevent passengers from interfering with the course of flight of the airplane.

(*D*) CONSTRUCTION OF COCKPIT AND CABIN.—The cockpit or cabin shall be constructed to afford suitable ventilation, adequate vision to pilot under normal flying requirements, and reasonable protection to pilot and passengers against possible propeller breakage.

Closed cabins of airplanes carrying passengers for hire shall have not less than two exits, affording maximum ease of operation. One

of such exits may be an emergency exit, in which case the size, location, and method of operation shall be subject to the approval of the Secretary of Commerce.

(*E*) POWER-PLANT REQUIREMENTS.—Engines which have passed the regular endurance tests of the United States Army Air Corps or the United States Navy Bureau of Aeronautics will be approved by the Department of Commerce. Other engines submitted for approval will be tested by the Department of Commerce at the Bureau of Standards.

The log shall include the following:

Piston displacement, compression ratio, type of ignition, carburetor and spark plugs; barometric reading and complete list of instruments and apparatus used for the test. The length of brake arm and gear ratio if propeller is geared shall also be given. The test shall be such as to develop the following data:

(1) A plot of power developed at various revolutions per minute with the throttle wide open.

(2) Record of log showing readings, at approximately 30-minute intervals, of—

(*a*) Revolutions per minute.
(*b*) Brake load in pounds.
(*c*) Oil pressure and temperatures.
(*d*) Water temperatures.
(*e*) Air temperature at carburetor intake.
(*f*) Blast velocity.
(*g*) Maximum cylinder temperature, head and barrel.
(*h*) Manifold depression in inches of mercury.
(*i*) Fuel time in seconds for at least 5 pounds.
(*j*) Oil in pounds.
(*k*) Brake horsepower.
(*l*) Remarks, including breakages and replacements.

The 50-hour endurance test shall be run in 10 five-hour periods. During the first 5 hours the engine shall be run with the throttle wide open, the speed being at least equal to the rated speed and the power being at least 10 per cent in excess of the rated power. During the remaining 45 hours the engine shall be run at approximately rated speed and the horsepower developed should at no time be less than the rated horsepower.

(3) Photographs showing set-up of the testing apparatus used and of any parts which failed or showed excessive wear.

(4) List of instruments and apparatus used.

(5) After completion of the test, and a complete tear down, a report of detailed inspection of engine parts, particular attention to be paid to excessive wear and signs of failure.

An engine, after successfully passing the required block tests, shall be installed in an airplane and given actual service tests.

A newly constructed engine of a type and design which has been tested in accordance with the provisions hereof must not be installed in an airplane until bench tested for at least two hours, during at least one-half hour of which the engine must be run at full throttle.

An engine which has been in storage for more than one year must not be installed in an airplane until it has been reconditioned in accordance with accepted safety practices.

A reconditioned engine must be run for at least 20 minutes at full throttle before it shall be used in propelling registered aircraft carrying persons or property for hire.

(*F*) Power-plant installation.—With water-cooled engine installation a fire wall of terneplate, galvanized steel, asbestos-aluminum combination, or equivalent thereof, shall be installed, insulating the engine section from other parts of the airplane. Aluminum or aluminum-alloy fire walls will be satisfactory for air-cooled engine installations. Openings made in the fire wall to allow the passage of pipes or wires shall be provided with suitable glands or gaskets.

Carburetor air intakes must open outside the cowling and shall be suitably drained. Exhaust manifolds shall be installed outside of cowling unless specifically authorized by the Secretary of Commerce.

Throttle-control and ignition switches on multi-engined airplanes shall be arranged to permit separate and simultaneous control.

Suitable ventilation shall be provided for the engine compartments.

Air-pressure gasoline-feed systems shall not be used without approval of the department.

Airplanes carrying passengers for hire shall have installed an adequate reserve-gasoline supply tank, or satisfactory, reliable apparatus for indicating to the pilot a depletion of the gasoline supply.

(*G*) Propellers.—Propellers which satisfy any of the following requirements will be acceptable:

(1) Designs from which a propeller has been flown for 150 hours without failure on an engine of equal or greater power than that for which the approved rating is sought. Affidavit specifying the number of hours of satisfactory service on a given engine and aircraft, the name of the operator, operating firm, and other pertinent data shall be forwarded to the Secretary of Commerce, together with detailed drawings of the propeller.

(2) Designs which embody only minor modifications from propellers accepted and used by the United States Army or Navy, the contractor to state specifically the departure, if any, from Government design.

(3) Designs which are approved designs, cut down to smaller diameters without reduction of the maximum ordinates, except in the immediate neighborhood of the tip.

Propellers not in accordance with the foregoing shall be tested in the manner prescribed by the Secretary of Commerce.

Propellers must have a ground clearance of at least 6 inches when the airplane is in a horizontal position.

On multi-engined airplanes a clearance of at least 1 inch must be provided between the tips of the propellers and the fuselage or any part of the structure.

The seats shall be so arranged that pilots and passengers are more than 12 inches forward or aft of the propeller disks. Surfaces near the propeller tips must be suitably stiffened against vibration.

(*H*) Performance requirements.—(1) The following flight tests with the design gross load shall be required:

(*a*) General flight, which includes a half-hour flight test with full load, to determine stability.

(*b*) General maneuverability, which includes, among other things, a flight with full load around two pylons or buoys 1,500 feet apart, making five successive figure 8's at 1,000 feet without varying more than 200 feet in height in a radius not to exceed the following:

Five hundred feet for airplanes of full load not in excess of 5,000 pounds.

Seven hundred and fifty feet for airplanes of full load over 5,000 pounds and not in excess of 12,500 pounds.

One thousand feet for all other airplanes.

(2) The landing speed shall not exceed 60 miles per hour when the airplane is provided with a braking device, or 50 miles per hour when not so provided. Landing speed may be assumed equal to the calculated stalling speed. If the foregoing landing speeds are exceeded, the following flight tests will full load shall be required:

(*a*) Take-off within 1,500 feet.

(*b*) Climb at least 250 feet the first minute after taking off.

(*c*) Land, coming to a full stop without external aid, within 1,000 feet from the point where the landing gear first touches the landing area.

(3) Tests are to be calculated upon the basis of air of a specific weight of 0.07635 and still-air conditions.

(4) The applicant must provide the person to make the flight tests, but an inspector from the Department of Commerce may pilot the airplane during such parts of the tests as may be deemed necessary.

(5) No persons other than the crew necessary to conduct the flight shall be carried during performance tests. Sand or similar ballast shall be carried as a substitute for the passengers or cargo ordinarily carried as pay load.

(*I*) EQUIPMENT AND INSTRUMENT REQUIREMENTS.—The equipment and instruments required, which must be serviceable and in operating condition, are:

(1) *Equipment.*—(*a*) Fire-extinguishing equipment, of design approved by the Secretary of Commerce. Cabin airplanes carrying passengers for hire must be equipped with at least one portable extinguisher accessible to the passengers.

(*b*) First-aid kits on airplanes carrying passengers for hire.

(*c*) Safety belts or equivalent apparatus for pilots and passengers in open-cockpit airplanes carrying passengers for hire. Seats or chairs in cabin planes shall be firmly secured in place.

(*d*) Electrical equipment and installation must be approved by the Secretary of Commerce.

(2) *Instruments.*—(*a*) Tachometer for each engine.

(*b*) Oil-pressure gauge where oil-pressure systems are employed.

(*c*) Water thermometer for water-cooled engines and oil thermometer for air-cooled engines.

(*d*) Altimeter.

(3) *Compass requirements.*—An airplane flying cross country 100 or more miles and an airplane, including a seaplane, operating over large bodies of water beyond the sight of land must be equipped with a compass.

(*J*) MANUFACTURER'S IDENTIFICATION DATA.—(1) The date of manufacture or date of remodeling and the name of the manufacturer or remodeler, together with the manufacturer's serial number and type of engine, must be permanently affixed in a visible location by means of a metal plate in the pilot's cockpit or compartment, of each airplane, in order that it may be distinguished from all other aircraft.

(2) The following specifications shall be stenciled or otherwise displayed on the outside of the fuselage or cabin at a point adjacent to the entrance into the cockpit or cabin in such manner as to be readily visible to persons proposing to enter the same: Over-all span; over-all length; weight empty; useful load; gross weight; seating capacity, exclusive of crew; gasoline capacity; oil capacity.

(3) Upon airplanes manufactured under an approved type certificate the manufacturer thereof may display the insignia shown herewith, in conjunction with the above data required to be stenciled on the airplane.

The insignia, if used, shall be 16 inches in width and 3 inches in height and shall be identical with that shown except that the name

FIG. 2

of the manufacturer shall appear above it in the manner indicated, and the number of the approved type certificate under which the identical airplane was manufactured shall appear in the circle thereof. The stenciled data shall appear under it in the manner and proportion shown. This insignia shall only be used on airplanes for which application for license has been made, or for which license has been issued, and must be obliterated in the event the airplane is subsequently disapproved for license.

Sec. 11. Approved Type Certificate, Requirements for.

(*A*) APPLICATION.—A manufacturer of airplanes, engines, or propellers in quantities and of an exact similarity of type, structure, materials, assembly, and workmanship may, at the option of the manufacturer, file with the Secretary of Commerce an application under oath for an approved type certificate accompanied by the data specified, in duplicate. The design or test data must bear the signature of the responsible engineer.

(*B*) Airplanes.—In order to represent the structure of an airplane satisfactorily the data submitted shall include—

(1) A general assembly drawing of each wing, with the sizes and locations of spars, drag struts, drag wires, leading edge, trailing edge, and members supporting the ailerons and controls clearly indicated.

(2) Drawings of a typical rib, showing method of attachment to the spars and leading edge.

(3) Line diagrams of the external wing bracing, showing truss dimensions, sizes of struts, and wires.

(4) Layout of the fuselage structure, showing sizes of all members of the primary structure.

(5) Drawings of the tail surfaces, showing sizes of spars, torque tubes, internal and external braces, location of hinges, and masts.

(6) Chassis drawings giving a layout of the landing gear that shows sizes of struts, axle, bracing wires, shock-absorber spindles, spreader tubes, and other important members and the method of connecting them. Drawings of the tail skid, showing the skid, its connection to the main fuselage structure, and the method of steering it, if any.

(7) For seaplanes, drawings of the floats, showing their lines, general dimensions, and a layout showing sizes of struts and wires and method of attachment to the fuselage.

(8) Detail drawings of the mechanisms used to adjust the stabilizer, fin, wing flaps, or similar surfaces while in flight.

(9) Drawings showing the main wing fittings and the control system are required. The analyses must cover the investigation of the strength of the main members of the wings, tail surfaces, landing gear, including tail skids, fuselage, fittings, and control systems for the load factors required in section 10. The analysis of secondary members carrying heavy loads and the investigation of main members subjected to eccentric loads are required.

(10) Stress analysis of the control surfaces and control systems may be supplanted by tests made on these units, assembled as in flight, if the tests show that the unit will carry 125 per cent of the design load, and if the stick, rudder bar, etc., withstand the loads specified in section 10 (*C*).

(11) The stress analysis shall state the material used for members, whether or not it is heat-treated, and what physical properties are guaranteed, not assumed, by the manufacturer. If metal members of special design are used, test data showing their strength properties under loads similar to those to which they will be subjected in the structure must be submitted.

(*C*) Engines.—Data on engines, showing that the requirements of section 10 (*E*) and (*F*) have been complied with, and a complete set of drawings, must be submitted in duplicate by the engine manufacturer when applying for an approved type certificate.

(*D*) Propellers.—Data on propellers, specified in section 10 (*G*), and a complete set of drawings must be submitted in duplicate by the propeller manufacturer when applying for an approved type certificate.

(*E*) Inspection and flight tests of airplanes.—If the submitted design is approved and the aircraft meets the requirements stated herein, it will be inspected for exact similarity with the submitted

design and specifications. Upon passing such inspection, the airplane must undergo the flight tests prescribed in section 10 (*H*). If such tests are passed, an approved type certificate will be issued to the manufacturer.

(*F*) CONSTRUCTION UNDER APPROVED TYPE CERTIFICATE.—The approved type certificate will be issued upon the condition that each quarter the manufacturer will file his affidavit with the Secretary of Commerce, showing the number of airplanes, engines, or propellers constructed under the approved type certificate during the quarter, with a statement that no airplane, engine, or propeller is being constructed, under such certificate, deviating from the terms thereof.

(*G*) CHANGES.—Changes in airplanes constructed under an approved type certificate are permissible, with the approval of the Secretary of Commerce.

(*H*) MANUFACTURER'S AFFIDAVIT.—Upon the sale by the manufacturer of airplanes of an exact similarity of type, structure, materials, assembly, and workmanship with the specimen for which the approved type certificate is issued, the manufacturer may deliver to the purchaser a manufacturer's affidavit, copies of which will be furnished the manufacturer upon request.

REQUIREMENTS FOR APPROVED TYPE CERTIFICATES

Airplane Structures
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Airplane Propellers

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SECTION I. AIRPLANE STRUCTURAL REQUIREMENTS AND RECOMMENDED PRACTICE

PART 1.—GENERAL

1. As a result of the discussion before a committee of engineering representatives of the aircraft manufacturers and the Department of Commerce, at the aeronautical conference held by the department in Washington during December, 1927, it was unanimously decided that the following set of rules embodying the requirements for structurally airworthy airplanes should be issued by the department, so that the industry might have a guide as to what will be required on new designs. It was fully appreciated that it would be impossible to develop a set of requirements that would apply to all types of airplanes, so an effort was made to draft the requirements in such a way that they would apply only to conventional types which may be licensed by the department for carrying persons or property for hire. With the constant advances being made in the science of aviation, new types of aircraft and new types of construction will doubtless be developed that can not be made to conform with any rigid set of rules. In cases where the deviation from conventional practice is but slight, the department will consider approving the aircraft if sufficient evidence is submitted to enable the department's engineers to determine whether or not the proposed deviation will affect the airworthiness of the design. Where the deviation from conventional practice is marked, the department may grant a special aircraft license, so that each case may be adjudged on its merits. Such aircraft will, by their very nature, be experimental and as such can not be licensed for general commercial use until their airworthiness has been established. The department has no desire to hamper development by requiring that aircraft conform in every respect to the rules given in the following pages.

2. These rules are based on the present development in the science of airplane structural design, and experience indicates that when applied to conventional types they will give a safe and well-proportioned structure. In developing unconventional types of structure these rules may be used as a guide in determining the general strength requirements, but it is fully appreciated that they can not be rigorously applied.

3. The procedure followed by the department when approving designs for a manufacturer's approved type certificate is as follows:

The drawings, which the manufacturer is required to furnish in duplicate, are examined for deviations from what is considered good practice, and they are then checked against the stress analyses to ascertain whether or not the structure investigated conforms with that to be built.

The stress analysis, which is also required to be submitted in duplicate, is checked for errors of assumption, for deviations from the approved methods, and for errors in arithmetic. If the strength of the structure is shown by the anlayses to meet the requirements listed in part 2 of Section I, herein, and if no deviations from what is commonly accepted as good practice are noted, the structure is approved.

If the submitted design is approved and the aircraft meets the requirements stated herein, it will be inspected for exact similarity to the submitted design and specifications. Upon passing such inspection, the airplane must undergo the flight tests prescribed in paragraph 11 of this part 1. If such tests are successfully accomplished and suitable manufacturing facilities are in evidence, an approved type certificate will be issued.

One set of the drawings and analyses is then impressed with the seal of the Department of Commerce and returned to the manufacturer to be used in the construction of his airplanes. The duplicate copy is placed in the department's files. Airplanes built to conform exactly to these approved drawings may, upon the manufacturer making affidavit to that fact, be accepted as structurally airworthy. Any deviations must, however, be approved by the department before being incorporated in the construction of any airplane.

The procedure followed in the case of licensing single airplanes of any type will be similar.

The approved type certificate will be issued upon the condition that each quarter the manufacturer will file his affidavit with the Secretary of Commerce, showing the number of airplanes constructed under the approved type certificate during the quarter, with a statement that no airplane is being constructed under the approved type certificate deviating from the terms thereof.

Technical data furnished by manufacturer for approved type certificate will be treated as confidential.

4. It is obvious that the foregoing procedure places upon the manufacturer the burden of proving that his design is structurally safe, and he should keep this in mind when submitting data to the department. In the majority of cases the department's engineers will not be able to see an airplane of the type they are examining, at the time their check is made, and the manufacturer must therefore

make his drawings and analyses as clear as possible if he wishes to avoid delay and controversy in obtaining an approval. Mere statements of opinion should never be included in an analysis without presenting the facts upon which the opinion is based. Deviations made from the methods of analysis herein described, because the structure being investigated does not conform with the conventional, or because the manufacturer has data indicating that his method is safe, must be accompanied by a clear explanation of the reasons for the deviation and supported by any data which may be available.

5. The drawings must be complete and well dimensioned and must show the structure in sufficient detail to be used for the construction of airplanes to be built under an approved type certificate. Assembly drawings of the major structural units, such as wings, stabilizer, elevator, etc., will suffice if they are completely dimensioned and if they show the cross sections of all wooden members, or metal members of special design, and the sizes of standard wires, tubes, etc., used in the assembly. The location of hinges, control masts, and the points of attachment of all brace struts or wires should be clearly shown. Drawings should be made to scale and important dimensions given. Dimensions referring to lengths of members, distances between supports, etc., are more satisfactory if shown in inch units rather than feet and inches.

In order to represent the structure satisfactorily, the drawings submitted should include:

(*a*) A general assembly of each wing with the sizes and locations of spars, drag struts, drag wires, leading edge, trailing edge, and members supporting the ailerons and controls clearly indicated.

(*b*) Drawings of a typical rib showing method of attachment to the spars and leading edge.

(*c*) Line diagrams of the external wing bracing showing truss dimensions, sizes of struts, and wires.

