Airplane Airworthiness; Normal, Utility And Acrobatic Categories

FEDERAL AVIATION AGENCY

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References to CAA will be changed to FAA where appropriate as page revisions to this manual are issued.

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Introductory Note

Civil Aeronautics Manual 3 contains in a consolidated form all manual material issued against Part 3 of the Civil Air Regulations to date.

CAA *rules* are supplementary regulations issued pursuant to authority expressly conferred on the Administrator in the Civil Air Regulations. Such rules are mandatory and must be complied with.

CAA *policies* provide detailed technical information on recommended methods of complying with the Civil Air Regulations. Such policies are for the guidance of the public and are not mandatory in nature.

CAA *interpretations* define or explain words and phrases of the Civil Air Regulations. Such interpretations are for the guidance of the public and will be followed by the Administration in determining compliance with the regulations.

The manual is arranged to show the number of each section of the regulation followed by the title of the particular section in italic letters. Any rules, policies, or interpretations follow the pertinent section of the regulation and are identified by consecutive dash numbers appended to the regulation section number.

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Subpart A—General

Applicability and Definitions

3.0 *Applicability of this part.* This part establishes standards with which compliance shall be demonstrated for the issuance of and changes to type certificates for normal, utility, and acrobatic category airplanes. This part, until superseded or rescinded, shall apply to all airplanes for which applications for type certification under this part were made between the effective date of this part (November 13, 1945) and March 31, 1953. For applications for a type certificate made after March 31, 1953, this part shall apply only to airplanes which have a maximum weight of 12,500 pounds or less.

3.1 *Definitions.* As used in this part terms are defined as follows:

(a) *Administration.*

(1) *Administrator.* The Administrator is the Administrator of Civil Aeronautics.

(2) *Applicant.* An applicant is a person or persons applying for approval of an airplane or any part thereof.

(3) *Approved.* Approved, when used alone or as modifying terms such as means, devices, specifications, etc., shall mean approved by the Administrator. (See sec. 3.18.)

(b) *General design.*

(1) *Standard atmosphere.* The standard atmosphere is an atmosphere [(see NACA Technical Report 1235)] defined as follows:

(i) The air is a dry, perfect gas,

(ii) The temperature at sea level is 59° F.,

(iii) The pressure at sea level is 29.92 inches Hg,

(iv) The temperature gradient from sea level to the altitude at which the temperature equals −[69.7°] F. is −0.003566° F./ft. and zero thereabove,

(v) The density ρ_0 at sea level under the above conditions is [0.002377] lb. sec.^{2}/ft.^{4}

(2) *Maximum anticipated air temperature.* The maximum anticipated air temperature is a temperature specified for the purpose of compliance with the powerplant cooling standards. (See sec. 3.583.)

(3) *Airplane configuration.* Airplane configuration is a term referring to the position of the various elements affecting the aerodynamic characteristics of the airplane (e. g. wing flaps, landing gear).

(4) *Aerodynamic coefficients.* Aerodynamic coefficients are nondimensional coefficients for forces and moments. They correspond with those adopted by the U. S. National Advisory Committee for Aeronautics.

(5) *Critical engine(s).* The critical engine(s) is that engine(s) the failure of which gives the most adverse effect on the airplane flight characteristics relative to the case under consideration.

(c) *Weights.*

(1) *Maximum weight.* The maximum weight of the airplane is that maximum at which compliance with the requirements of this part of the Civil Air Regulations is demonstrated. (See sec. 3.74.)

(2) *Minimum weight.* The minimum weight of the airplane is that minimum at which compliance with the requirements of this part of the Civil Air Regulations is demonstrated. (See sec. 3.75.)

(3) *Empty weight.* The empty weight of the airplane is a readily reproducible weight which is used in the determination of the operating weights. (See sec. 3.73.)

(4) *Design maximum weight.* The design maximum weight is the maximum weigh[t] of the airplane at which compliance is sho[wn] with the structural loading conditions. [(See] sec. 3.181.)

(5) *Design minimum weight.* [The de-]sign minimum weight is the minimu[m] of the airplane at which complianc[e]

with the structural loading conditions. (See sec. 3.181.)

(6) *Design landing weight.* The design landing weight is the maximum airplane weight used in structural design for landing conditions at the maximum velocity of descent. (See sec. 3.242.)

(7) *Design unit weight.* The design unit weight is a representative weight used to show compliance with the structural design requirements:

(i) Gasoline 6 pounds per U. S. gallon.

(ii) Lubricating oil 7.5 pounds per U. S. gallon.

(iii) Crew and passengers 170 pounds per person.

(d) *Speeds.*

(1) *IAS.* Indicated air speed is equal to the pitot static air-speed indicator reading as installed in the airplane without correction for air-speed indicator system errors but including the sea level standard adiabatic compressible flow correction. (This latter correction is included in the calibration of the air-speed instrument dials.)

(2) *CAS.* Calibrated air speed is equal to the air-speed indicator reading corrected for position and instrument error. (As a result of the sea level adiabatic compressible flow correction to the air-speed instrument dial, CAS is equal to the true air speed TAS in standard atmosphere at sea level.)

(3) *EAS.* Equivalent air speed is equal to the air-speed indicator reading corrected for position error, instrument error, and for adiabatic compressible flow for the particular altitude. (EAS is equal to CAS at sea level in standard atmosphere.)

(4) *TAS.* True air speed of the airplane relative to undisturbed air. ($TAS=EAS(_{\rho_0/\rho})^{1/2}$.)

(5) V_c. The design cruising speed. (See sec. 3.184.)

(6) V_d. The design diving speed. (See sec. 3.184.)

(7) V_f. The design flap speed for flight loading conditions with wing flaps in the landing position. (See sec. 3.190.)

(8) V_{fe}. The flap extended speed is a maximum speed with wing flaps in a prescribed extended position. (See sec. 3.742.)

(9) V_h. The maximum speed obtainable in level flight with rated rpm and power.

(10) V_{mc}. The minimum control speed with the critical engine inoperative. (See sec. 3.111.)

(11) V_{ne}. The never-exceed speed. (See sec. 3.739.)

(12) V_{no}. The maximum structural cruising speed. (See sec. 3.740.)

(13) V_p. The design maneuvering speed. (See sec. 3.184.)

(14) V_{sf}. The stalling speed computed at the design landing weight with the flaps fully extended. (See sec. 3.190.)

(15) V_{s_0}. The stalling speed or the minimum steady flight speed with wing flaps in the landing position. (See sec. 3.82.)

(16) V_{s_1}. The stalling speed or the minimum steady flight speed obtained in a specified configuration. (See sec. 3.82.)

(17) V_x. The speed for best angle of climb.

(18) V_y. The speed for best rate of climb.

(e) *Structural.*

(1) *Limit load.* A limit load is the maximum load anticipated in normal conditions of operation. (See sec. 3.171.)

(2) *Ultimate load.* An ultimate load is a limit load multiplied by the appropriate factor of safety. (See sec. 3.173.)

(3) *Factor of safety.* The factor of safety is a design factor used to provide for the possibility of loads greater than those anticipated in normal conditions of operation and for uncertainties in design. (See sec. 3.172.)

(4) *Load factor.* The load factor is the ratio of a specified load to the total weight of the airplane; the specified load may be expressed in terms of any of the following: aerodynamic forces, inertia forces, or ground or water reactions.

(5) *Limit load factor.* The limit load factor is the load factor corresponding with limit loads.

(6) *Ultimate load factor.* The ultimate load factor is the load factor corresponding with ultimate loads.

(7) *Design wing area.* The design wing area is the area enclosed by the wing outline (including wing flaps in the retracted position

and ailerons, but excluding fillets or fairings) on a surface containing the wing chords. The outline is assumed to be extended through the nacelles and fuselage to the plane of symmetry in any reasonable manner.

(8) *Balancing tail load.* A balancing tail load is that load necessary to place the airplane in equilibrium with zero pitch acceleration.

(9) *Fitting.* A fitting is a part or terminal used to join one structural member to another. (See sec. 3.306.)

(f) *Power installation.*[1]

(1) *Brake horsepower.* Brake horsepower is the power delivered at the propeller shaft of the engine.

(2) *Take-off power.* Take-off power is the brake horsepower developed under standard sea level conditions, under the maximum conditions of crankshaft rotational speed and engine manifold pressure approved for use in the normal take-off, and limited in use to a maximum continuous period as indicated in the approved engine specifications.

(3) *Maximum continuous power.* Maximum continuous power is the brake horsepower developed in standard atmosphere at a specified altitude under the maximum conditions of crankshaft rotational speed and engine manifold pressure approved for use during periods of unrestricted duration.

(4) *Manifold pressure.* Manifold pressure is the absolute pressure measured at the appropriate point in the induction system, usually in inches of mercury.