(*d*) A layout of the fuselage structure showing sizes of all members of the primary structure.

(*e*) Drawings of the tail surfaces showing sizes of spars, torque tubes, internal and external braces, location of hinges, and masts.

(*f*) Chassis drawings giving a layout of the landing gear showing sizes of struts, axle, bracing wires, shock-absorber spindles, spreader tubes and other important members, and detail drawings or sketches showing the arrangement of members at their lower ends and the method of connecting them. Drawings of the tail skid showing the skid, its connection to the main fuselage structure, and the method of steering it, if any.

(*g*) For seaplanes, drawings of the floats showing their lines, general dimensions, and a layout showing sizes of struts and wires and method of attachment to the fuselage.

(*h*) Detail drawings of the mechanisms used to adjust the stabilizer, fin, wing flaps, or similar surfaces while in flight are also desired.

(*i*) Drawings showing the main wing fittings and the control system are required.

The analyses must cover the investigation of the strength of the main members of the wings, tail surfaces, landing gear, including tail skids, fuselage, fittings, and control systems for the load factors

required in paragraph 2, part 2, Section I. The analysis of secondary members carrying heavy loads and the investigation of ma members subjected to eccentric loads are required.

Stress analyses of the control surfaces and control systems may be supplanted by tests made on these units, assembled as in flight, if the tests show that the unit will carry 125 per cent of the design load and if the stick, rudder bar, etc., withstand the loads specified in paragraph 5*d*, part 3, Section I. It is recommended that both analyses and tests be made on these units and on the main wing fittings, though both are not required.

It is particularly important that the stress analysis states the material used for various members, whether or not it is heat treated, and what physical properties are guaranteed, not assumed, by the manufacturer. If metal members of special design are used, test data showing their strength properties under loads similar to those to which they will be subjected in the structure must be submitted.

An analysis should be carefully checked as to arithmetic before being submitted for approval, as the department will return analyses for correction when arithmetical errors are found, except when their effect is obviously negligible.

The stress analysis shall bear the signature of the responsible engineer.

6. The cockpit must be constructed to afford suitable ventilation, adequate vision to pilot under normal flying requirements, and reasonable protection to pilot and passengers against possible propeller breakage.

Closed cabins of airplanes carrying passengers for hire must have not less than two exits, affording maximum ease of operation. One of these exits may be an emergency exit, in which case the size, location, and method of operation shall be subject to the approval of the department in each individual case.

Dual controls on airplanes carrying passengers for hire shall be so constructed or arranged as to prevent passengers from interfering with the course of flight of the airplane.

7. A fire wall of terneplate, galvanized steel, or asbestos-aluminum combination must be installed, insulating the engine section from other parts of an airplane with water-cooled engine installation. Aluminum or aluminum-alloy fire walls will be satisfactory for air-cooled engine installations. Openings made in the fire wall to allow the passage of pipes or wires must be provided with suitable glands or gaskets.

Carburetor air intakes must open outside the cowling and must be suitably drained. Exhaust manifolds shall be installed outside of cowling unless otherwise specifically authorized by the Department of Commerce.

Throttle control and ignition switches on multi-engined airplanes must be arranged to permit separate and simultaneous control.

Suitable ventilation must be provided for the engine compartments.

Air-pressure gasoline-feed systems must not be used without approval of the department.

Airplanes carrying passengers for hire must have installed an adequate reserve gasoline-supply tank, or satisfactory, reliable apparatus for indicating to the pilot a depletion of the gasoline supply.

8. Electrical equipment.—(*a*) All electrical equipment shall be suitably protected from gasoline, oil, water, and other detrimental substances.

(*b*) Electrical wiring shall be well supported and suitably insulated to prevent short circuits; terminals or other bare portions of conducting circuits to be covered with proper insulating material where there is any possibility of inadvertent short-circuiting.

(*c*) Electrical circuits for lights employing a generating system shall contain suitable fuses which will break the circuit at a sufficiently small current flow to adequately protect the lights and other parts of the circuit. Fuses shall be so located that they can be readily replaced during flight. On aircraft for which lights are required a complete set of spare fuses shall be carried.

(*d*) Batteries which contain acid or other corrosive substance shall be provided with pans or equivalent provision for the collection of possible leakage of such substance. That part of the airplane structure which is within 12 inches of such batteries shall be protected with a suitable acid-proof coating.

(*e*) Batteries shall be easily accessible and adequately isolated from gasoline or oil tanks and from gasoline or oil pipes. Suitable covers shall be provided.

(*f*) There shall be a minimum clearance of 2 inches between all electrical wiring and gasoline or oil tanks; gasoline or oil pipes, carburetors, exhaust pipes, and moving parts.

(*g*) If the structural members of an airplane are of metal, a ground-return or single-wire lighting circuit may be used, the airplane acting as the "ground" or return conductor. Wherever a conductor is connected to ground, it must be accessible for repair. Ground-return connections shall be made to members of the primary structure, and the surfaces on which the terminals make contact shall be clean and free from oxide or paint.

(*h*) In the case of airplanes in which the structural members are of wood or other nonconducting material a two-wire or insulated-return lighting system shall be used.

(*i*) All switches installed in airplane electrical systems must be two-pole switches connected in such manner that the minimum possible length of conductor is connected to the battery or generator when the switch is off.

9. Propellers must have a minimum ground clearance of 6 inches when the airplane is in a horizontal position.

On multi-engined airplanes a clearance of at least 1 inch must be provided between the tips of the propellers and the fuselage or any part of the structure. The seats shall be so arranged that pilots and passengers are more than 12 inches forward or aft of the propeller disks. Surfaces near the propeller tips must be suitably stiffened against vibration.

10. (*a*) Except as provided in subparagraph (*b*), all airplanes must have a landing speed below 60 miles an hour when provided with brakes and 50 miles an hour when not provided with brakes, and must climb at least 250 feet the first minute after taking off. Landing speed may be assumed equal to the calculated stalling speed.

(*b*) If the above landing speeds are exceeded, the airplane must pass the following flight tests with full load:

1. Take off within 1,200 feet.

II. Land, coming to full stop without external aid within 1,0[illegible] feet from the point where the landing gear first touches the landi[illegible] area.

11. The flight tests required are of two kinds:

(*a*) General flight, which is a one-half hour flying test with full load to determine stability.

(*b*) General maneuverability, which includes among other things a flight with full load around two pylons or buoys 1,500 feet apart, making five successive figure 8's at 1,000 feet without varying more than 200 feet in height in a radius not to exceed the following:

I. Five hundred feet for airplanes of full load not in excess of 5,000 pounds.

II. Seven hundred and fifty feet for airplanes of full load over 5,000 pounds and not in excess of 12,500 pounds.

III. One thousand feet for all other airplanes.

The applicant must provide the person to make the flight tests, but the inspector will pilot the airplane during such parts of the tests as may be deemed necessary.

PART 2.—REQUIRED LOADS FOR AIRPLANE STRUCTURES

1. Definitions.—The National Advisory Committee Report on Nomenclature, No. 240, gives definitions of the words and phrases used in aeronautics, but some of those having special applications in the field of structural design are not included. That the subsequent paragraphs of this section may be clear, the meaning of such words and phrases, as they are commonly employed in the design of airplane structures, is given here.

Aspect ratio.—The aspect ratio of a wing or control surface is the ratio of the square of the span to the wing area. It is defined in this manner in order to make the definition easily applicable to wings that are not rectangular in shape. In computing the aspect ratio of a wing, part of which is covered by nacelles or a fuselage, the portion of the wing so covered shall be included in the wing area. Such covered portions of the wings shall not be included in the wing area, however, in any other computations.

Basic load.—The basic loads on (or stresses in) an airplane or part are the loads on (or stresses in) the airplane or part when the airplane is at rest or in a condition of unaccelerated flight. (Compare design load, ultimate load, load factor, factor of safety, margin of safety, normal load.)

Beam direction.[1]—The direction parallel to the intersection of the plane of the spar web and the plane of symmetry. (Compare chord, drag, lift, side, and vertical directions.)

Beam force (or component).[1]—A force (or component) in the beam direction—that is, parallel to the intersection of the plane of the spar web and the plane of symmetry. (Compare chord, drag, lift, side, and vertical forces.)

Center of gravity.—Usually refers to the center of gravity of the airplane fully loaded. Sometimes used for the center of gravity of a single load or a special set of loads, in which case the load or loads concerned should be stated or clearly implied by the context.

Chord direction.[1]—The direction of the intersection of the plane of the internal-wing truss with the plane of symmetry. When a

wing has two internal trusses in nonparallel planes, the plane bisecting the dihedral angle between those two planes should be used. (Compare beam, drag, lift, side, and vertical directions.)

Chord force (or component).[1]—A force (or component) in the chord direction—that is, parallel to the plane of the internal wing truss and to the plane of symmetry. (Compare beam, drag, lift, side, and vertical forces.)

Design load.—The load for which a member is designed. It is usually obtained by multiplying a basic load by a specified load factor. (Compare basic load, factor of safety, margin of safety, normal load, ultimate load.)

Double-taper (spars).—A spar is double-tapered when both width and depth vary along the span.

Double-taper (wings).—A wing is double-tapered when both the chord and thickness ratio vary along the span. (Compare equivalent wing, taper in plan only, taper in thickness ratio only, thickness ratio.)

Drag direction.—The direction of the relative wind. The angle between the chord and drag directions is usually small and often zero. (Compare beam, chord, lift, side, and vertical directions.)

Drag force (or component).—A force (or component) in the drag direction—that is, parallel to the relative wind. (Compare beam, chord, lift, side, and vertical forces.)

Effective span.—The true span of a wing less corrections for tip loss, area of cut-outs, width of nacelles, etc. For the method of computing the effective span in a given case, see Figures 3, 4, and 5.

Efficiency of wings.—The ratio of the normal gross beam load per square foot on the upper wing to that on the lower. For the method of computation see Figure 2.

Equivalent wing.—A wing of the same span as the actual wing but with the chord at each section reduced in proportion to the ratio of the average beam load at that section to the average beam load at the section taken as standard.

Factor of safety.—The ratio of the design load or the ultimate load to the maximum probable load. In airplane design the least factor of safety is usually about 2. The term is often used incorrectly in place of load factor. (Compare load factor and margin of safety.)

Flying conditions.—The loading conditions of high angle of attack, low angle of attack, inverted flight, nose dive, and maximum tail load.

Gap.—The distance between the planes of the chords of two adjacent wings of an airplane, measured along a line perpendicular to the chord of the upper wing at any designated point of its leading edge.

Gross weight.—The total weight of the airplane, including useful load. (Compare net weight.)

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[1] Beam and chord directions and forces are to be distinguished from lift and drag directions and forces. The latter depend on the direction of the relative wind while the former depend upon the dimensions of the airplane. The beam direction and forces are so named because forces in that direction are considered as loads to be supported by the wing spars acting as beams. The chord direction and forces are so named because they are approximately, at least, in the plane of the wing chord.

High angle of attack.—A loading condition for the wings and fuselage representing the condition of flying at a high angle of attack while pulling out of a dive. (See low angle of attack, inverted flight, and nose dive.)

Inverted flight.—A loading condition for the wings representing the conditions of flying upside down and of commencing a dive. (Compare high angle of attack, low angle of attack, and nose dive.)

Investigation of stresses.—When an investigation of the stresses in a unit in a given condition of loading is called for, only those computations are required that are needed to show which members, if any, should be designed for that condition and the stresses in those members. Often an investigation will show that no members need be designed for the loading condition considered.

Level landing.—A loading condition for the fuselage and landing gear representing conditions in a tail-high landing.

Lift direction.—The direction in the plane of symmetry perpendicular to the relative wind. (Compare beam, chord, drag, side, and vertical directions.)

Lift force (or component).—A force (or component) in the lift direction. (Compare beam, chord, drag, side, and vertical forces.)

Load factor.—The ratio of some other load to the basic load in which the relative distribution of the forces is the same. This other load may be any of the following: The load applied during some special maneuver, the maximum probable load on the airplane or part, the design load, or the ultimate load. Whenever a load factor is mentioned, the context should show clearly what load is being compared to the basic load. (Compare basic load, design load, factor of safety, margin of safety, and ultimate load.)

Low angle of attack.—A loading condition for the wings and fuselage, corresponding to a condition of flying with a small angle of attack while pulling out of a dive, and designed to insure adequate strength in some members that are lightly loaded in the other loading conditions. (Compare high angle of attack, inverted flight, nose dive, and maximum tail load condition.)

Margin of safety.—The ratio of the ultimate strength of a member minus the design load to the design load. Usually a member with a positive margin of safety is satisfactory and one with a negative margin of safety is unsatisfactory. (Compare design load, factor of safety, load factor, and ultimate load.)

Maximum tail load conditions.—Loading conditions for the tail surfaces and fuselage which represent conditions when the loads on the tail surfaces are at a maximum.

Minimum size.—The smallest-sized member that should be used in a given location as determined by considerations of maintenance, loads in handling, and similar practical considerations.

Net weight.—The gross weight, less some specified partial weight. Very often the partial weight is the dead weight of the wings, or it may be the useful load. The partial weight in question should always be clearly indicated in the context.

Normal load.—The load on that part of a wing assumed to be unaffected by tip losses or similar corrections. In any given case it may be a basic, design, gross, net, or ultimate load, depending on the context.

Nose dive.—A loading condition on the wings representing the conditions in a dive at the angle of attack of zero lift. (Compare high angle of attack, low angle of attack, and inverted flight.)

Primary structure.—Each structural member, the failure of which would seriously impair the safety of the airplane, is considered a part of the primary structure.

Restraint coefficient.—A coefficient used in column formulas to represent the degree of restraint of the ends of the columns. For pin ended columns $c=1$, and for rigidly fixed ended columns $c=4$. In airplane work it is very seldom advisable to use a value of c greater than 2.

Side direction.—The direction perpendicular to the plane of symmetry. **(Compare beam, chord, drag, lift, and vertical directions.)**

Side force (or component).—A force (or component) perpendicular to the plane of symmetry. (Compare beam, chord, drag, lift, and vertical forces.)

Stagger.—The angle between a line joining the forward third points of the wings and a line perpendicular to the chord of the upper wing, both lines being drawn in a plane parallel to the plane of symmetry. Stagger is considered positive if the forward third point of the upper wing is in advance of that of the lower. Sometimes stagger is expressed in per cent of gap, in which case the per cent of stagger is equal to the tangent of the angle of stagger. It is indicated by θ in Figure 2.

Taper in plan only.—A wing is "tapered in plan only" when the wing chord varies along the span but all cross sections of the wing are geometrically similar. (Compare double taper, taper in thickness ratio only, and equivalent wing.)

Taper in thickness ratio only.—A wing is "tapered in thickness ratio only" when the wing chord is constant but the cross sections are not geometrically similar. (Compare double taper, taper in plan only, equivalent wing, and thickness ratio.)

Thickness ratio.—The ratio of the maximum thickness of an airfoil section to its chord. (Compare taper in thickness ratio only and equivalent wing.)

Three-point landing.—A loading condition for the fuselage and landing gear representing the maneuver of landing with wheels and tail skid touching the ground simultaneously. (Compare level landing.)

Ultimate load.—The load that caused failure in a test or the load that, according to computations, should cause failure in a member. (Compare basic load, design load, normal load, factor of safety, margin of safety, and load factor.)

Ultimate load factor.—The ratio of the ultimate load to the basic load in the same loading condition. (Compare basic load and ultimate load.)

Useful load.—The crew and passengers, oil, and fuel, ballast other than emergency, and portable equipment.

Vertical direction.—In airplane structural work this is usually assumed to be in the plane of symmetry and normal to the propeller axis. Sometimes, as in the three-point landing condition, it may be assumed normal to some other line, such as the ground line. In any

case the context should always show what direction is assumed to be vertical. (Compare beam, chord, drag, lift, and side directions.)

Vertical force (or component).—A force (or component) in the vertical direction. (Compare beam, chord, drag, lift, and side forces.)

Wing tip.— A line parallel to the plane of symmetry and tangent to the end of the wing. (See fig. 3.)

2. Load factors.—To define the structural strength requirements for commercial airplanes they are divided into two types. The requirements for racers or airplanes that are to be used for acrobatics may be obtained from the department to cover any specific design.

The primary members of the airplane structure shall be, as when new, of sufficient strength to withstand the stresses in them for the conditions listed below.

In the high angle of attack condition the load factors shall be obtained from Figure 1 for the appropriate weight and horsepower loading. The curves, as drawn, apply directly to airplanes with land gear or floats. For flying boats and amphibians of 5,000 pounds gross weight or less, use the factors for 5,000 pounds. For gross weights in excess of 5,000 pounds, the factors shall be the same as for other airplanes.

For multi-engined airplanes, the horsepower to be used in computing the loading in pounds per horsepower shall be that necessary to maintain level flight with full load at sea level.

It is to be noted that the load factors given in Figure 1 are not sufficiently high to conform to the international requirements throughout the entire range of horsepower loading. Manufacturers designing airplanes for export should take into account the load factors and design requirements of the country to which the airplane is to be exported.

The load factors for airplanes whose gross weights lie between 2,500 and 12,500 pounds will be found by interpolation between the appropriate curves. For instance, a 3,600-pound airplane, having a 200-horsepower engine, will have a factor in the high angle of attack condition of $6.75-\frac{3,600-2,500}{5,000-2,500}(6.75-5.20)=6.07$.

In the low angle of attack condition the load factor will be 65 per cent of the high angle of attack factor but in no case less than 3. In inverted flight and nose dive the factors will be 40 per cent of the high angle of attack factor, but in no case shall they be less than 2. The nose-dive condition will be found described more fully in paragraph 14 of this section.

Table 1 gives the load factors for the landing conditions, the height of free drop required for landing-gear design, and the required design load for the tail surfaces. For airplanes whose gross weight lies between 2,500 and 5,000 pounds the landing load factors shall be obtained by interpolation between 6.5 and 5.5. For airplanes between 5,000 and 12,500 pounds the landing factors shall be obtained by interpolation between 5.5 and 5.0.

TABLE 1.—*Load factors*

Class	Factor for landing conditions	Height of drop for design of shock absorber (inches)	Loads per square foot on ailerons and horizontal tail surfaces	Vertical tail surfaces
2,500 pounds or less	6.5	24	30	75 per cent of load required on horizontal surfaces.
5,000 pounds	5.5	22	25	
12,500 pounds	5.0		20	
12,500 pounds or over	4.5		20	

For seaplanes the load factors shall be the same as for land planes, except that the load factor to be used on the floats and the members to which they are attached, including the "carry-through," tubes in the fuselage, shall be designed for a factor in landing of 8.0.

The chassis shock absorber shall be designed to absorb the energy corresponding to the free drop listed above without subjecting the chassis to forces greater than those corresponding to the load factors in landing. The tail-skid shock absorber shall be designed to resist a free drop in the three-point landing attitude equal to that listed above without imposing loads on the skid or fuselage greater than those corresponding to the load factors for landing. If a shock absorber designed to dissipate the energy of the free drop by the flow of a liquid through an orifice—the oleo or oleo-pneumatic type of absorber—is used, the load factors required in the design of the landing gear may be reduced not to exceed 25 per cent. When this type of shock absorber is used suitable provision must be made to carry the shocks due to taxiing after the shock absorber has been forced to the full extent of its travel. The method to be used must be clearly shown as described on the chassis drawings. No reduction may be made on the required height of free drop for the shock absorber.

3. Loads on wings.—The beam and chord components of the loads on the wings for the high and low angles of attack and inverted-flight conditions shall be computed according to the rules given in this paragraph. The principal variations for use with structures varying from the conventional biplane are given in the succeeding paragraphs.

(*a*) The computations should be made in the following order:

I. Compute or estimate gross weight of airplane.
II. Compute effective span of each wing.
III. Compute or estimate dead weight of wings.
IV. Compute relative efficiency of the wings.
V. Compute normal gross beam load on each wing.
VI. Compute normal net beam load on each wing.
VII. Compute normal chord load on each wing.
VIII. Compute all other values required for plotting curves of beam and chord loads on each wing.

(*b*) The computations under (*a*) for the conventional type of design shall be made as follows:

I. The gross weight to be used in the calculations is the weight of the airplane with full pay load. The air load on the tail surfaces

should be neglected. It is recommended that for new designs, when the weights of items must be estimated, an allowance of 5 per cent be made to provide for unforeseen additions.

II. The effective span of a wing of constant chord and section is the distance from wing tip to wing tip, less that portion of the length, if any, that is covered by the fuselage and nacelles, and an allowance for tip loss. The "effective semispan" S_e, in Figures 4 and 5, is one-half of the effective span, L. The method of determining the decrease in effective span to allow for tip loss is shown in Figure 4 for externally braced and in Figure 5 for internally braced wings. For externally braced wings with excessive overhang refer to paragraph 3 (*b*).

III. The total dead weight W_d of the wings shall be estimated or found by weighing a cellule after construction. The dead weight per inch run, w_d, shall be determined by dividing W_d by the sum of the effective spans of the wings.

IV. The relative efficiency of the wings shall be determined from Figure 2. This figure gives the ratio of the normal unit load on the upper wing to that on the lower wing as a function of the gap-chord ratio and the stagger in degrees. In computing the efficiency with this chart, the gap-chord ratio and the stagger shall be measured on a vertical section midway between the side of the fuselage and the effective tip of the shorter wing. The same relative efficiency shall be used for all flying conditions.

V. The normal gross loads for each wing are to be obtained from the formulas:

$$w_{lg}=\frac{W_g}{eL_u+L_l} \text{ and } w_{ug}=ew_{lg}$$

Where W_g = gross weight of airplane in pounds.
e = efficiency of upper wing from Figure 2.
L_u = effective span of upper wing in inches.
L_l = effective span of lower wing in inches.
w_{lg} = normal gross beam load on lower wing in pounds per inch run.
w_{ug} = normal gross beam load on upper wing in pounds per inch run.

W_g may be taken as half the gross weight and L_u and L_l as the effective semispans S_{eu} and S_{el}, if desired.

VI. The normal net beam loads, w_u and w_l, are obtained by subtracting the dead weight of the wings, w_d, from w_{ug}, and w_{lg}, respectively. At this point it is advisable to check the results by computing the total net weight from the computed net loads per inch run.

VII. The normal chord load per inch run on each wing shall be taken as the proportion of the normal net beam load given in Table 2 in the high angle of attack, as 15 per cent of the normal net beam load in the low angle of attack condition, and as zero in inverted flight. In the high angle of attack condition this load shall be assumed to act toward the leading edge. In low angle of attack it shall be assumed to act toward the trailing edge.

VIII. (*a*) The chord load along the wing from the plane of symmetry to the wing tip is assumed to be constant and equal to the normal chord load—that is, it is assumed that there is no tip loss affecting the chord load.

(*b*) The distribution of the beam load along the span is shown by curves *A*, *B*, *C*, in Figures 4 and 5. Figure 4 applies to conventional, externally braced biplanes where the outer strut point is approximately one chord length from the wing tip and shall be used in obtaining the stresses in all members inboard of the outer strut point for such airplanes. In computing the design loads for those sections of the spars between the outer point of inflection of the outer bay and the wing tip the bending moments obtained from the loading represented by curves *A*, *B*, *C*, of Figure 4 shall be increased 30 per cent. The design loads in the outer struts and outer bay lift wires and that part of the axial loads and shears in the spars between the outer point of inflection and the outer strut point that is due to the loads in the lift truss shall be increased by 15 per cent. The axial loads in other parts of the lift truss need not be increased. For externally braced monoplanes or biplanes, where the outer strut point is at a distance materially greater than one chord length from the tip, the following method shall be used. For the investigation of stresses inboard of the outer strut point the load curve represented by *A*, *B*, *C*, of Figure 4 shall be used. The design loads in the outer struts or brace wires, the bending moments and shears in the spars, and that part of the axial loads in the spars that is due to loads in the lift truss shall, for the section between the outer point of inflection and the wing tip, be computed for the tip loss shown by part *A*, *B*, of the loading curve of Figure 5, point B being at a point one chord length from the tip. For internally braced wings the loading represented by the curve *A*, *B*, *C*, of Figure 5, shall be used, point *B*, for tapered wings or wings of constant chord, being at the section where the chord equals the distance to the wing tip.

The purpose of the rules in the above subparagraph is to obtain a structure having sufficient strength to withstand the variations in tip loss which pressure-distribution tests indicate to be liable to occur. These rules apply only to wings having square, circular, or elliptical tips and must not be used for raked tips where the end of the wing makes an angle of more than 15° with the plane of symmetry of the airplane. For such special conditions the designer should apply to the department for a special ruling before proceeding with the analysis of the wings.

4. Effect of cut-outs.—In computing the loads on wings with trailing edge cut-outs the following modifications shall be made in the method outlined in paragraph 3. The effective span of the wing with the cut-out shall be computed as before, except that an amount equal to the area of the cut-out divided by the wing chord shall be subtracted from the extreme span as well as the other quantities mentioned in paragraph 3. In this case the net beam load over the section between cabane struts will not be the normal net beam load, w_u, but will be equal to $\frac{l_2-l_1}{l_2}w_u$, where l_2 is the distance between the cabane struts and l_1 the area of the cut-out divided by the chord. When l_2 is zero on account of the center section struts meeting at the center of the upper wing, as is the case with some designs, no correction will be made in the curve of net beam load, though the correction to the effective length will be applied. Changes in wing section at

gas tanks and similar design features will be treated in a similar manner.

5. Effect of different chord lengths.—When an airplane has wings of different chord length, the modifications to the method of paragraph 3 are as follows:

(*a*) Two values must be computed for the dead weight per inch run w_d. These values can be obtained from the formulas.

$$w_{ud} = \frac{W_d}{L_u + \frac{C_1}{C_u} L_1} \text{ and } w_{1d} = \frac{C_1}{C_u} w_{ud}$$

Where W_d = dead weight of wings in pounds.
w_{ud} = dead weight of upper wing in pounds per inch run.
w_{1d} = dead weight of lower wing in pounds per inch run.
C_u = chord of upper wing.
C_1 = chord of lower wing.
L_u = effective span of upper wing in inches.
L_1 = effective span of lower wing in inches.

If the weights per square foot of the two wings are known to vary, this formula may be suitably modified.

(*b*) In computing the relative efficiencies of the wings the gap-chord ratio will be the ratio of the gap to the arithmetic mean of C_u and C_1.

(*c*) In computing the normal gross beam load on each wing the coefficient $\frac{C_u}{C_1}$ shall be inserted in the formulas of paragraph 3 (*b*) v, changing them to

$$w_{1g} = \frac{W_g}{\frac{C_u}{C_1} e L_u + L_1} \text{ and } w_{ug} = \frac{C_u}{C_1} e w_{1g}$$

(*d*) The normal net beam loads are obtained by subtracting w_{ud} and w_{1d} from w_{ug} and w_{1g}, respectively.

6. Effects of decalage or use of different airfoils for upper and lower wings.—If the wing chords are set at an angle to each other (decalage) or the upper and lower wings are of different airfoil section, the relative efficiency can not be obtained from Figure 2. Under such conditions satisfactory data must be submitted to the department to justify the distribution used.

7. Effects of ailerons.—In the design of the wing spars the load on an aileron used for computing the stresses on the wing structures shall be assumed the same as on a similarly located part of the wing—that is, no account need be taken of possible special loads due to the ailerons functioning as control surfaces, except in the design of the aileron and adjacent portions of the wing structure. Where the aileron projects beyond the tip of the wing proper, the following modifications shall be made in computing the loads on the wings:

(*a*) In computing the design loads on the front truss the projecting portion of the aileron may be neglected.

(*b*) In computing the design loads on the rear truss the projecting portion of the aileron may be neglected, except in computing the bending moments and shears in the rear spar outside of the outer

point of inflection of the outer bay. In that section of the rear spar the load on the projecting portion of the aileron shall be allowed for as follows:

I. The load on the projecting portion, Wa, will be found by multiplying the area of that portion A, B, C, D. Figure 6–A, by 0.8 times the normal net load per square foot on the wing.

II. The design shear on the section of the rear spar between the outer strut and the wing tip shall be that determined by the rules of paragraph 3 plus Wa. The shear inside of the outer strut shall be computed as outlined in paragraph 3.

III. The bending moment shall be increased by the amount shown by the curve G, E, F, of Figure 6–B.

IV. In computing these shears and bending moments the arbitrary increase called for in paragraph 3 shall be applied only to that part of the quantity due to the load on the portion of the structure inside of the tip of the wing proper. It will not be applied to the part due to the load on the projecting portion of the aileron.

8. Effect of taper in plan only.—In determining the loads on a wing tapered in plan only the procedure is similar to that for constant chord wings, except that areas are dealt with instead of lengths or spans. After the distribution per square foot of area has been determined, the results are multiplied by the appropriate chord lengths to find the values of load per inch of span. The detailed modifications in the procedure of paragraph 3 are as follows:

(*a*) An effective area is computed in place of the effective span of paragraph 3. This effective area is the area inboard of the effective tip, the location of which is determined by the methods indicated in Figures 3, 4, and 5. For externally braced wings the section DE, Figure 3, is located at the outer struts, the tip length L_t is represented by BF, Figure 3, and the effective tip is at 0.25 L_t from the wing tip. For internally braced wings the section DE, Figure 3, and B, Figure 5, are located so that the chord equals the distance to the extreme tip—that is, so that BF equals DE in Figure 3. The tip length, L_t, is represented by BF, Figure 3, and the effective tip is at 0.10 L_t from the wing tip.

(*b*) The dead weight of the wings is computed in pounds per square foot by dividing the total dead weight, W_d, by the total area of wing, A—that is, $w_d = \frac{W_d}{A}$.

(*c*) In computing the relative efficiency the gap-chord ratio and stagger shall be determined by the dimensions of the intersection of the airfoil sections with a plane parallel to the plane of symmetry and passing through the geometrical centroid of the effective area of the smaller wing.

(*d*) The formulas for the normal gross beam load are modified as follows:

$$w_{1g} = \frac{W_g}{eA_u + A_1} \text{ and } w_{ug} = ew_{1g}$$

Where A_u and A_1 are the effective areas in square feet of the upper and lower wings, respectively, and w_{1g} and w_{ug} are expressed in pounds per square foot.

(*e*) The normal chord load and the normal net beam load are computed as for a constant chord wing, except that quantities are in pounds per square foot instead of pounds per inch run.

(*f*) The ordinates to the curves of load distribution are found as designated in paragraph 3. Figures 4 and 5 may be considered as applying to the tapered wing except that the values are in pounds per square foot. Additional curves should then be drawn to show the distribution in pounds per inch run. These curves are found by multiplying each ordinate of the preceding curves by the chord in inches and dividing by 144. If the leading and trailing edges are straight lines, the curves of net beam loading over the tip section will be slightly curved. It is permissible in this case to compute the ordinates of load per inch run at each end of the tip section and assume the included portion of the loading curves to be straight lines.

9. Effect of taper in thickness ratio.—When the thickness ratio of a wing varies, it is assumed that the air forces and the dead weight of a wing per square foot vary in proportion. The loading curves for such a wing are determined by computing the dimensions of an "equivalent wing" assumed to be of constant thickness ratio and finding the loading curves for the equivalent wing by the rules given above. The span of the equivalent wing is assumed to be the same as that of the true wing, but the chord at any section is the chord of the true wing at the section multiplied by the thickness ratio at the section and divided by the thickness ratio at the section of maximum chord. If the true wing has parallel leading and trailing edges and the thickness ratio has a straight-line variation (the usual case), the chord of the equivalent wing will also have a straight-line variation.

In computing the relative efficiency the gap-chord ratio and the stagger shall be determined by the dimensions of the intersections of the actual wings (not the equivalent wings) with a vertical plane parallel to the plane of symmetry and passing through the geometrical centroid of the effective area of the smaller actual wing.

In certain cases, such as those of metal or plywood covered wings, where the weight of the covering forms a large percentage of the total weight of the wings, the department may require that the dead weight of the wings be assumed to be distributed uniformly over the actual area of the wing instead of being varied in proportion to the thickness ratio.

10. If the true wing is tapered in plan or the thickness ratio has not a straight-line variation, the chord of the equivalent wing will not have a straight-line variation. In such a case the equivalent wing should be divided into parts, in each of which a straight-line variation of the chord and loading curves may be assumed. On an externally braced wing the dividing lines between these parts should be at the interplane strut location. On internally braced wings these dividing lines shall be at the sections where the chord of the actual wing equals the distance to the wing tip, the section where the wing is connected to the fuselage, and a reasonable number of intermediate sections.

11. Effect of several variables.—Where the design of a wing incorporates more than one of the features discussed in paragraphs 4 to 10, the computation of the loads shall include all the modifica-

tions needed to take care of the special case. Where a designer is uncertain as to the method of applying these rules to an airplane being designed by him, he should request the department for a special ruling. The request should be accompanied by drawings showing the relative location of the wings, their dimensions, interplane bracing, data as to the airfoil section used, the total weight of the airplane, the dead weight of the wings, and any other pertinent data that may be available.

12. Application of beam loads to spars.—With two-spar construction the beam load on each element of the wing in the high and low angle of attack and inverted-flight conditions shall be assumed to act at the center of pressure and be divided between the spars in inverse proportion to the distances from the center of pressure location to the spars. For wings having more than two spars the load shall be distributed among the spars by a method approved by the department. Table 2 gives the center of pressure location to be used for several airfoil sections and is based on its most forward location for high angle of attack and inverted flight and on its location when

$$ky = 0.25\ ky\ \text{max.}$$

for low angle of attack. These values shall be used for all speed ranges.

TABLE 2.—*Center of pressure location and ratio of chord to beam components for various airfoils*

Airfoil	C. P. location in per cent of chord		Ratio of chord component to beam component in H. A. A.
	H. A. A. and I. F.	L. A. A.	
Aeromarine 2A	29	48	−0.178
Boeing 103A	28	45	−.158
Clark Y	31	51	−.144
Curtiss C-62	28	40	−.056
Curtiss C-72	31	54	−.130
Göttingen 387	31	52	−.156
Göttingen 398	30	50	−.167
Göttingen 436	31	50	−.135
Loening 10A	28	45	−.166
M-6	[1] 25	[2] 40	−.178
M-12	[1] 25	[2] 40	−.140
N-9	28	45	−.132
N-10	29	48	−.150
N-22	28	48	−.155
R. A. F. 15	29	38	−.127
Sloane 105	30	48	−.078
U. S. A. 27	29	50	−.175
U. S. A. 35A	33	56	−.126
U. S. A. 35B	30	49	−.151
U. S. A. 45	27	38	−.191

[1] Based on C. P. at *ky* max.
[2] Based on C. P. with ailerons or flaps set at an angle of 25°.

The center of pressure location for "stable" airfoils in the low angle of attack condition should be obtained from wind-tunnel tests made with ailerons or flaps set at 20 or 25°. If such data are not available, a location between 45 and 50 per cent of the chord should be assumed.

The ratio of net chord component to net beam component will be assumed as 0.15 in low angle of attack condition and zero in inverted flight for all airfoils. In high angle of attack condition it is obtained from the formula:

$$C/B = \tan(\cot^{-1} L/D - i)$$

Where C/B = the ratio of the chord component to the beam component.

i = angle of attack when the center of pressure is at its most forward location.