(5) *Critical altitude.* The critical altitude is the maximum altitude at which in standard atmosphere it is possible to maintain at a specified rotational speed, a specified power or a specified manifold pressure. Unless otherwise stated, the critical altitude is the maximum altitude at which it is possible to maintain, at the maximum continuous rotational speed, one of the following:

(i) The maximum continuous power, in the case of engines for which this power rating is the same at sea level and at the rated altitude.

(ii) The maximum continuous rated manifold pressure, in the case of engines the maximum continuous power of which is governed by a constant manifold pressure.

(6) *Pitch setting.* Pitch setting is the propeller blade setting determined by the blade angle measured in a manner, and at a radius, specified in the instruction manual for the propeller.

(7) *Feathered pitch.* Feathered pitch is the pitch setting, which in flight, with the engines stopped, gives approximately the minimum drag and corresponds with a windmilling torque of approximately zero.

(8) *Reverse pitch.* Reverse pitch is the propeller pitch setting for any blade angle used beyond zero pitch (e. g., the negative angle used for reverse thrust).

(g) *Fire protection.*

(1) *Fireproof.* Fireproof material means material which will withstand heat at least as well as steel in dimensions appropriate for the purpose for which it is to be used. When applied to material and parts used to confine fires in designated fire zones, fireproof means that the material or part will perform this function under the most severe conditions of fire and duration likely to occur in such zones.

(2) *Fire-resistant.* When applied to sheet or structural members, fire-resistant material means a material which will withstand heat at least as well as aluminum alloy in dimensions appropriate for the purpose for which it is to be used. When applied to fluid-carrying lines, other flammable fluid system components, wiring, air ducts, fittings, and powerplant controls, this term refers to a line and fitting assembly, component, wiring, or duct, or controls which will perform the intended functions under the heat and other conditions likely to occur at the particular location.

(3) *Flame-resistant.* Flame-resistant material means material which will not support combustion to the point of propagating, beyond safe limits, a flame after the removal of the ignition source.

(4) *Flash-resistant.* Flash-resistant material means material which will not burn violently when ignited.

(5) *Flammable.* Flammable pertains to those fluids or gases which will ignite readily or explode.

[1] For engine airworthiness requirements see Part 13 of this subchapter. For propeller airworthiness requirements see Part 14 of this subchapter.

Certification

3.10 *Eligibility for type certificate.* An airplane shall be eligible for type certification under the provisions of this part if it complies with the airworthiness provisions hereinafter established or if the Administrator finds that the provision or provisions not complied with are compensated for by factors which provide an equivalent level of safety: *Provided,* That the Administrator finds no feature or characteristic of the airplane which renders it unsafe for the category in which it is certificated.

3.11 *Designation of applicable regulations.* The provisions of this section shall apply to all airplane types certificated under this part irrespective of the date of application for type certificate.

(a) Unless otherwise established by the Board, the airplane shall comply with the provisions of this part together with all amendments thereto effective on the date of application for type certificate, except that compliance with later effective amendments may be elected or required pursuant to paragraphs (c), (d), and (e) of this section.

(b) If the interval between the date of application for type certificate and the issuance of the corresponding type certificate exceeds three years, a new application for type certificate shall be required, except that for applications pending on May 1, 1954, such three-year period shall commence on that date. At the option of the applicant, a new application may be filed prior to the expiration of the three-year period. In either instance the applicable regulations shall be those effective on the date of the new application in accordance with paragraph (a) of this section.

(c) During the interval between filing the application and the issuance of a type certificate, the applicant may elect to show compliance with any amendment of this part which becomes effective during that interval, in which case all other amendments found by the Administrator to be directly related shall be complied with.

(d) Except as otherwise provided by the Board, or by the Administrator pursuant to section 1.24 of this subchapter, a change to a type certificate (see sec. 3.13 (b)) may be accomplished, at the option of the holder of the type certificate, either in accordance with the regulations incorporated by reference in the type certificate pursuant to section 3.13 (c), or in accordance with subsequent amendments to such regulations in effect on the date of application for approval of the change, subject to the following provisions:

(1) When the applicant elects to show compliance with an amendment to the regulations in effect on the date of application for approval of a change, he shall show compliance with all amendments which the Administrator finds are directly related to the particular amendment selected by the applicant.