The lift and beam components of the air load are assumed to be equal, as the error involved is negligible.

13. Application of chord loads to internal wing truss.—The beam loads are to be assumed to act on the spars as distributed loads, although in practice they are applied as small concentrated loads at the ribs. The chord loads shall be considered as concentrated loads at the panel points of the internal wing trusses. The load at each such joint shall be the product of the normal chord load per inch run at the joint and the distance between mid-points of the adjacent panels of the internal truss. For this purpose the extreme wing tip shall be considered the mid-point of a truss panel.

14. The nose-dive condition.—In the nose-dive condition the beam loads on the front spars shall be the same as the design ultimate loads on those spars in the inverted-flight condition. By taking moments about the rear spars it is possible to determine the load required on the tail surfaces, assumed to be applied at the tail post, to maintain equilibrium. Since the sum of the forces parallel to the beam direction must be zero, if the airplane is to be in equilibrium, the total upward load on the rear spars will equal the sum of the downward loads on the front spars and tail surfaces. This load is distributed in the usual manner. The chord components of the air load on the wings shall be proportional to the chord loads in the low angle of attack condition, but their total magnitude shall equal the gross weight of the airplane. These chord loads may be assumed to be held in equilibrium by loads acting on the fuselage structure at panel points near the center of gravity. The loads on the wing framework shall be assumed to be carried by the incidence bracing, interplane struts, rear flying wires, and drag truss members only, unless a least-work analysis is made.

Internally braced, fabric-covered, cantilever monoplanes shall have a double system of drag trussing, spaced as far apart as possible.

The drag wires or their equivalent shall have a margin of safety of 200 per cent, the load being assumed to be divided equally between the two systems. The design ultimate loads on the counter wires shall be the same as for the drag wires.

Drag ribs shall be designed to withstand the maximum shear 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 ultimate design loads, the remaining wires in these two bays being assumed to be out of action.

Plywood and metal covered wings shall have equivalent strength. This nose-dive condition shall be investigated regardless of the airfoil used or the type of airplane being designed.

15. Dead loads carried by the wings.—Where dead weights are carried by the wings, such as landing lights, wing-tip floats or skids, gas tanks, etc., their effect shall be allowed for as follows: If the weight is small and in such a location that it has little effect on the computed stresses, it may be included in W_d, the total dead weight of the wing. Loads that are large or are so placed that they would cause an appreciable change in the computed stress system shall be separately provided for. If the load is the weight and inertia of a compact unit, it shall be treated as a concentrated load on each spar, the relation between these loads depending on the location of the center of gravity of the unit. If the unit is spread over a considerable extent of wing area, like a gas tank or a wing radiator, it shall be considered either as composed of concentrated loads at the points where it is supported by the spars, or as a uniformly distributed load, depending on the manner of support. If a dead weight is supported in such a manner that it can be considered as a uniform load and extends over two-thirds or more of a panel of the lift truss, it may be assumed as distributed over the entire panel. When the unit is one of variable weight, as a fuel tank, the primary analysis shall be made, using the maximum weight. An investigation using the minimum weight shall also be started and carried far enough to determine the design of any member limited by this condition or to show clearly that there are no such members. The weights of all such loads shall be assumed to act in the beam direction in the high and low angle of attack and inverted-flight conditions, and in the chord direction in the nose dive. The effect of nacelles of multi-engine airplanes and their contents shall be computed according to the same basic principles, the application of which to the special case is outlined more fully in paragraphs 41 and 42.

16. Internally braced biplanes shall be designed by a method acceptable to the department; that described in Air Corps Information Circular (Aviation) No. 440 is recommended.

17. Loads on control surfaces.—The average ultimate loads to be used for the design of control surfaces are given in Table 1. The loads on the fixed surfaces are to be assumed uniformly distributed, but on movable surfaces the intensity of loading at the hinges shall be equal to the loading on the fixed surfaces in front of them and shall decrease uniformly to an intensity at the trailing edge of one-third the former value. Portions of the movable surface in front of the hinge line, except auxiliary vanes, shall carry the same loading as the fixed surface.

(*a*) When auxiliary vanes are used for balancing, they shall be assumed to be subjected to the same intensity of loading as the fixed surfaces when computing the distribution of load and the stresses in the remainder of the movable surface. The vanes themselves and their attachment to the main movable surface shall be strong enough to carry the load required to balance the load on the main portion of the movable surface.

(*b*) The control surfaces must be designed to carry the specified load acting in either direction.

(*c*) Although no load parallel to the chord of the fixed tail surfaces or torsional load is specified, provisions shall be made to carry a reasonable amount of such load.

18. Loads from control system.—The forces in the control wires or push rods operating the movable surfaces must be computed and their effect on the rest of the structure investigated and allowed for in the design of the structure. Special care must be taken to make sure that the portion of the wing between the rear spar and the aileron is strong enough to carry the load for which the aileron is designed, and the structural members in this part of the wing shall be analyzed in the aileron analysis.

19. Landing gears—Level landing.—The resultant of the ground reaction is 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. The propeller axis is assumed horizontal and the basic value of the vertical component of the resultant of the ground reaction equal to the gross weight of the airplane minus chassis and wheels. The horizontal component is of the magnitude required to give the resultant force the specified direction. The resultant of the ground reaction is assumed to be divided equally between wheels and to be applied at the axles. Landing gears provided with shock absorbers which subject the structure to increased loads with increased deflection of the shock absorber—the rubber-cord or rubber-disk types—shall be designed for the fully deflected position of the shock absorber unless a partial deflection of not less than one-half the maximum would cause greater stresses. Shock absorbers which subject the landing gear to approximately constant loads throughout their travel—the oleo type—shall be assumed to be in their undeflected position unless the arrangement of the structure is such that a partial deflection of the shock absorber will result in higher loads in some or all of the members.

20. Landing with side load.—The landing gear and cabane and that part of the fuselage adjacent to them should be analyzed under the following combination of loads:

(*a*) The loads specified in paragraph 19 for the level-landing condition.

(*b*) Side components of the ground reactions equal to one-fourth of their vertical components. These side components of the ground reactions shall be assumed to be equally divided between wheels and to act at the ground, the tire being assumed to have deflected one-half the nominal diameter of its cross section.

(*c*) Side components of the weight and inertia forces equal to one-fourth of their vertical components. These side components may be assumed to act on the fuselage joints to which the wings and chassis are attached and to be divided between joints in such a manner that their resultant would pass through the center of gravity of the airplane.

(*d*) A couple composed of vertical forces acting at the main connections of the wings to the fuselage and cabane and of sufficient magnitude to bring the entire system into equilibrium.

The required load factor will be one-half that specified in landing conditions, Table 1.

In the case of the formerly conventional V type of chassis the spreader tubes and diagonal brace wires may be designed to carry a load of 1.5 times the weight of the airplane acting in either direction

and applied to one wheel at its axle, in place of the loading specified above.

21. Three-point landing.—The basic value of the sum of the ground reactions is the gross weight of the airplane. The total load is divided between the chassis and tail skid in inverse proportion to the distances, measured parallel to the ground line, from the center of gravity of the airplane to the points of contact with the ground. The net basic load on the chassis shall be equal to the total load on the chassis minus the chassis weight, and it shall be divided equally between wheels. Loads are assumed to be perpendicular to the ground line in the three-point landing attitude, with all shock absorbers extended to the same degree as in the level-landing condition.

22. Reactions.—In all loading conditions for the landing gear the reactions of the fuselage or wings are the forces required to balance the net loads on the axles. The equilibrants of these reactions will be used in the analysis of the fuselage in the landing conditions.

23. Braked landing.—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 just clears the ground. The full weight of the airplane will 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 times a coefficient of friction of 0.55, or equal to the load required to keep the airplane in equilibrium with the tail skid just clear of the ground, whichever is the less. The chassis structure shall have sufficient strength to withstand the axial loads, bending, and torsion to which it will be subjected under this combination of loads, the load factor being one-half that specified in Table 1 for the landing conditions, with no reduction permissible because of the use of oleo shock absorbers.

24. Seaplane landing with inclined reactions.—Assume propeller axis horizontal and the resultant water reaction acting in the plane of symmetry and passing through the center of gravity of the airplane 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 act in a direction parallel to the water reaction. The weight of the floats and float bracing should be deducted from the gross weight of the airplane. A load factor of 8 is required. For water-landing conditions flying boats and amphibians shall be considered as seaplanes.

25. Seaplane landing with vertical reactions.—Assume propeller axis horizontal and the resultant water reaction vertical and passing through the center of gravity. The weight of the floats and float bracing should be deducted from the gross weight of the airplane. A load factor of 8 is required on the float bracing and the members in the fuselage which are subjected to direct loads from the float-brace attachments.

Flying-boat and seaplane-float hulls shall be designed to withstand without permanent set a load of 8 pounds per square inch over that portion of the hull lying between the first step and a section 25 per cent of the distance between the step and the bow. A load of

4 pounds per square inch shall be assumed from that section to a section at 75 per cent of the distance between the step and the bow In designing the bow portion of the hull due attention must be paid to the effect of striking floating objects. For the section between the first and second steps a load of 4 pounds per square inch shall be used. If but one step is used, this load shall extend over that portion of the hull between the step and a section 50 per cent of the length of the hull aft of the step.

26. Seaplane landing with side load.—Assume the propeller axis horizontal and the resultant water reaction in the vertical plane which passes through the center of gravity and is perpendicular to the propeller axis. A load factor of 8 vertically and 2 horizontally is required, to be applied to the gross weight of the airplane less weight of floats and float bracing. The vertical load shall be applied through the keel or keels of the float or floats, evenly divided between the floats when twin floats are used. The horizontal load shall be applied along a line approximately halfway 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 comes.

27. Fuselages.—The fuselage shall be investigated for seven loading conditions—high angle of attack, low angle of attack, maximum stabilizer and elevator load, maximum fin and rudder load, level landing, three-point landing, and nosing over. The entire fuselage structure will not have to be analyzed for each of these conditions, but all must be considered and each stress investigation carried far enough to insure that the loads in all members limited by that condition are computed. Certain additional stress investigations will also be needed where eccentricities or bending loads occur and the results used in connection with the standard loading conditions in the final design of the member affected. By careful planning it will be possible to obtain the maximum loads in all members with a minimum of work, and in the description of the various loading conditions this phase of the problem will be considered. Usually the analysis of the fuselage should not be made until after those of wings and landing gear. When it is not desired to carry out the work in this order, the fuselage analysis should be corrected in accordance with the results of the other analyses.

28. Table of panel loads.—

TABLE 3

Unit	Weight	Vertical arm	Vertical moment	Horizontal arm	Panel points 1, 2, 3, 4, etc.; a, b, c, d, e	Horizontal moment
Propeller						
Engine						
Gas tank						
Etc.						
Center of gravity						
Totals						

In the horizontal line beginning "Horizontal arm" fill in the distance from the datum point to the panel points in question in the spaces occupied in the sample table by *a*, *b*, *c*, etc.

In the vertical column under "Horizontal arm" fill in the horizontal distance from the datum point to the center of gravity of the unit considered.

In the column headed "Vertical arm" fill in the corresponding vertical distance.

Any point may be used as the datum, but the point selected should be clearly stated. The front face of the rear propeller flange will usually be found convenient and is recommended.

In the column under the heading "Panel points" should be inserted the portion of the weight of each unit carried at each point. The horizontal totals will be the weights of the units, and the vertical totals, the panel loads.

All weights should be given, including wings and chassis, but the panel loads should be modified when certain weights are omitted in the fuselage analysis, as, for example, the weight of the wings in the high angle of attack condition.

Panel points may be numbered either front to rear, or vice versa.

As many horizontal and vertical lines should be used as are necessary to represent the design in question, the form shown being only a guide.

In computing the panel points loads to be tabulated, as indicated in Table 3, the following rules shall be observed:

(*a*) Items located between two adjacent panel points of the side trusses shall be divided between those panel points in inverse proportions to the distances from them to the center of gravity of the item.

(*b*) Items to the rear of the tail post shall be represented in the table by a load and a horizontal couple at the tail post.

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

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

(*e*) If desired, the loads may be divided between the upper and lower panel points in proportions determined by the vertical location of the units they represent. All loads are assumed to lie in the plane of symmetry and to be divided equally between the two vertical trusses of the fuselage.

29. Panel point loads in three-point landing.—A separate table of panel point loads should be computed for the three-point landing condition unless the loads are carefully divided between upper and lower longerons, as suggested in the preceding paragraph. The labor involved can be greatly decreased by grouping items that are supported by the same panel points and treating the group as a unit. The distance of the centroid of the group from the origin measured parallel to the ground can be found from the formula:

$$x' = \frac{\Sigma M_h \cos\theta + \Sigma M_v \sin\theta}{\Sigma W}$$

Where x'=distance from the origin to centroid, measured parallel to the ground line.
ΣM_h=sum of horizontal moments of units in group from balance schedule.
ΣM_v=sum of vertical moments of units in group from balance schedule.
ΣW=sum of weights of units in group from balance schedule.
θ=angle between ground line and propeller axis.

The distance measured parallel to the ground line from the origin to the panel points can be scaled from the drawing or computed from the usual formula for the rotation of axes.

$$x' = x \cos \theta + y \sin \theta$$

Where x'=distance measured parallel to ground line.
x=distance measured parallel to propeller axis.
y=distance measured normal to propeller axis.
θ=angle between propeller axis and ground line.

y is positive if the point is above the propeller axis and negative if it is below.

Whichever method is used to determine the panel loads for the three-point landing condition. care must be taken that the equations of equilibrium are satisfied by the panel loads and reactions used.

30. Low angle of attack condition.—In the low angle of attack condition the fuselage shall be assumed to be subjected to the following loads:

(*a*) Weight of fuselage and contents multiplied by the low angle of attack load factor. These weights are assumed to act normal to the propeller axis.

(*b*) A load at the tail post, or on the horizontal tail surfaces perpendicular to the propeller axis, and of the magnitude required to obtain equilibrium.

(*c*) Loads from the wings equal to the forces supporting the wings in the low angle of attack condition multiplied by a factor sufficient to make the sum of the forces perpendicular to the propeller axis equal to zero.

(*d*) A load along the propeller axis sufficient to make the sum of the forces parallel to that axis equal to zero. The loads and factors under (*b*), (*c*), and (*d*) are to be found by applying the equations of equilibrium to the entire structure. The entire fuselage side truss and some crossties will usually have to be analyzed for this loading condition.

31. Three-point landing.—In the three-point landing condition the fuselage shall be assumed to be subjected to the following loads:

(*a*) Weight of the entire airplane. excepting only such units as are carried by the landing gear without the aid of the fuselage. These loads are assumed to act perpendicular to the ground line. The load factor for this condition is specified in Table 1.

(*b*) The landing-gear reaction, which, as determined from the landing-gear analysis for this condition with no reduction for the use of oleo shock absorbers, should balance the weights. The entire fuselage side truss will usually have to be analyzed for this condition.

32. Maximum stabilizer and elevator load.—The fuselage is to be analyzed under the maximum loads on the horizontal tail surfaces specified in Table 1 and the required supporting forces at the wing connections. The load on the tail is assumed to act down. Usually it will be found that at some section to the rear of the connection to the wings the loads in the fuselage members in this condition are less than in low incidence. The analysis for this loading need not be carried forward of that section. In designs without tail skids, as seaplanes, a similar loading condition must be investigated in which a load equal to half that specified in Table 1 is assumed to act upward on the stabilizer and elevator.

33. Maximum fin and rudder load.—The fuselage is to be analyzed under the maximum loads on the vertical tail surfaces, specified in Table 1, and the required supporting forces at the wing connections. These loads must be assumed to act in either direction. In analyzing for this condition proper allowance must be made for the torsion due to the fact that the center of pressure on the vertical surfaces is above the fuselage, where that is the case. On many airplanes this can best be done by applying a load greater than the tail load on the upper truss and a load in the opposite direction on the lower, the resultants of these two loads being equal to the resultant load on the tail. On small airplanes even this conservative method will indicate sizes smaller than could be allowed from other practical considerations. Where this method is considered too conservative, the method of least work or some similar method of allowing for the torsion may be used. In all cases proper account shall be taken of cockpit openings and similar places where the trussing is not continuous. This condition will usually determine the design of the top and bottom trusses of the fuselage, except in the forward portion, where they are affected by the wing and landing-gear loads.

34. Level landing.—In the level-landing condition the fuselage shall be assumed to be subjected to the following loads:

(*a*) Weight of the airplane, except the chassis, assumed to act normal to the propeller axis.

(*b*) The landing-gear reactions from the level-landing chassis analysis with no reduction for the use of oleo shock absorbers. The vertical component of these reactions should balance the weights.

(*c*) Horizontal loads representing the inertia forces due to slowing down of the airplane. As the rear portion of the fuselage will not be limited by this loading condition, the following procedure will be found convenient.

(*d*) Compute the loads in some panel to the rear of the wing connections by the method of sections, or its equivalent, and apply these loads as external forces at the forward end of the panel. In making this computation consider only the weights of fuselage and contents to the rear of the section investigated.

(*e*) Divide the load under (*c*) into a load on each longeron at the forward end of the panel mentioned in (*d*) in such manner that the front part of the fuselage truss will be in equilibrium under the external forces. This condition will usually be critical only for members in the forward part of the fuselage. If desired, the panel loads may be assumed to act parallel to the resultant of the landing-gear reactions.

35. High angle of attack.—In the high angle of attack condition the fuselage shall be assumed to be subjected to the following loads:

(*a*) Weight of fuselage and contents multiplied by the high angle of attack load factor. These forces are assumed to act normal to the propeller axis.

(*b*) The high angle of attack wing reactions multiplied by the factor required for equilibrium.