(2) When the change consists of a new design or a substantially complete redesign of a component, equipment installation, or system installation of the airplane, and the Administrator finds that the regulations incorporated by reference in the type certificate pursuant to section 3.13 (c) do not provide complete standards with respect to such change, he shall require compliance with such provisions of the regulations in effect on the date of application for approval of the change as he finds will provide a level of safety equal to that established by the regulations incorporated by reference at the time of issuance of the type certificate.

NOTE: Examples of new or redesigned components and installations which might require compliance with regulations in effect on the date of application for approval, are: New powerplant installation which is likely to introduce additional fire or operational hazards unless additional protective measures are incorporated; the installation of an auto-pilot or a new electric power system.

(e) If changes listed in subparagraphs (1) through (3) of this paragraph are made, the airplane shall be considered as a new type, in which case a new application for type certificate shall be required and the regulations together with all amendments thereto effective on the date of the new application shall be made applicable in accordance with paragraphs (a), (b), (c), and (d) of this section.

(1) A change in the number of engines;

(2) A change to engines employing different principles of operation or propulsion;

(3) A change in design, configuration, power, or weight which the Administrator finds is so extensive as to require a substantially complete investigation of compliance with the regulations.

3.12 *Recording of applicable regulations.* **The Administrator, upon the issuance of a type certificate, shall record the applicable regulations with which compliance was demonstrated. Thereafter, the Administrator shall record the applicable regulations for each change in the type certificate which is accomplished in accordance with regulations other than those recorded at the time of issuance of the type certificate. (See sec. 3.11.)**

3.13 *Type certificate.*

(a) An applicant shall be issued a type certificate when he demonstrates the eligibility of the airplane by complying with the requirements of this part in addition to the applicable requirements in Part 1 of this subchapter.

(b) The type certificate shall be deemed to include the type design (see sec. 3.14 (b)), the operating limitations for the airplane (see sec. 3.737), and any other conditions or limitations prescribed by the regulations in this subchapter.

(c) The applicable provisions of this part recorded by the Administrator in accordance with section 3.12 shall be considered as incorporated in the type certificate as though set forth in full.

3.14 *Data required.*

(a) The applicant for a type certificate shall submit to the Administrator such descriptive data, test reports, and computations as are necessary to demonstrate that the airplane complies with the requirements of this part.

(b) The descriptive data required in paragraph (a) of this section shall be known as the type design and shall consist of such drawings and specifications as are necessary to disclose the configuration of the airplane and all the design features covered in the requirements of this part, such information on dimensions, materials, and processes as is necessary to define the structural strength of the airplane, and such other data as are necessary to permit by comparison the determination of the airworthiness of subsequent airplanes of the same type.

3.15 *Inspections and tests.* **Inspections and tests shall include all those found necessary by the Administrator to insure that the airplane complies with the applicable airworthiness requirements and conforms to the following:**

(a) All materials and products are in accordance with the specifications in the type design.

(b) All parts of the airplane are constructed in accordance with the drawings in the type design.

(c) All manufacturing processes, construction, and assembly are as specified in the type design.

3.16 *Flight tests.* **After proof of compliance with the structural requirements contained in this part, and upon completion of all necessary inspections and testing on the ground, and proof of the conformity of the airplane with the type design, and upon receipt from the applicant of a report of flight tests performed by him, the following shall be conducted:**

(a) Such official flight tests as the Administrator finds necessary to determine compliance with the requirements of this part.

(b) After the conclusion of flight tests specified in paragraph (a) of this section, such additional flight tests, on airplanes having a maximum certificated take-off weight of more than 6,000 pounds, as the Administrator finds necessary to ascertain whether there is reasonable assurance that the airplane, its components, and equipment are reliable and function properly. The extent of such additional flight tests shall depend upon the complexity of the airplane, the number and nature of new design features, and the record of previous tests and experience for the particular airplane type, its components, and equipment. If practicable, these flight tests shall be conducted on the same airplane used in the flight tests specified in paragraph (a) of this section.

3.16–1 *Accelerated service tests for aircraft having a maximum certificated take-off weight of more than 6,000 pounds (CAA policies which apply to sec. 3.16).*

(a) *Terms.* Terms used in this section are defined as follows:

(1) *T. C. Board.* The Type Certification Board set up by the CAA Field Offices for each new type aircraft project.

(2) *Routine CAR tests.* The flight tests prescribed in the regulations in this part to determine performance, flight characteristics, power plant characteristics, etc. (e. g. secs. 3.61 through 3.780), conducted in accordance with existing procedures.