(*c*) A load on the nose of the fuselage, acting along the propeller axis, and of the magnitude required by the conditions of equilibrium. This load may be considered to represent the resultant of the propeller thrust, drag of the fuselage, and the components of the weight and inertia forces of the fuselage and contents parallel to the propeller axis.

(*d*) An arbitrary load at the tail post, acting normal to the propeller axis, of the magnitude and acting in the direction required by the conditions of equilibrium.

(*e*) The torque load treated in the manner described in paragraph 37. In order to determine the factor required for load (*b*), and the magnitude of loads (*c*) and (*d*), it will usually be necessary to write down and solve the equations of equilibrium for the entire structure in a manner similar to that suggested in paragraph 30 for computing the loads in the low angle of attack.

36. Nosing over.—The front part of the fuselage shall be designed to resist the forces to which it would be subjected in nosing over unless the landing gear is of such construction that the probability of such an accident is remote. To simulate this condition, assume the airplane to be resting on the wheels, with the tires deflected through a distance equal to one-half the tire diameter, and the center of the propeller hub or that portion of the primary structure of the fuselage that would strike the ground first. Assume the gross weight of the airplane to act at the center of gravity and perpendicular to the ground. The load factor required shall be that for the three-point landing condition.

37. Computation of torque load.—The engine torque is equal to:

$$T = 63{,}000\ P/n$$

where T = torque in inch pounds

P = horsepower of engine

n = propeller speed in revolutions per minute.

The torque load on each engine bearer is T/d where d is the distance in inches between engine bearers. This load acts down on one engine bearer and up on the other.

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

of symmetry. In case of direct-drive engines the stresses due to torque load and its equilibrants shall be multiplied by a factor of safety of 2 and added to the stresses found for the high angle of attack condition; in the case of geared engines the factor of safety to be used shall be 3. The stresses are to be added arithmetically and not algebraically, because if the forces induced by the torque load in any member are opposite in sign 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 torque loads and dead weights will be of the same sign.

When a direct-drive engine is carried by engine bearers that are supported at two or more points the torque load shall be divided between the points of support in the same proportions as the weights carried by the engine bearer. When a direct-drive engine is supported by a vertical plate the torque shall be assumed to act at the center of gravity of the engine. In the case of a geared engine the torque shall be assumed to act at the gear box.

38. Special conditions.—In addition to the standard loadings described above, which give only the axial loads in the side trusses, the following points must be investigated and computations made if necessary:

(*a*) If units situated between panel points or loads from other major assemblies are carried by bending in members of the main trussing, numerical estimates of the amount of such bending must be made, and, if serious, this bending must be allowed for in the design of the affected members.

(*b*) The strength of the fuselage members, not part of the main trussing but transmitting loads to it, must be investigated. In general, these members shall be designed for the same load factor as the part of the main fuselage structure to which they are connected.

(*c*) Care should be taken that the fuselage is strong enough to carry the loads from the chassis in the level-landing condition with side load or in a one-wheel landing condition. The loads from the chassis in these cases are unsymmetrical, and the bulkheads where the fuselage and chassis are connected should be strong enough to equalize them between the fuselage trusses.

(*d*) The engine-mount section of the fuselage shall be capable of carrying the weight of its contents, acting sideways, with an ultimate load factor equal to one-half of the high angle of attack factor.

(*e*) Wherever there is an eccentricity in the fuselage trussing the stress diagram shall be drawn for a theoretical truss in which the eccentricity has been eliminated. In designing the members, however, the bending due to the eccentricity shall be computed and used in determining the design sizes.

(*f*) The forward portion of the fuselage shall be analyzed for the condition of inverted flight in all cases where it is not necessary to analyze this part of the structure for the nosing-over condition.

(*g*) *Design of new engine mount for existing airplane.*—When a new engine mount is to be designed for an existing airplane and it is not necessary, because of the margins of safety in the existing structure, to make an entire new fuselage analysis, it shall be designed for the following conditions:

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I. High angle of attack and torque: The analysis shall be made as specified for the fuselage for those conditions, using load factors based on the new power loading if any change is made, except that the reactions required may be assumed to be applied at the points where the new structure is joined to the structure of the original design.

II. Inverted flight: The load in each member shall be assumed to be of opposite sign to the load in that member, due to high angle of attack and torque, and its magnitude equal to 75 per cent of that load. This condition will be assumed to cover the condition of nosing over as well as inverted flight, the high ratio of reversal of load being used so this will be possible.

III. Side load: This condition shall be analyzed as specified above in subparagraph (*d*).

IV. In designs for which the design load factor for the landing conditions is greater than that for high angle of attack, the landing factor shall be used in place of the high angle of attack factor in the analysis for the condition of high angle of attack and torque.

(*h*) When the point of contact of the tail skid with the ground is in front of the tail post a distance greater than 1.5 times the vertical distance between longerons at the tail-skid station, the rear portion of the fuselage shall be investigated for a special three-point landing condition in which the basic load on the tail skid shall be assumed to act at the lower end of the tail post. The load factor for this condition shall be that for the landing conditions minus 1.

39. Cabane.—In the inverted-flight and nose-dive conditions the cabane should be considered a part of the wing structure and analyzed with it. In the high and low angle of attack conditions it will be simpler, in general, to analyze it as part of the fuselage structure. When there is some doubt as to how the loads are divided between cabane members, all possibilities should be considered and the analysis made according to assumptions that will be conservative for both cabane and fuselage. This may require more than one analysis of the cabane and forward section of the fuselage in either high or low angle of attack. The most usual case where there will be doubt regarding the action of the cabane is when the front spar is attached to the fuselage by a tripod of struts. In this case the cabane should be investigated by the method of least work.

In order to allow for unsymmetrical loads on the wings, as in a roll, the cabane shall be investigated for the stresses that would be present in the high and low angle of attack conditions if 75 per cent of the design load were acting on one wing and 50 per cent of the design load on the other.

40. Nacelles.—The loads on a nacelle must be considered from two standpoints—that of determining the design of the nacelle itself and that of determining the design of the internacelle wing structure. For the design of the nacelle itself the following conditions must be considered:

(*a*) Weight of nacelle and contents acting in beam direction multiplied by high angle of attack load factor, torque load multiplied by a load factor of 2 for direct drive, 3 for geared drive, and reaction from wing structure.

(*b*) The weight of nacelle and contents acting sideways multiplied by one-half the high angle of attack load factor and reaction from the wing structure.

(*c*) Condition (*a*) with the direction of loads reversed and the inverted-flight load factor. The torque load may be decreased proportionately to the other forces.

41. Effect of nacelles on beam forces.—In the wing analyses, for high and low angle of attack and inverted flight, the weight of the nacelle and contents must be treated as a beam load on the wing structure. The division of this load between points where the nacelle is supported will be obtained from a nacelle analysis. Usually it will be simplest to make the nacelle analysis in such a way that the effect of the torque and variable weights can be separated from the effect of the fixed weights. The nacelle load on each truss can then be considered to act as a single load at the nacelle center line or as equal loads at the sides of the nacelle, depending on the local construction. Two sets of conditions must be investigated—one with both fixed and variable weights and the other with only the fixed weights considered. The torque may be neglected in figuring the effect of the nacelle in the wing structure, as its effect is simply to shift the center of gravity of the nacelle slightly.

42. Effect of nacelles on chord forces.—In designs with nacelles the following rules should be used for computing the loads in the internal or drag trusses:

(*a*) The internal wing trusses shall be assumed to be supported at the cabane and fuselage in all flying conditions.

(*b*) The head resistance of the nacelles shall be represented, in each of the loading conditions, by chord loads equal to 0.4 of the chord components of the total air load on the wings, divided equally between nacelles and acting along the propeller axes. These loads will be assumed to act backward.

(*c*) The chord components of the weight plus inertia of the nacelles and contents shall be computed by multiplying the weight of the nacelles and contents by the load factor and the ratio of the chord component to the beam component of the air load on the wings, in the high and low angle of attack conditions, and by the safety factor 2 in the nose-dive condition. These forces shall be assumed to act at the propeller axes and in the direction opposite to the chord components of the air load.

(*d*) When nacelles are placed on the wings of monoplanes, the same rules will apply, except that in the nose-dive condition all loads will be increased as specified in paragraph 14.

(*e*) Rule (*b*) is based upon the assumption that the total drag of the airplane is produced 50 per cent by the wings, 30 per cent by the fuselage, and 20 per cent by the nacelles. Whenever sufficient data exist for a particular design to justify a different assumption as to the division of the total drag, a special ruling should be requested. Rule (*c*) is based upon the assumption that the weight and inertia of the wings act in a direction directly opposite to the resultant gross air load.

43. Effect of nacelles on fuselage analysis.—The presence of nacelles in a design will affect the analysis of the fuselage principally through changing the location of the power plant and the amount

of the forces from the wings. In order to balance the force system, it will usually be necessary to apply arbitrary forces at some of the wing connections, and the amounts of such forces should be definitely stated. In the landing condition, allowance should be made for the way in which the weights of wings, nacelles, etc., are applied, if at all, to the fuselage structure.

44. Effect of landing gear attached to the wings.—When landing gears are attached directly to the wings, that part of the wing structure between the outermost landing-gear members must be considered as a part of the fuselage in the landing conditions.

PART 3.—RECOMMENDED PRACTICE

The following practices in airplane construction have been developed over a long period of years. They are standard in character and are considered essential in the development and production of approved type aircraft. Deviation from them may be made if and when it is demonstrated that such deviation is productive of the same or better results.

1. Wings.—(*a*) Fittings shall be designed to carry loads 20 per cent in excess of the design loads for the members to which they are connected. Since, in small wires, the initial tensions due to rigging may impose a considerable load on the fittings, it is required that wires and cables up to and including the 3,400-pound size shall be attached to lugs having the same rated strength as the wires or cables. (See fig. 9 for sizes of standard lugs.) Heat-treated or alloy-steel fittings should be used with caution because of the element of danger involved in making field repairs.

(*b*) When streamline wires are used for lift wires, they shall be double unless the wings are so designed that with any lift wire removed the strength of the remaining wing structure will be adequate to develop load factors of not less than 50 per cent of those specified in Figure 1. When double wires are used each shall be capable of carrying 65 per cent of the design load. Loop terminals will be assumed to have 85 per cent of the rated wire strength.

(*c*) The maximum wing rib spacing permitted with fabric covering is given in Table 4. This spacing is determined by the necessity of preventing the covering from deflecting to such an extent that the properties of the wing will be affected by the change in the effective airfoil section. Usually at least one intermediate false rib will be required in the nose portion to keep the surface free from irregularities.

TABLE 4.—*Maximum rib spacing, in inches*

High speed in miles per hour	In slip stream	Outside of slip stream	High speed in miles per hour	In slip stream	Outside of slip stream
100 [1]	14	18	160	8	11.5
120	12	16	180	6	8.5
140	10	14			

[1] For speeds less than 100 miles per hour use the figures for 100 miles per hour.

(*d*) Thin plywood or metal covering should be used on both surfaces of each wing within the slip stream, between the leading edge and front spar. On all airplanes whose high speed is greater than

100 miles per hour it is advisable to extend this covering along the entire leading edge.

(*e*) In the design of wing spars and other members subjected to combined axial and transverse loads proper allowance must be made for secondary stresses. The Berry method given in Aeroplane Structures by Pippard and Pritchard, the method given in Graphische Statik by Muller Breslau, or the Army Air Corps "precise" method, given in Airplane Design by A. S. Niles and in Information Circular 493, are acceptable. Of the three, the last is preferred.

(*f*) In determining the allowable stresses in metal members subjected to combined bending and compression the allowable stress will be assumed to lie between that which the member would carry as a column and that which it would carry as a beam. The exact value will be determined from the ratio of stress due to bending to the total stress; that is, if the stress due to bending is three-fourths of the total stress, the allowable stress under combined loading will be equal to the stress that the member would carry as a column plus three-fourths of the difference between its modulus of rupture in bending and its strength as a column. The allowable stress in wooden members shall be determined from Figure 7.

(*g*) Tests on I and box spars have shown that a decrease in the web thickness results in a serious decrease in the modulus of rupture. For purposes of design, the modulus of rupture of such spars shall be that obtained from Figure 7. In addition to decreasing the modulus of rupture, thin webs result in increased deflections and secondary stresses. The study of the problem has not yet progressed to a point where it is possible to state the most efficient web thickness for a given spar. In the case of solid spars the web should be at least ½ inch thick, except on very small airplanes, and it will often be advantageous to use a solid rectangular section instead of an I. Less is known about the proper web thickness for box spars, but ⅛ inch is considered a good minimum thickness for each web. The web may be made of either two-ply or three-ply material, but in either case the grain of the individual plies should make an angle of 45° with the axis of the spar. Where box spars are used, it is desirable to check the proportions of the design by constructing a short section of spar and testing it to the required shear and bending moment.

The allowable total unit stress in spruce members subjected to combined bending and compression shall be computed by the method developed by the Forest Products Laboratory, through the aid of Figure 7. On the right-hand side of this figure are two families of curves—the upper one for the determination of the modulus of rupture, the lower one for the elastic limit in bending. Each of these quantities is dependent on the ratio of compression flange thickness to total depth of beam and the ratio of web thickness to total width of beam. On the left-hand side of the figure are two additional families of curves. The "horizontal" family indicates the elastic limit under combined bending and compression, and the "vertical" family the effect of various slenderness ratios on the former quantity.

The allowable total stress under combined load, F_t, is found as follows:

1. For the cross section of the given beam, find the elastic limit in bending and the modulus of rupture from the ratios of compression

flange thickness to total depth, and web thickness to total width, locating points such as A and B.

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

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

IV. Draw ED.

V. Locate F on ED, with an abscissa equal to the computed ratio of bending to total stress. The ordinate of F represents the desired value of F_t.

The following rules should be observed in the use of Figure 7:

I. The slenderness ratio should be that between points of inflection. When the ends of a member are restrained, this need not be computed with excessive precision, as a small error in L/ρ will not result in a large error in F_t. For wing spars, the following approximations are acceptable:

In computing the margin of safety L may be taken as the distance between points of inflection under side load alone, unless the designer wishes to compute the distance by a precise formula.

In computing the margin of safety near the outer strut L may be taken as twice the distance from the support to the outer point of inflection.

In computing ρ for the purpose of applying the curves of Figure 7 filler blocks may be neglected and, in case of a tapered spar, the average value should be used.

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

III. The bending moment from which f_b is computed must include an allowance for secondary bending. When possible, this should be done by using a precise method for computing the bending moments. When this can not be done, a rough allowance must be made for their effect.

For beams subjected to a uniformly distributed lateral load, the distance between points of inflection can be obtained from the nomogram Figure 8.

(*h*) At points of application of large concentrated loads, the routing of I spars shall be omitted and filler blocks used in the space between the flanges of box spars. These filler blocks must be designed to carry the local stresses efficiently and to avoid undue concentrations of stress. In order to avoid an abrupt change of section at the ends of the blocks, they should taper with an angle of not more than 30° to a thickness of not more than ¼ inch. The middle portion of the block may, if desirable, have sides perpendicular to the spar axis. The size and weight of filler blocks can often be reduced and the tendency to check eliminated by constructing them of plywood, the individual plies being from ⅛ to ¼ inch in thickness. The plies adjacent to the spar webs should have their grain parallel to the spar axis.

(*i*) In computing the area, moment of inertia, etc., of wood spars pierced by bolts, the diameter of the bolt hole shall be assumed to be 25 per cent greater than the diameter of the bolt.

(*j*) In computing the properties of box spars for final design only that portion of the web with its grain parallel to the spar axis and one-half of that portion of the web with its grain at an angle of 45° to the spar axis shall be considered. If desired, the more conservative method of neglecting the web entirely may be employed. When computing the form factor for box spars, the total thickness of both webs shall be used.

(*k*) For preliminary work, a box spar may be considered as a truss with a continuous web. The total load in each flange, due to bending, will then be the bending moment divided by the distance between the centroids of the two flanges. Dividing this value by the flange area will give the average unit stress in the flange. The average unit stresses in each flange due to bending and axial load should be added algebraically to find the total unit stresses. In this method the allowable net stress in the compression flange should be taken as 5,000 pounds per square inch and in the tension flange as 10,000 pounds per square inch for first-class spruce.

(*l*) All spruce spars should have a ratio of height to width less than 5, whether they are of box or solid construction, unless particular attention has been paid to lateral support. In all beams, whether of wood or metal, if the moment of inertia about the vertical axis is less than one-fifth that about the horizontal axis of the beam, an investigation shall be made of the strength of the beam against lateral buckling. The following procedure is acceptable but the designer may, if he desires, submit an alternative procedure for consideration. Assume that the load to be carried is the end load in the front beam plus the end load in the rear beam for a given condition of flight; that each beam acts as a pin-ended column whose length is the length of a drag bay and that the strength available is the strength of the front beam plus the strength of the rear beam. To allow for the strength contributed by the leading edge, fabric, etc., the load factor required is two-thirds of the normal load factor specified for the condition investigated. In box spars the area of the tension flange should always be at least one-half the area of the compression flange.

(*m*) In designing the webs of box spars the maximum intensity of stress in longitudinal shear shall be computed by the formula, $f=SQ/bI$, where S is the magnitude of the vertical shear at the section, Q is the statical moment about the neutral axis of the area above the section under consideration, b is the entire web thickness, and I the moment of inertia of the cross section. The allowable unit stress in longitudinal shear for two-ply spruce plywood having the grain at 45° to the axis of the spar may be obtained from

$$Fs=960+\frac{3,140}{\sqrt[3]{C}}-45.5D$$

when C is the spacing of the stiffeners or diaphragms, in inches, D is the distance between the centroids of the top and bottom chords of the beam, in inches. This formula gives satisfactory results for beams whose effective depth—the distance between chord centroids—

is greater than 4.5 inches. For other woods the allowable strengths may be computed from the relation between the strength properties of the wood used and those of spruce, or by test. For more complete data on the shear strength of plywood webs of box beams, see Air Corps Information Circular No. 587, by Roy A. Miller.

(*n*) When the deflections of box, solid rectangular, or I spars are desired, they should be computed by one of the common formulas of mechanics. Deflections of trusses, no matter how shallow, should be computed by one of the standard methods applicable to trusses. The use of a formula applicable to beams for the computation of truss deflections will not be approved, except where tests have been made, on the actual truss to determine its effective "*EI*."

(*o*) Where a joint in a spar is designed to transmit bending from one section of the spar to another, the stresses in each part of the structure shall be calculated on the assumption that the joint is 100 per cent efficient and also under the assumption that the bending moment transmitted by the joint is but 50 per cent of that obtained under the assumption of perfect continuity. Each part of the structure shall then be designed to carry the most severe load as determined from the above assumptions.

(*p*) If eccentric fittings are used on any member, due allowance must be made for the resulting changes in stress distribution. The effect on the member itself must be carefully investigated.

(*q*) The slenderness ratio of struts—length divided by minimum radius of gyration—shall be less than 200.

2. Fuselage.—(*a*) In all fuselages the longerons, vertical, and diagonal struts shall be designed with a coefficient of restraint between 1 and 2, according to the judgment of the designer and the type of fitting. In all cases the value assumed for the cofficient of restraint must be stated by the designer.

(*b*) Except for airplanes weighing less than 1,000 pounds, the minimum sizes of steel tubing recommended for the primary members of wire-braced fuselage structures are: ¾ inch by 0.035 for longerons and web members of the side trusses and members of the engine mounting; ⅝ inch by 0.035 for web members of the top and bottom trusses.

For fuselages of all tubular construction, each of these sizes may be reduced ⅛ inch.

These sizes are advisable as minima, where their length will not be greater than 30 to 36 inches, from the standpoint of maintenance and handling, although they may appear to be greatly overstrength for the loads imposed on them in the design conditions. For larger fuselages the diameters given above should be increased, or the tubes made sufficiently strong to carry a load equal to the sum of the vertical, or horizontal, components of all wires attached to the member in question when such wires carry an initial tension of 25 per cent of their rated strength.

(*c*) It is recommended that the attachment of items of equipment, etc., to main members of the fuselage structure be made by means of clips or pins, not by welding.

3. Landing gear.—(*a*) Shock-absorber cord installations shall be designed to withstand the shocks of landing, and care must be taken in construction that the cord is applied with the proper initial

nsion to give the shock-absorbing characteristics upon which the design was based.

(*b*) The effect of eccentricities in the chassis due to the rise of the axle or the relative locations of other members shall be carefully considered in the design.

4. Fittings.—(*a*) The dimensions of all lugs should be such that they will fit standard clevises of corresponding sizes. The average strength of wire lugs of any design shall be at least 15 per cent in excess of the rated strength, and the actual strength of any individual lug must not be below the rated strength. Lugs made of cold rolled steel, in conformity to the limiting dimensions given in Figure 9, are considered to comply with the above requirements. Lug designs of other materials, or not in conformity with the dimensions of Figure 9, will have to be tested before acceptance. Such tests must be carried to destruction of the lug and the type of failure reported. The following rules govern the use of Figure 9.

I. Dimensions T and D are determined by the dimensions of the fork ends of the standard terminals and the area required for bearing of the pin.

II. Dimension K is determined by the required strength and the fork-end dimensions. It shall be neither exceeded nor diminished, except to the extent of the manufacturing tolerances.

III. Dimension M is a minimum which may be exceeded if desired.

IV. Dimension B is the least width of shank allowable with the given plate thickness, t, and may be exceeded if desired.

V. When pin plates are used, care must be taken that the stress imposed on the pin plates by the pin can be carried into the shank of the lug.

(*b*) In fitting design care must be taken not to pierce a structural member at a point that will seriously weaken it, or to make wires or struts unnecessarily eccentric. Proper allowance shall be made for material removed by bolt holes in designing spars, longerons, and other members carrying stress.

(*c*) Clevis pins should always be inserted in their fittings with the head uppermost on assembly of the airplane, and wire, cable and strut terminals, and bolts should be so inserted. The fittings should be designed accordingly. This is to reduce the possibility of the pin or bolt falling out in the event of failure to safety it properly with wire or cotter pins.

(*d*) The threaded portion of a bolt should not be used to take a shear load when the shear load on the bolt is greater than 25 per cent of the bolt's rated shear strength. When the shear load exceeds this value, not over one-fourth thickness of the fitting should bear on the threads. This condition may be obtained by using washers where necessary.

(*e*) In determining the sizes of solid steel bolts to be used in wood the strength of the wood in bearing against the bolts shall be obtained from Figure 10. No bolt smaller than the ¼–28 size should be used in the primary structure. The use of hollow bolts with comparatively thin walls is to be avoided, as they tend to bend and throw the load toward the outer portions of the bolt hole. Tests at the Forest Products Laboratory show that such bolts are little, if any, more efficient than solid bolts of the same area.

Standard eyebolts or similar special bolts having a fillet between the head and shank with a radius less than 1/8 inch must not be used in places where they will be subjected to bending and vibration. Particular care must be taken to avoid the use of such bolts in locations in the control systems and surfaces where vibration might cause fatigue failure.

(*f*) Commercial machine screws shall not be used as substitutes for aircraft bolts in any part of the primary structure.

(*g*) It is recommended that fittings be made up of the smallest practicable number of constituent parts, so that reliability will be increased and welding, brazing, and riveting in their manufacture be reduced to a minimum. To attain this end, the use of pin plates should be avoided whenever possible. When wires of different sizes must be attached to the same plate, spot facing can usually be resorted to in order to avoid the necessity for pin plates.

Flame brazing or partial dip brazing shall not be practiced on assemblies that constitute parts of the primary structure.

5. Control system.—(*a*) Where a wheel control is used, the wheel should have an angular motion varying from 180° to 360° from the neutral position. The wheel diameter should be from 14 to 20 inches. The distance from the wheel in its rearmost position to the back of the pilot should be 14 inches, and its forward travel should be from 18 to 24 inches. The height of the wheel should be such that it will clear the pilot's legs with the seat in highest position, so that the range of seat adjustment will not be limited. The rudder pedals on large airplanes may have a motion up to 10 inches total.

(*b*) The movement of the control surfaces should be not less than 20° past neutral in either direction. In some airplanes it may be desirable to favor an upward elevator movement. There should be no interference between the surfaces or their bracing when one is held in its extreme position and the other operated through its full angular movement.

(*c*) All control systems should be provided with stops, so that their movements will be limited by these stops rather than be accidental interferences. When possible, such a stop should be placed in proper relation to the control stick or column near its fulcrum, so as to limit its movement to the desired maximum in every direction. This will prevent the control from sticking or bearing against the pilot's limbs or body in the event of a structural failure in flight or in crash. It will also serve to protect the instruments and prevent jamming of the stick behind other controls or instruments.

(*d*) The maximum pull on the stick that can normally be applied by a pilot in flying position is assumed to be 200 pounds. The stick should be designed to withstand a push or pull of 300 pounds in a fore-and-aft direction and a force of 150 pounds in a lateral direction. The stick should be made of a nonferrous material, unless the position of the compass is such that the use of a control stick of ferrous material would not cause deviation of the needle. A wheel control should be designed to withstand a force of 300 pounds fore and aft, and a couple equal to 125 pounds times the diameter of the wheel applied tangent to the rim of the wheel.

The rudder bar or pedals should be built to withstand a force of 300 pounds applied at the point of contact with the foot. On air-

planes having a gross weight of 1,000 pounds or less, two-thirds of the above values may be used. In any case the strength of the control system shall be sufficient to withstand loads on the control surfaces 25 per cent greater than those specified in Table 1 for the design of the surfaces.

(*e*) The maximum movement of the control surfaces should correspond to the maximum movement of the controls.

(*f*) Care should be exercised in the design that friction in controls is reduced to a minimum. Properly designed bell cranks, carefully hung pulleys, and, on large airplanes, the use of ball bearings will aid in this.

(*g*) Unless the design of the control surface hinges is such that the possibility of failure is very remote, at least three hinges should be used on each surface.

(*h*) It is important that the linkage be true to prevent binding of some of the parts. Play or lag in controls should also be reduced to a minimum.

(*i*) Care must be taken in the design to prevent the possibility of the jamming of the controls in case of failure of the seat, floor, cowling, or similar parts of the airplane.

(*j*) When speed is of great importance, cables and control masts should, when practicable, be internal. When this is done, however, inspection holes should be provided, as it is essential that controls be inspected frequently.

(*k*) Strength calculations should always be made for the important members in control systems. In these calculations the strength of all members carrying loads from the control system to the primary structure should be investigated, and also the possibility of the arrangements used causing excessive stresses in the primary structure.

(*l*) The points of attachment of control wires or push rods to ailerons should be located opposite drag struts in such a manner as to prevent the application of torsion to the rear spar. If this is not feasible, the effect of such torsion should be considered in design.

(*m*) Deductions must be made for bolt holes which pierce a member at a critical section. If possible, they should be avoided. The stick especially should be investigated.

(*n*) Where tubes are subjected to both bending and torsion they must be investigated for the effects of the stresses due to the combined loading. The allowable shearing stress may be determined from Figure 11.

(*o*) External bracing of tail surfaces is desirable on all types of airplanes. If unbraced surfaces are used, the support from the fuselage must be especially rigid.

(*p*) The attachment of control masts to surfaces should be given careful attention. In general, a broad base for the mast must be provided. The masts and ribs to which they may be attached must be placed in line with the respective control cables. As a rule, hollow streamline steel masts have been found to give the best results.

(*q*) In a control system the attachment of levers to tubes is important. The use of a sleeve to reinforce the tube at the point of attachment of the lever is recommended.

(*r*) In control systems welds shall in no case be employed to carry tension without reinforcement from rivets or bolts.

(*s*) Bushings and pins at bearing points should be large enough to prevent distortion and consequent looseness and have a means of lubrication to prevent wear. They should be designed for a maximum bearing pressure of 750 pounds per square inch, based on design loads.

6. Struts and columns.—Long, slender struts shall be designed by the Euler formula,

$$P/A = c\pi^2 E/(L/\rho)^2 \text{ or } P = c\pi^2 EI/L^2.$$

Short struts of spruce or steel shall be designed by the Johnson parabolic formula,

$$P/A = f - f^2L^2/4c\pi^2E\rho^2 \text{ or } P = fA - (fLA)^2/4c\pi^2EI.$$

Short struts of duralumin tubing shall be designed by the straight line formula,

$$P/A = f - KL/\rho\sqrt{c} \text{ or } P = fA - KLA/\rho\sqrt{c}.$$

The quantity c in the above formulas is the restraint coefficient which is to be taken as 1 for pin-ended struts and as 4 for truly fixed-ended struts. In the design of a strut its value must be assumed by judgment, but in airplane work, as it is very unlikely that a degree of restraint greater than that represented by $c=2$ can ever be counted on, that value is the maximum which will be permitted by the department. The quantity K in the formula for duralumin, an empirically derived constant, is equal to 400; "f" is the yield-point strength of the material except for duralumin, in which case it is 48,000. E, I, L, and ρ have their usual significance.

PART 4.—PROPERTIES OF MATERIALS

1. The materials used in aircraft structures must be of the best. Since it is impossible, at the present time, for the department to draft a complete set of specifications or to inspect and approve all materials to be used in aircraft, it is accepting those that conform to the specifications of the United States Army or Navy, the Society of Automotive Engineers, or other recognized standard. In any case the use of definite physical properties for a given material will be assumed by the department as a guaranty by an aircraft manufacturer that he will use only materials having those properties.

The use of materials of inferior quality or of those which experience has shown to lack uniformity of quality or strength will be regarded as sufficient cause for withholding approval of a new design or for canceling approved type certificates or licenses already granted.

2. The following tabulations of strength properties are, for the most part, based on materials conforming to Army-Navy specifications, and they are given solely for the information of manufacturers as to what can be expected from materials of a quality satisfactory for use in aircraft construction. The Department of Commerce does not guarantee them nor assume responsibility for them in any way but offers them solely as consistent and reasonable values for materials which may be obtained commercially under standard specifications. Manufacturers prepared to guarantee higher strength of properties for their materials may do so by presenting test data that shall be satisfactory to the Department of Commerce.

3. Spruce and ash.—

Properties for airplane spruce and ash conforming to Army-Navy specifications

		Spruce	Ash
Moisture content	per cent	15	15
Modulus of elasticity	pounds per square inch	1,300,000	1,460,000
Allowable tensile stress	do	10,000	16,000
Modulus of rupture (rectangular sections)	do	9,400	14,800
Ultimate compression, parallel to grain	do	5,000	7,000
Ultimate compression, perpendicular to grain	do	840	1,920
Longitudinal shear strength	do	750	1,380
Average weight per cubic foot	pounds	27	41
Average specific gravity (based on oven-dry weight and volume)		0.40	0.62
Minimum specific gravity permitted		.36	.56

The strength of two-ply spruce webs on box spars or compression ribs may be determined by the use of the formula in paragraph 1(*m*) of part 3, Section I.

4. Steel.—

Allowable stresses in pounds per square inch

	"Commercial"	Mild carbon	Chrome-molybdenum	Heat-treated alloy steels		
Steel No.	1,015-1,020	1,025	4,130	2,330, 3,130, 3,240, 4,130, 6,123, etc.		
Ultimate tensile strength	42,000	55,000	90,000	125,000	150,000	180,000
Yield point	27,000	36,000	60,000	105,000	125,000	150,000
Shearing strength	25,000	35,000	60,000	75,000	90,000	105,000
Bearing strength	65,000	95,000	125,000	160,000	190,000	210,000
Modulus of elasticity	28×10^6	28×10^6	29×10^6	29×10^6	29×10^6	29×10^6

The efficiency of a welded joint in unheat-treated steel should be taken as 80 per cent. A welded joint made after steel is heat treated reduces the strength properties to those for the unheat-treated material. Higher values should not be assumed unless the assembly is heat treated after welding.

Streamline tube formed from flat sheet and welded is rated as "commercial," and its strength properties may be obtained from that column. Drawn seamless streamline tubing may be obtained in mild carbon, chrome-molybdenum, or nickel steel, and the properties employed should conform to the type of steel used.

The properties listed for the heat-treated steels will vary somewhat for the different steels, but the values listed above may be used as a guide.

5. Aluminum alloy (duralumin).—

Weight per cubic foot	pounds	175
Ultimate tensile strength	pounds per square inch	55,000
Yield point	do	30,000
Ultimate shearing strength of rivets	do	30,000
Ultimate bearing strength	do	75,000
Modulus of elasticity	do	10,000,000

The properties listed above will vary with the different alloys, but those given may be used as a guide for the more commonly available alloys, such as 17ST, made by the Aluminum Co. of America.

SECTION II. AIRPLANE-ENGINE TEST REQUIREMENTS

1. In accordance with section 10 of the Air Commerce Regulations, it is necessary that the power plant and power-plant installation of all airplanes, that are to be considered as airworthy, be approved by the Department of Commerce.

Engines which have passed the regular endurance tests of the United States Army Air Corps or the United States Navy will be approved by the Department of Commerce. Other engines submitted for approval will be tested by the Department of Commerce at the Bureau of Standards. An exception to this latter provision may be made in the case of established aircraft engine manufacturers whose product has previously been approved, which will permit certain tests to be run by such manufacturers provided their testing equipment and facilities have been approved by the department. In such case, however, the tests will need to be run in the presence of an observer from the department.

2. Test data.—The log shall include the following: Piston displacement, compression ratio, type of ignition, carburetor and spark plugs; barometric reading and complete list of instruments and apparatus used for the test. The length of brake arm and gear ratio if propeller is geared, shall also be given. The test shall be such as to develop the following data:

(*a*) A plot of the power developed at various speeds with the throttle wide open.

(*b*) Log of 50-hour endurance test showing readings, at approximately 30-minute intervals, of:

I. Revolutions per minute.
II. Brake load, in pounds.
III. Oil pressure and temperatures.
IV. Water temperatures.
V. Air temperature at varburetor intake.
VI. Blast velocity.
VII. Maximum cylinder-head and cylinder-barrel temperatures.
VIII. Manifold depression, in inches of mercury.
IX. Fuel time, in seconds, for at least 5 pounds.
X. Oil, in pounds.
XI. Brake horsepower.
XII. Remarks, including breakages and replacements.

(Item IV does not apply to air-cooled engines and Items VI and VII are not applicable to water-cooled enginers.)

(*c*) Photographs showing set-up used and any parts which failed or showed excessive wear.

(*d*) Lists of instruments and apparatus used.

(*e*) Report of final tear-down inspection.

3. Preliminary requirements.—Engines should be designed so that some type of starter can be fitted. If a starter is regularly supplied with the engine, such equipment either shall be used for all starts during the 50-hour test or shall be tested for reliability.

Dual ignition systems, with at least two spark plugs per cylinder, will be required on all engines.

Before it is tested the engine shall have been run by the manufacturer for at least 25 hours, and a log of this run shall be submitted to the Department of Commerce. The only requirement for this preliminary test is that the horsepower developed shall not average less than one-half rated power.

The manufacturer may be called upon for an explanation if his engine shows an excessive manifold depression at full throttle.

4. Block tests.—The 50-hour endurance test shall be run in ten 5-hour periods. During the first five hours the engine shall be run with the throttle wide open, the speed being at least equal to the rated speed and the power at least 10 per cent in excess of the rated power. During the remaining 45 hours the engine shall be run at approximately rated speed, and the horsepower developed should at no time be less than the rated horsepower.