(3) *Official functioning and reliability tests.* That portion of the fight tests conducted in showing compliance with the regulations quoted in subparagraph (2) of this paragraph, which is under the immediate supervision of the T. C. Board, as described in this section.

(4) *Supplementary experience.* Other flight tests and experience with an airplane type which is taken into consideration in establishing the extent of the official portion of the tests. This supplementary experience may be obtained by the manufacturer, military services, airlines, etc.

(5) *Simulated tests.* Tests on the ground or in an airplane of components and equipment under conditions simulating those likely to be obtained in service, which are taken into consideration in establishing the extent of the official portion of the tests.

(b) *Additional flight tests.* To satisfactorily accomplish the objectives of sec. 3.16 concerning additional flight tests and the extent thereof, the Administrator deems it necessary that:

(1) A comprehensive and systematic check be made in flight of the operation of all components to determine whether they "function properly", i. e., perform their intended function without introducing safety hazards.

(2) Sufficient testing and supplementary experience under actual, or a combination of simulated and actual, experience be obtained and evaluated to give reasonable assurance that the airplane is "reliable", i. e., should continue to function properly in service.

NOTE: In order to obtain wider experience, manufacturers should be encouraged to cooperate with airlines or other responsible operators in operating experimental airplanes of the same type under service conditions.

(3) Appropriate corrective action be taken when the need therefor is determined under subparagraph (1) or (2) of this paragraph.

NOTE: The CAA should be concerned only to the extent that the airplane can be operated safely under suitable inspection and maintenance procedures, but should not be concerned with maintenance costs.

(c) *Test program.* The Type Certification Board for each project should decide upon a proposed official test program at the time of the Preflight meeting of the Board (prior to the routine CAR flight tests) and coordinate this with the airplane manufacturer. At the conclusion of the routine CAR tests, the T. C. Board should meet again to review the experience gained in those tests, changes made in the design, and any additional supplementary experience, and to revise the proposed program accordingly.

(d) *Planning and execution of test program.* The following points should be considered:

(1) The test program should be sufficiently well planned to enable its execution in an efficient manner without overlooking important items.[1] The T. C. Board should review the design features and equipment with respect to the general objectives, and prepare a list showing:

(i) Components and system[2] to be checked in subparagraph (4) of this paragraph,

(ii) A brief description of the operations to be performed, where these are not obvious (referencing any necessary operating instructions),

(iii) Special checks or likely critical conditions,

(iv) Estimated flight time required.

(2) Allowances may be made for the functional tests already required by the routine CAR tests. Allowance may also be made for simulated testing of new features and equipment; however, the flight test program will be planned to determine the adequacy of the simulated tests (e. g. to determine whether the actual environmental conditions of temperature,[3] vibration, etc, are covered by the simulated tests) when these may be critical, and to determine whether the installation and connected systems are satisfactory. The T. C. Board will then make a consolidated estimate of the total flight time required, allowing for overlapping, and adjust this in accordance with the "test time" section outlined in paragraph (e) of this section

(3) The program will be arranged to per-

[1] It is not intended that the "paper work" be over-emphasized to the detriment of the practical results, and it should be reduced to a minimum wherever possible.

[2] Tests of anti-icing systems under actual icing conditions will in many cases be impracticable prior to type certification. A policy is in preparation regarding the approval and use of such systems in air-carrier operations. This will outline the flight testing required at various stages.

[3] This does not imply that flight tests must be conducted under the most severe outside air temperatures likely to be encountered in service. It should normally be possible to determine the effects of extreme outside temperatures on local temperatures by extrapolation or by suitable correction factors.

mit the Aviation Safety Agent in charge to become thoroughly familiar with the characteristics of the airplane, particularly those not specifically covered in the routine CAR tests.

(4) In accordance with paragraph (b) (1) of this section, all components of the airplane should be intensively[4] operated and studied under all operating conditions expected in service and obtainable within the time and geographic limitations of the tests. Particular attention will be given to the emergency procedures which would be required in the event of malfunctioning of any component, source of crew error, and overtaxing of crew abilities. This intensive type of testing should be conducted in all cases, but the length of time for which it is continued will depend upon the simulated and supplementary experience available for the particular type, as outlined in "test time" under paragraph (e) of this section.

(5) Ground inspections should be made at appropriate intervals during the test program to determine whether there are any failures or incipient failures in any of the components which might be a hazard to safe flight.

(6) When design changes are made during the course of the test, or when the official test airplane differs from those on which supplementary experience is obtained, or from modified versions of the same basic airplane type, the revised or modified items should be rechecked in accordance with the above procedure, but every effort should be made to include such items in the program in such a way as to avoid unduly extending the over-all test time. To this end, the Administrator may accept, in lieu of additional flight tests:

(i) Special tests of the original and revised components in which the conditions causing failure are intensified, and

(ii) Simulated tests of differing components.

(e) *Test time.* It is highly desirable that functioning and reliability test programs be administered uniformly in the sense that the program and flight time for a given project would be approximately the same regardless of which T. C. Board administered the project. This is difficult to achieve without establishing fixed arbitrary test times. However, such fixed arbitrary times would obviously be contrary to the intent of the Regulations. The following procedure with regard to establishing the required test time which permits considerable flexibility is, therefore, established for the guidance of T. C. Boards.

(1) The times suggested in this paragraph apply when supplementary experience is not taken into account, and are for airplanes which are conventional in regard to complexity and design features. Those times may be reduced to allow for supplementary experience, as outlined in subparagraph (2) of this paragraph, and for simulated testing, as outlined in paragraph (d) (1) of this section. In extreme cases of complexity[5] radically new design features, or difficulties in earlier flights, these times may be increased. Non-Transport (this part)—40 to 150 hours, depending on complexity.

(2) When satisfactory supplementary experience is available and taken into account, the following allowances should be used as a guide and applied with judgment in reducing the official flight test time determined in accordance with subparagraph (1) of this paragraph. However, in any case, the official program should provide sufficient time to accomplish the objective in paragraph (b) (1) of this section in accordance with paragraph (d) (3) and (4) of this section.

(i) *For intensive experience.* When the allowance is based on the total time of any one airplane in airline crew training and similar intensive operations, two hours of such operation may be considered equivalent to one hour of official testing.

(ii) *For miscellaneous experience.* When the allowance is based on the total time of any one airplane, five hours of such experience may be considered equivalent to one hour of official testing.

(iii) *Reduction for supplementary experience.* Whenever a reduction of official test time is desired on the basis of supplementary experience, such experience must be adequately

[4] Intensive operation means repeated operation of components in various sequences and combinations likely to occur in service.

[5] An example of extreme complexity would be transport intended for operation at 40,000 feet altitude, with automatic dive recovery flaps, turbos, variable jet exhaust, two speed cooling fans, retractable wind screens, automatic control of engine cooling, turbos, intercoolers, jet exhaust, etc. The test program for such an airplane might require as much as 300 hours if no supplementary experience were available.

recorded and submitted to the T. C. Board, as described in paragraph (f) of this section.

(f) *Reports and records.*

(1) A log should be kept of all flight tests, and accurate and complete records kept of the inspections made and of all defects, difficulties, and unusual characteristics and sources of crew error discovered during the tests, and of the recommendations made and action taken. Items for which design changes may be required will be reported to the manufacturer and the appropriate CAA engineering division.

(2) If supplementary experience is to be taken into account, similar records of such experience should be kept and submitted to the T. C. Board, together with a list of the differences between the airplane on which the experience was obtained and the official test airplane. When supplementary experience is obtained on a large fleet of airplanes (for example, military operations) of the same or a comparable type (see item 5 under Test Program), these records may consist of statistical summaries in lieu of complete records for each individual airplane.

(3) At the conclusion of the official tests, a summary report should be prepared by the T. C. Board and forwarded to Washington for inclusion in the Type Inspection Report.

(g) *Administration.* The CAA Aviation Safety Agent in charge should act as coordinator of all flight activities of the T. C. Board during the official program and the agent or an alternate designated by him will participate in all flights. He should collaborate with the manufacturers' pilots in all these activities, particularly in regard to flight plans and procedures. The manufacturers' pilot should be in command of all flights, but CAA pilots should fly the airplane at least sufficiently to accomplish paragraph (d) (3) of this section.

(1) Other CAA personnel (e. g. representatives of other Divisions and specialists) should participate in the flight tests when deemed necessary by the T. C. Board to accomplish the purposes of the tests.

(2) When supplementary experience is obtained in airline operations, a CAA Aviation Safety Agent should be assigned to follow the operations, review the operator's records, and supplement these by reports to the T. C. Board.