During the test not more than three forced stops shall be allowed. A penalty run of two hours will be added to any period in which a forced stop is made. If the speed during a propeller-load run drops to less than 97 per cent of rated speed or the power in a dynamometer run drops as much as 10 per cent, this shall require a forced stop. Excessive water, fuel, or oil leaks developing at the engine shall require forced stops. In all cases the Bureau of Standards shall be the judge as to what constitutes a forced stop.

Minor repairs and replacements may be made between test periods if suitable tools and parts are available, but the test schedule shall not be delayed on this account.

The failure of a major part of the engine shall terminate the test.

If the engine is submitted for test without qualification as to fuel, any available gasoline meeting the Federal specification for United States motor gasoline may be used. If the manufacturer does not consider that his engine can safely be run on all grades of motor gasoline, the engine shall be tested on the grade of fuel which he specifies and will be certified only for use on fuels of this grade or better.

On completion of the 50-hour endurance test, the necessary runs will be made to obtain a curve of power against speed by varying the load with the throttle wide open. Readings should be made at speeds ranging from 75 per cent of normal speed to approximately 110 per cent of normal speed. Each speed should be maintained at least for five minutes before a reading is taken.

5. Inspection.—After completion of the block tests, a complete tear down and detailed inspection of engine parts shall be made, particular attention being paid to excessive wear or signs of failure.

6. Optional tests.—(*a*) A tear-down inspection of the engine may be made, at the option of the department, prior to the 50-hour endurance test.

(*b*) After successfully passing the required block test, a bench test of the carburetor may be made, at the option of the department, to determine whether the carburetor will continue to supply a satisfactory mixture when it is tilted at various angles.

7. After successfully passing the required block test, the engine shall be installed in an airplane and given an actual service test of at least 10 hours in the air, which shall include a climb at full throttle to the service ceiling of the airplane.

8. Engines which pass the requirements of this section to the satisfaction of the department will be approved for use in licensed airplanes or may be granted an approved type certificate in accordance with the provisions of section 11 of the Air Commerce Regulations.

SECTION III. AIRPLANE-PROPELLER REQUIREMENTS

1. All airplanes which are to be licensed by the Department of Commerce or to be built under the provisions of an approved type certificate shall be equipped with propellers approved for a power equal to or greater than the rated power of the engine on which they are to be installed.

2. The propeller drawings shall be submitted to the Department of Commerce in duplicate, with the data on which the manufacturer bases his request for approval, supported by affidavit. If the design meets the requirements of paragraph G, section 10, of the Air Commerce Regulations, one set of drawings will be returned to the manufacturer, impressed with the seal of the Department of Commerce, with the notation of the maximum power for which the approval is valid. Propellers built to conform, within the usual tolerances, to these approved drawings will be accepted as airworthy and may, upon the manufacturer's making application in accordance with section 11 of the Air Commerce Regulations, be granted an approved type certificate. Any deviations must, however, be approved by the department before being incorporated in the design.

3. Propellers which satisfy any of the following requirements will be accepted:

(*a*) Designs from which a propeller has been flown for 150 hours, without failure, on an engine of equal or greater power than that for which the approved rating is sought. Affidavit, specifying the number of hours' satisfactory service on a given engine and aircraft, the name of the operator, operating firm, and other pertinent data, shall be forwarded to the department with the drawings.

(*b*) Designs which embody only minor modifications from propellers accepted and used by the United States Army or Navy, the contractor to state specifically departure from Government design, if any.

(*c*) Designs which are approved designs, cut down to smaller diameters without reduction of the maximum ordinates, except in the immediate neighborhood of the tip.

(*d*) Designs from which a propeller has passed, without failure, the following 10-hour overload test, driven by a stationary engine:

Metal propellers, 10 hours, at 100 per cent power overload, if tested by electrical apparatus, or 75 per cent power overload, if tested by means of an internal-combustion engine.

Wooden propellers, 10 hours, at 50 per cent power overload, if tested by electrical apparatus, or 40 per cent overload, if tested by an internal-combustion engine.

The power overload may be determined according to any sound dynomometer theory, or may be based on the assumption that the power absorbed is proportional to the cube of the speed, in which case the revolutions per minute at which the test run is made shall be calculated by the following formula:

$$N_t = \sqrt[3]{1 + \%\text{ power overload}}\ N$$

N_t is the revolutions per minute at which the test is to be run, and N is the ground revolutions per minute at which the propeller will absorb the rated power.

A log of the test run and a photograph of the test set-up, preferably taken after the run, shall be forwarded to the department with the drawings.

In running such a test, a breaking point should be provided between the propeller and the source of power; otherwise, failure of the propeller will destroy the motor or engine.

Controllable-pitch propellers shall be tested at 100 per cent power overload, on either electrical or internal-combustion apparatus, and the operation of the pitch-control mechanism checked over the speed range.

If any propellers undergoing one of the above tests develops a violent flutter before reaching the overload speed, the propeller shall be run at the fluttering speed for an additional 10 hours.

4. Failure is defined as actual breakage or cracking of any part of the blades, hubs, bolts, lock nuts, keyways, etc., or the pulling of the blades out of the hub. In the case of metal propellers a permanent deformation or the development of any fatigue or hair cracks will be considered a failure. If there is any doubt about the presence of a fatigue failure, the propeller shall be etched in accordance with Department of Commerce instructions and examined by the department inspector.

Wooden propellers will also be considered to have failed if the tipping pulls or cracks, glue joints open, or if there is any local failure or crushing around the hub or bolts.

Similar considerations will apply to propellers of any patented composition or variation of the conventional wood or metal construction. In case of doubt of the classification of his design the manufacturer shall apply to the department for a ruling.

5. No propeller in which an abrupt change in cross-section area occurs will be approved.

6. If more than 5 per cent of the propellers in service, made to an approved design, fail in the air under normal flying conditions on engines within the propeller's power rating, the Department of Commerce may rescind its approval of that design.

7. The approval of a metal propeller which has been bent is canceled until it is straightened and reheat-treated by a competent concern. The department should be promptly informed of propeller accidents and the kind and extent of repairs.

8. It is strongly recommended that manufacturers conform, as far as practicable, to Army and Navy standards in hubs, blade shanks, etc. The latter is shown in Figure 12. The fillet radii indicated should not be decreased in any case.

A stress analysis should be made for any new propeller which appreciably departs from conventional designs. It should include:

(*a*) Curve of intensity of centrifugal force at 120 per cent normal revolutions per minute against radius.

(*b*) Curve of centrifugal force against radius.

(*c*) Curve of fiber stress caused by centrifugal force.

(NOTE.—Curve *c* equals curve *b* divided by cross-sectional area at each station.)

(*d*) Thrust curve normal flight conditions against radius; equivalent to load curve on cantilever beam.

(*e*) Curve *d*, increased by 40 per cent, equals approximate thrust during take-offs or climb.

(*f*) Second integral of curve *d* equals bending-moment curve against radius.

(*g*) Fiber stress caused by bending moment (during take-off or climb).

(*h*) Combined fiber-stress curve (pounds per square inch) against radius. Addition of curve *c* and curve *g*.

For wood: Maximum fiber stress not to exceed 4,000 pounds per square inch. For metal: Maximum fiber stress not to exceed 12,000 pounds per square inch—(25S) aluminum alloy.

9. Limitations on joints (shear shoulders) between detachable blades and hubs.

Army and Navy standard stub ends of blades are recommended for split-hub propellers. The blade elements should be tangent to the cylindrical portion of the shank under the clamp ring.

Metal propellers which have engineering data or trade-marks stamped upon the working part of the blade will not be approved. It is recommended that such information be painted on.

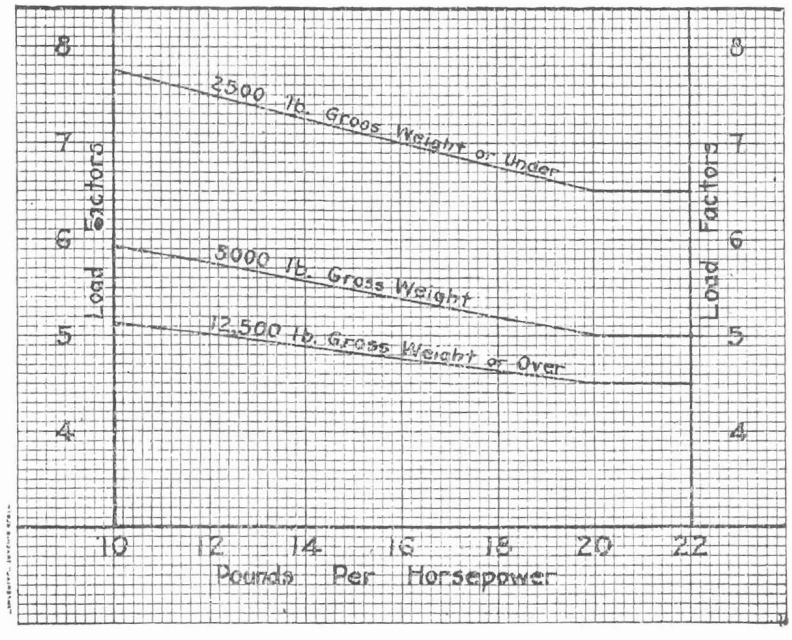


FIG. 1.—*Load factors—High angle of attack*

For flying boats and amphibians of 5,000 pounds gross weight or less, use the 5,000-pound gross weight factors. Horsepower loading shall be based on rated engine horsepower except that for multi-engined airplanes the horsepower shall be taken as that necessary to maintain level flight with full load.

NOTE.—It is to be noted that the load factors given in this chart are not sufficiently high to conform to the international requirements throughout the entire range of horsepower loading. Manufacturers designing airplanes for export should take into account the load factors and design requirements of the country to which the airplane is to be exported.

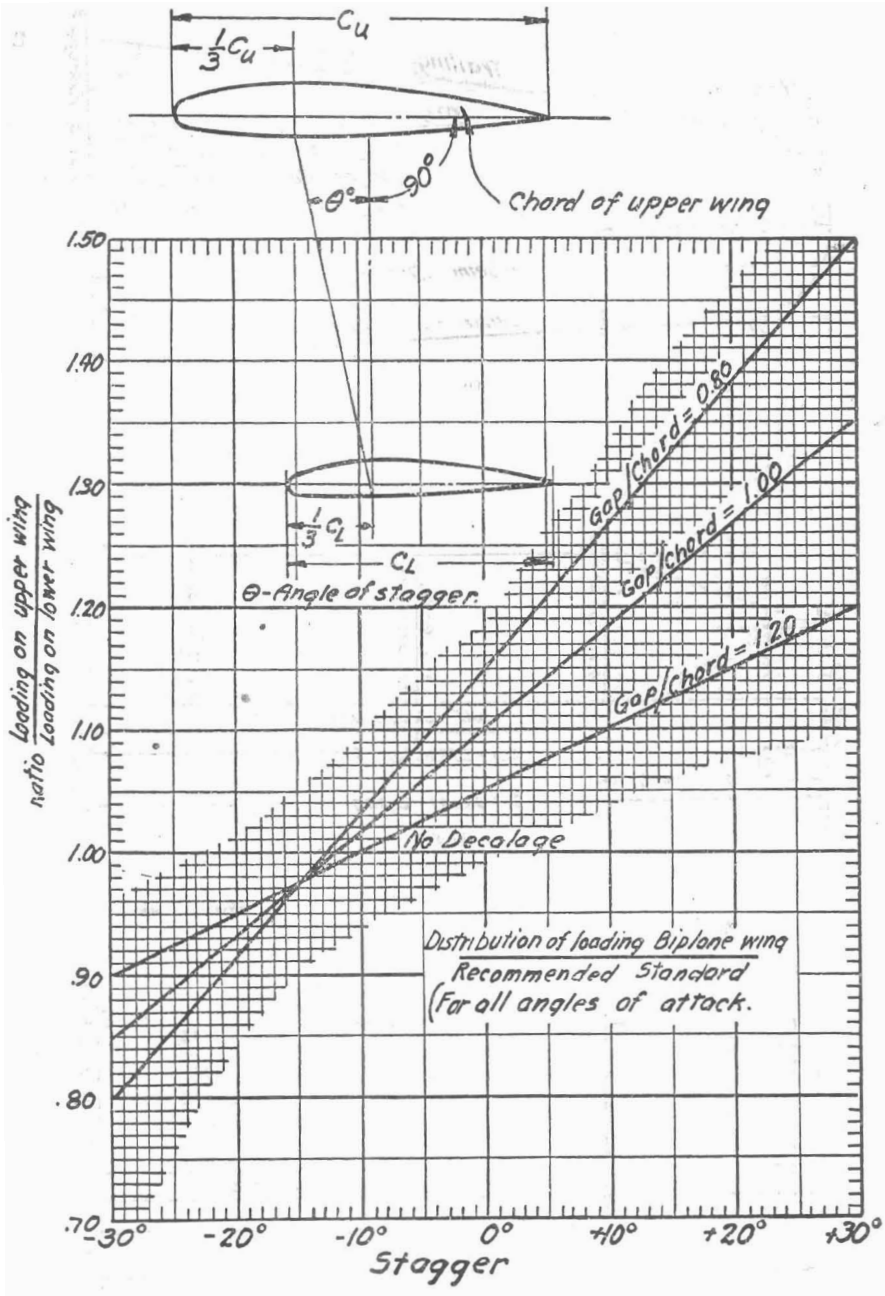

C_u
$\frac{1}{3}C_u$
$\theta°$
90°
Chord of upper wing
$\frac{1}{3}C_L$
C_L
θ-Angle of stagger.
Gap/Chord = 0.80
Gap/Chord = 1.00
Gap/Chord = 1.20
No Decalage
Distribution of loading Biplane wing
Recommended Standard
(For all angles of attack.
ratio Loading on upper wing / Loading on lower wing
1.50
1.40
1.30
1.20
1.10
1.00
.90
.80
.70
-30°
-20°
-10°
0°
+10°
+20°
+30°
Stagger

FIG. 2

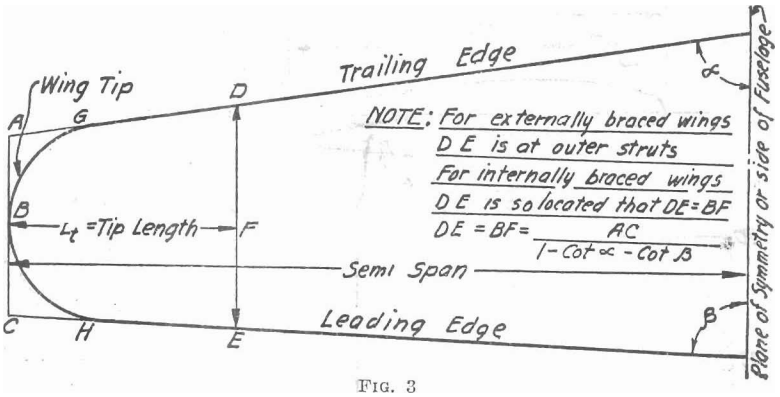


FIG. 3

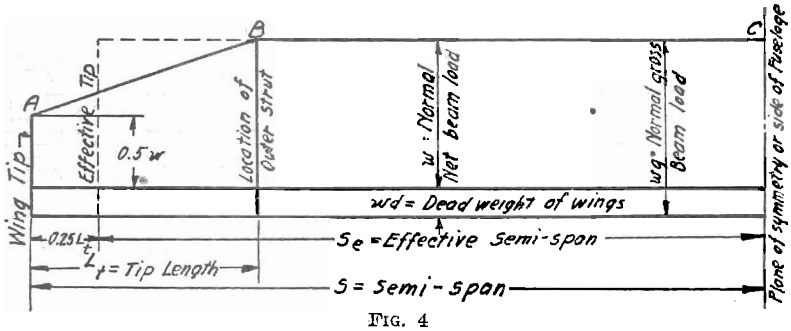


FIG. 4

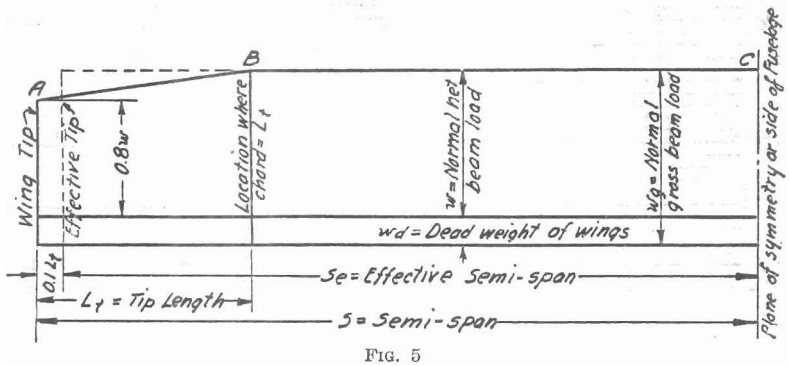


FIG. 5

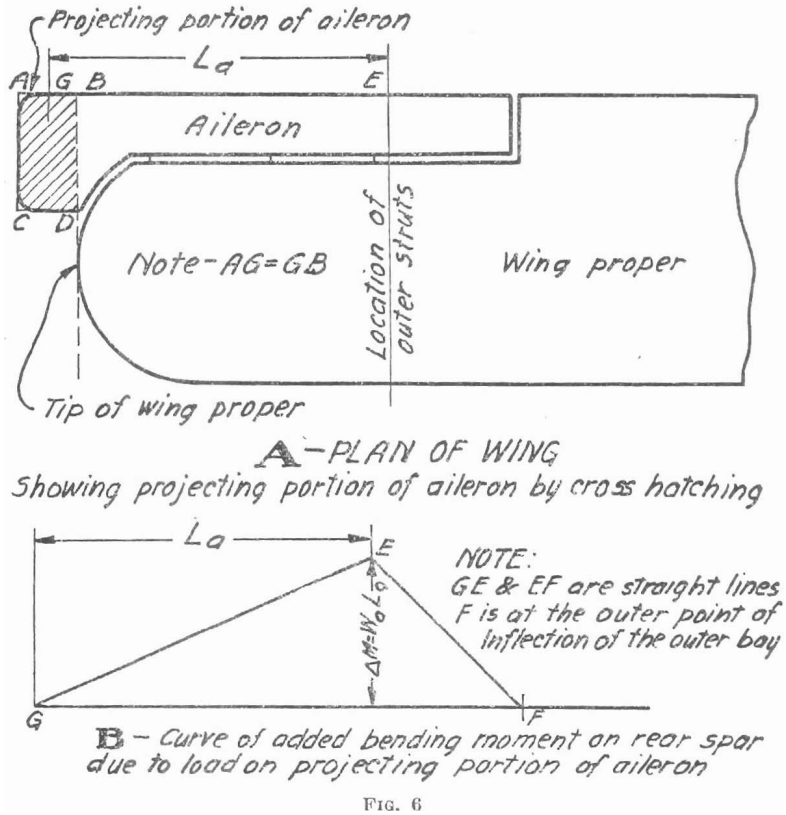

Projecting portion of aileron
L_a
A G B
E
Aileron
C D
Note-AG=GB
Location of outer struts
Wing proper
Tip of wing proper
A-PLAN OF WING
Showing projecting portion of aileron by cross hatching
L_a
E
NOTE:
GE & EF are straight lines
F is at the outer point of inflection of the outer bay
$\Delta M = W_a L_a$
G
F
B - Curve of added bending moment on rear spar due to load on projecting portion of aileron