(h) *Test airplane.* Section 3.19 contains the phrase "If practicable, the flight tests * * * shall be conducted on the same airplane. * * *" This phrase will be liberally interpreted to facilitate completion of the type certification procedure. Thus, one airplane may be used for the official functioning and reliability tests while another airplane (or airplanes) is used for the routine CAR tests. In this case the test time on at least one airplane must be sufficient to accomplish the objective of paragraph (b) (2) of this section.

(i) *Modified types.* The procedure outlined above applies to new type designs. When a design employs components identical to those used in previous designs, credit may be given for the supplementary experience available for such components. When a design is modified (for example, several versions of the same basic type with different engines, propellers, etc.) the modified features and components should be treated in accordance with paragraph (d) (6) of this section.

(CAR, Supp. 10, 16 F. R. 3279, Apr. 14, 1951. Redesignated and amended by Supp. 14, 17 F. R. 9065, Oct. 11, 1952.)

3.17 *Airworthiness, experimental, and production certificates.* (For requirements with regard to these certificates see Part 1 of this subchapter.)

3.18 *Approval of materials, parts, processes, and appliances.*

(a) Materials, parts, processes, and appliances shall be approved upon a basis and in a manner found necessary by the Administrator to implement the pertinent provisions of the regulations in this subchapter. The Administrator may adopt and publish such specifications as he finds necessary to administer this regulation, and shall incorporate therein such portions of the aviation industry, Federal, and military specifications respecting such materials, parts, processes, and appliances as he finds appropriate.

NOTE: The provisions of this paragraph are intended to allow approval of materials, parts, processes, and appliances under the system of Technical Standard Orders, or in conjunction with type certification procedures for an airplane, or by any other form of approval by the Administrator.

(b) Any material, part, process, or appliance

shall be deemed to have met the requirements for approval when it meets the pertinent specifications adopted by the Administrator, and the manufacturer so certifies in a manner prescribed by the Administrator.

3.18–1 *Approval of materials, parts, processes, and appliances (CAA rules which apply to sec. 3.18).* Aircraft materials, parts, processes, and appliances made the subject of Technical Standard Orders shall be approved upon the basis and in the manner prescribed in Part 514[6] of this title, Technical Standard Orders—C-Series—Aircraft Components.

(CAR, Supp. 14, 17 F. R. 9065, Oct. 11, 1952.)

3.18–2 *Application of the Technical Standard Orders (TSO) System; C Series (CAA policies which apply to sec. 3.18).*

(a) *Purpose of Technical Standard Orders.* Technical Standard Orders are a means by which the Administrator adopts and publishes the specifications for which authority is provided in section 3.18 (a).

(b) *Applicability of Technical Standard Order requirements.*

(1) The applicability of and effective dates for TSO items are set forth in each TSO.

(2) Each Technical Standard Order sets forth the conditions under which materials, parts, processes, and applicances approved by the Administrator prior to establishment of an applicable TSO, may continue to be used in aircraft.

(3) The establishment of a Technical Standard Order for any product does not preclude the possibility of establishing the acceptability of a similar product as part of an aircraft, engine, or propeller, under the type certification or modification procedures, if there is established a level of safety equivalent to that provided in the regulations in this subchapter as implemented by the appropriate Technical Standard Order and the product is identified as part of the airplane, engine, or propeller.

(c) *Administration of the Technical Standard Order (TSO) system.* The principles which apply in administering the Technical Standard Order system are as follows:

[6] Copies of individual TSO's contained in Part 514 of this title are available upon application to the Aeronautical Reference Division, Attn. W–126, Civil Aeronautics Administration, Washington 25, D. C.

(1) Technical Standard Orders will reference performance provisions of recognized government specifications, or established industry specifications which have been found acceptable by the CAA. If no satisfactory specification exists, the Orders will include criteria prepared by the Administrator. In preparing criteria of this type, the Administrator will give consideration to recommendations made by the industry.

(2) Minimum performance requirements established by the Civil Aeronautics Administration and published in Technical Standard Orders will serve as a means by which materials, parts, processes, and appliances intended for use in certificated aircraft will be accepted.

(3) TSO's set forth the minimum requirements for safety. Every effort will be made by the CAA to keep the requirements at the minimum levels of safety and TSO's will not be used to set forth "desirable" standards.