FIG. 6

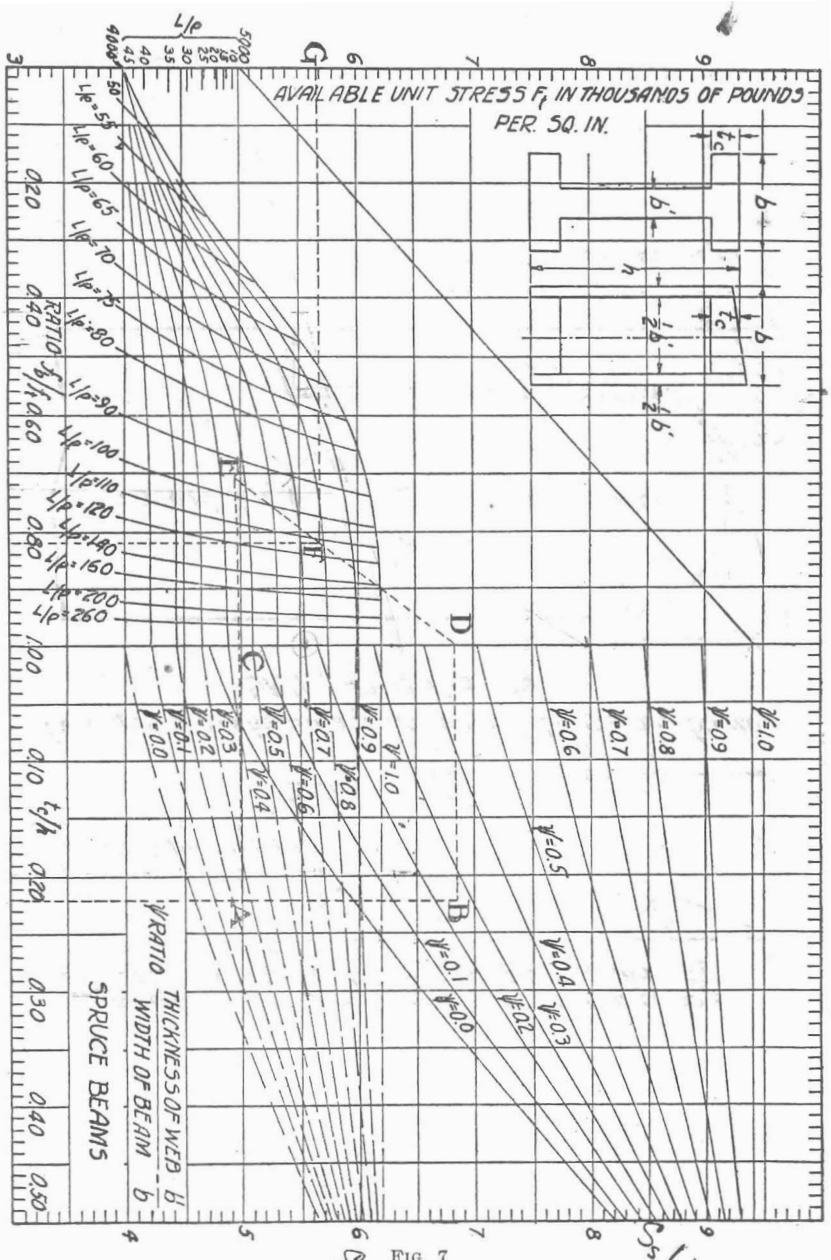
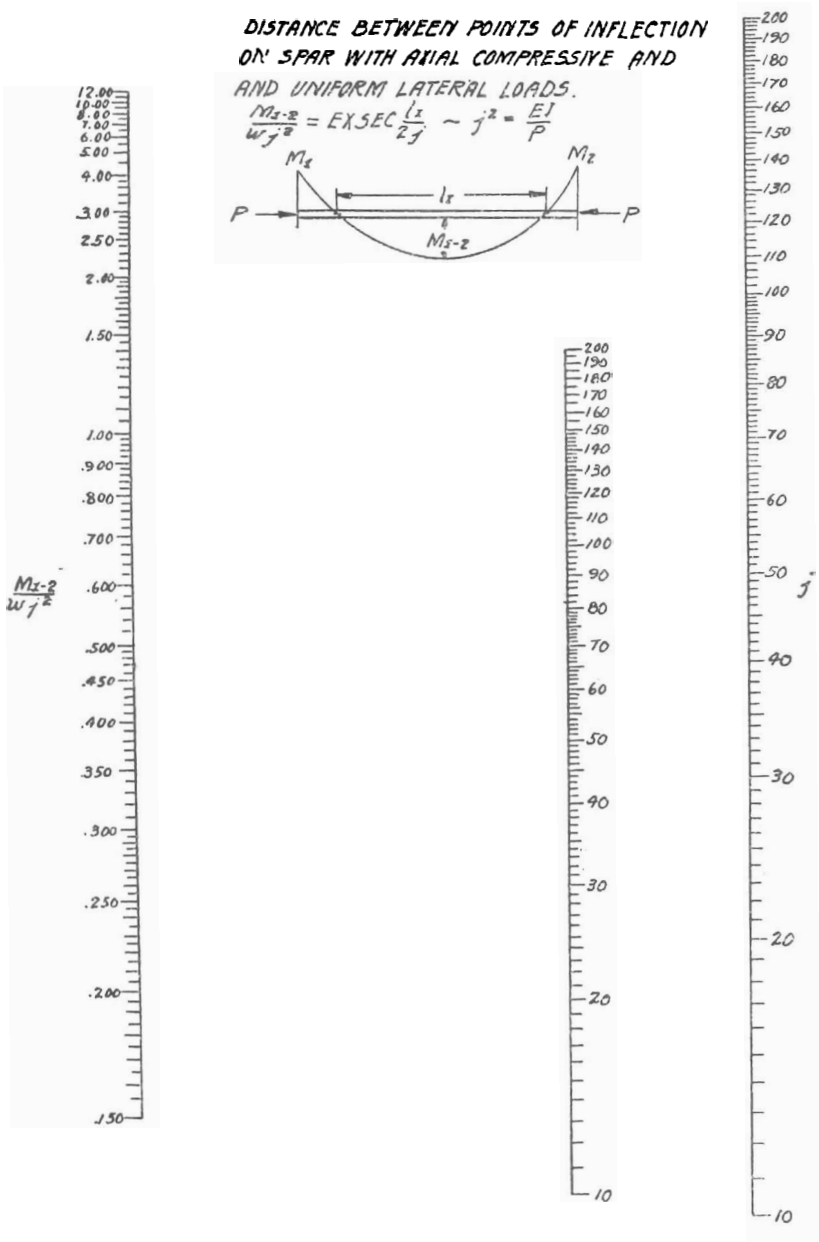
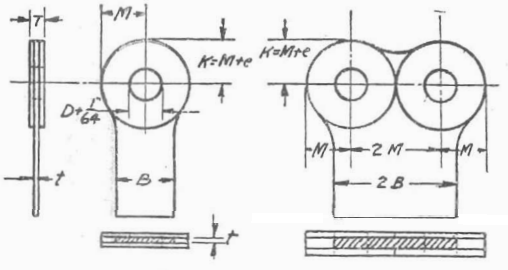

AVAILABLE UNIT STRESS F_f IN THOUSANDS OF POUNDS PER SQ. IN.
L/p
L/p=50
L/p=55
L/p=60
L/p=65
L/p=70
L/p=75
L/p=80
L/p=90
L/p=100
L/p=110
L/p=120
L/p=140
L/p=160
L/p=200
L/p=260
RATIO S_f/f
t_c/h
ψ RATIO THICKNESS OF WEB / WIDTH OF BEAM b'/b
SPRUCE BEAMS
S_s elastic limit
S_s Ult.

FIG. 7


DISTANCE BETWEEN POINTS OF INFLECTION
ON SPAR WITH AXIAL COMPRESSIVE AND
AND UNIFORM LATERAL LOADS.
$\frac{M_{x-z}}{wj^2} = EX\,SEC\,\frac{l_x}{2j} \sim j^2 = \frac{EI}{P}$
M_x
M_z
l_x
P
P
M_{x-z}
$\frac{M_{x-z}}{wj^2}$
j



No.	Rated Strength	D.	K.	M	e	T	Width of Shank B								
							t= 1/32	1/20	1/16	3/32	1/8	3/16	1/4	5/16	3/8
11	1,100	3/16	17/64	7/32	3/64	3/32	3/4	1/2	3/8	1/4					
21	2,100	3/16	11/32	9/32	1/16	1/8	1 7/16	15/16	3/4	1/2	3/8				
34	3,400	1/4	13/32	11/32	1/16	3/16	2 5/16	1 7/16	1 5/16	13/16	5/8	7/16			
46	4,600	5/16	17/32	7/16	3/32	3/16		1 13/16	1 9/16	1 1/16	13/16	9/16			
61	6,100	3/8	45/64	9/16	9/64	3/16		2 9/16	2 1/16	1 3/8	1 1/16	3/4			
80	8,000	3/8	45/64	9/16	9/64	1/4			2 1/8	1 15/16	1 7/16	1	3/4		
125	12,500	7/16	13/16	21/32	5/32	5/16				2 13/16	2 1/8	1 7/16	1 11/16	7/8	
175	17,500	1/2	15/16	3/4	3/16	3/8					2 3/8	1 3/4	1 5/16	1 11/16	7/8

FIG. 9

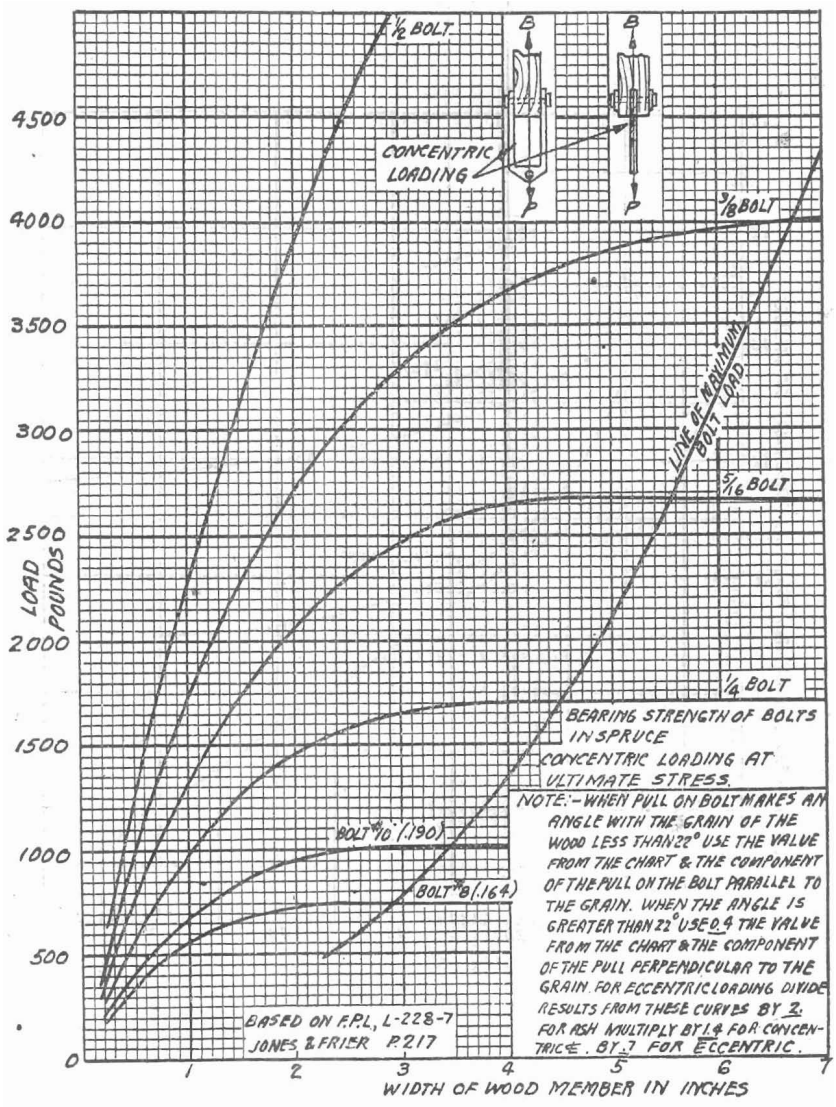

1/2 BOLT
CONCENTRIC LOADING
B
P
3/8 BOLT
LINE OF MAXIMUM BOLT LOAD
5/16 BOLT
1/4 BOLT
BOLT #10 (.190)
BOLT #8 (.164)
BEARING STRENGTH OF BOLTS IN SPRUCE CONCENTRIC LOADING AT ULTIMATE STRESS.
NOTE:- WHEN PULL ON BOLT MAKES AN ANGLE WITH THE GRAIN OF THE WOOD LESS THAN 22° USE THE VALUE FROM THE CHART & THE COMPONENT OF THE PULL ON THE BOLT PARALLEL TO THE GRAIN. WHEN THE ANGLE IS GREATER THAN 22° USE 0.4 THE VALUE FROM THE CHART & THE COMPONENT OF THE PULL PERPENDICULAR TO THE GRAIN. FOR ECCENTRIC LOADING DIVIDE RESULTS FROM THESE CURVES BY 2. FOR ASH MULTIPLY BY 1.4 FOR CONCENTRIC & BY .7 FOR ECCENTRIC.
BASED ON F.P.L. L-228-7 JONES & FRIER P.217
LOAD POUNDS
4500
4000
3500
3000
2500
2000
1500
1000
500
0
1
2
3
4
5
6
7
WIDTH OF WOOD MEMBER IN INCHES

FIG. 10

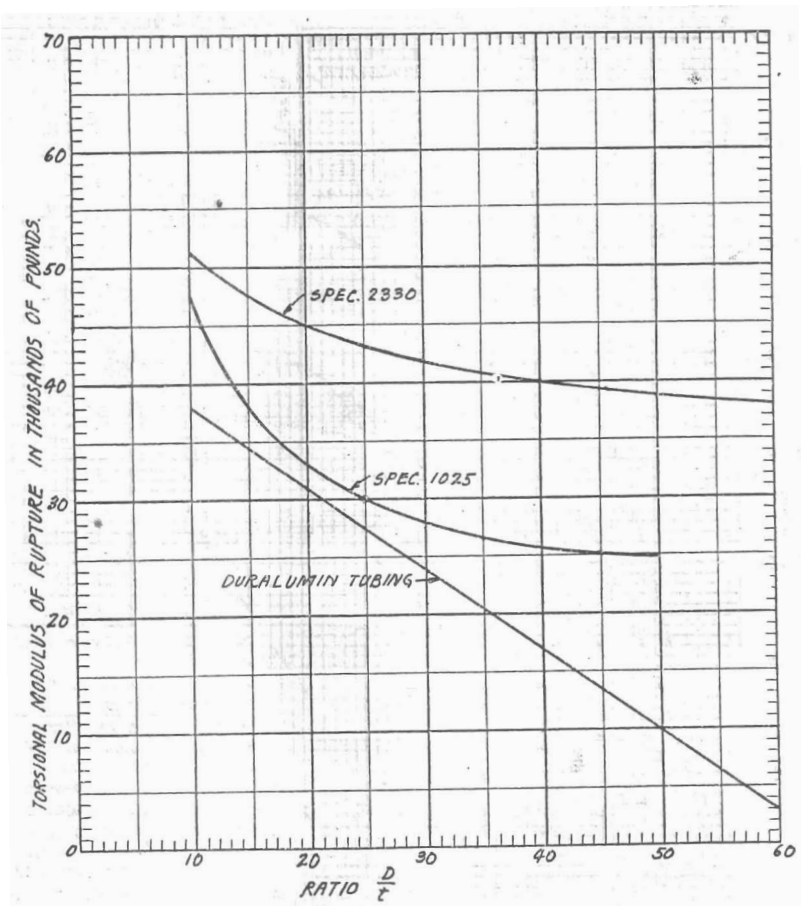

TORSIONAL MODULUS OF RUPTURE IN THOUSANDS OF POUNDS.
70
60
50
40
30
20
10
0
SPEC. 2330
SPEC. 1025
DURALUMIN TUBING
10
20
30
40
50
60
RATIO $\frac{D}{t}$

FIG. 11