(4) It will be the responsibility of the person submitting a statement of conformance to the CAA, certifying that his product meets the requirements of the TSO, to conduct the necessary tests demonstrating compliance therewith. This person will be held responsible for maintaining quality control adequate to assure that products which he guarantees to meet the requirements of a TSO do, in fact, meet these standards. The CAA will not formally approve such products as meeting the requirements of TSO's nor exercise direct inspection control over them. The statement of conformance with the provisions of a Technical Standard Order normally will be accepted by the CAA as sufficient indication that the applicable requirements have been fulfilled.

Any TSO item which is modified must continue to comply with the requirements of the TSO; and the person authorizing the modification will be responsible for such compliance.

(d) *Numbering of Technical Standard Orders.* Each Technical Standard Order will be assigned a designation consisting of the letters "TSO," a series code letter "C" indicating aircraft materials, parts, processes, and appliances, and a serial number to be assigned in sequence for each of the TSO's issued in the "C" series, e. g., TSO–C–1, "Smoke Detectors." Revisions are indicated by the addition of letters a, b, c, etc., after the number.

(CAR, Supp. 14, 17 F. R. 9065, Oct. 11, 1952.)

[3.18–3 *Manufacturer (CAA interpretations which apply to sec. 3.18 (b)).* For the purpose of accepting a statement of conformance for a Technical Standard Order Product, the word "manufacturer" is interpreted to mean a person who designs and fabricates by welding, cutting, drilling, forming, bolting, riveting, glueing, soldering, sewing, etc., a product. This includes products which are composed in whole or in part of components of TSO products. One who merely cleans products or appliances or repairs them by replacing standard parts and/or by replacing components with identical ones is not considered to be the manufacturer. Nor is the distributor of another person's completed product considered to be the manufacturer.]

(23 F. R. 10323, Dec. 25, 1958, effective Jan. 31, 1959.)

[3.18–4 *Products approved as a part of the airplane (CAA policies which apply to sec. 3.18).*

[(a) A material, part or appliance (hereinafter called product) may be approved as a part of the airplane type design under a type certificate or a supplemental type certificate.

[(1) If a Technical Standard Order covering the product is in effect, the applicant for approval should submit type design data showing that the product meets the performance standards of the Technical Standard Order except that:

[(i) Deviations from the performance standards may be allowed when the applicant for, or holder of, the type certificate or the supplemental type certificate substantiates that other performance standards are applicable for the product as installed in the airplane. Any deviation from standards prescribed in this part may be allowed only in accordance with section 3.10.

[(2) Where no TSO covering the product exists, the applicant for approval should submit type design data showing compliance with all the requirements of this part which are applicable to the product.

[(3) Products previously approved by the CAA by means of letters of approval, Repair and Alteration Form ACA–337, or listing on CAA Product and Process Specifications will continue to be eligible for installation in aircraft unless the eligibility is restricted by applicable regulations or airworthiness directives issued under section 1.24 of this subchapter.

[(b) Products approved as a part of the airplane type design under a type certificate should be identified by an airplane part number on the approved drawing list.

[(c) Products approved as a part of the airplane under a supplemental type certificate should be identified by a part or drawing number on such certificate.

[(d) Each TSO product that is approved as a part of the airplane should have the TSO identification removed and be identified as set forth in paragraph (b) or (c) of this section, whichever is applicable.]

(23 F. R. 10323, Dec. 25, 1958, effective Jan. 31, 1959.)

3.19 *Changes in type design.* (For requirements with regard to changes in type design and the designation of applicable regulations therefor, see secs. 3.11 (d) and (e) and Part 1 of this subchapter.)

3.19–1 *Changes of engines (CAA policies which apply to sec. 3.19).*

(a) There are currently available newly designed engines of approximately the same size and weight as previously designed engines, but with considerable variations in power. It is possible to interchange these engines with little or no installation changes, and although minor changes in engine weight may be involved, it will still be practical to operate the aircraft at the originally approved gross weight. Under section 3.185, the maneuvering load factor is not dependent upon engine power, and under section 3.184 the design airspeeds can be independent of engine power. Therefore, a change which involves or permits a practical power increase by exchange of engines shall be approved by the Administrator: *Provided,* That such exchange of engines is not accompanied by an increase in the gross weight of the aircraft, or an increase in placard speeds. Under those conditions it will not be necessary to restrict the maximum continuous horsepower by a placard because of the airplane speed limitations since the latter are indicated on the speed placards.

(b) Aircraft alterations involving weight or speed changes beyond those set forth above will be approved by the Administrator, if the applicant shows compliance with the applicable airworthiness requirements.

(c) Under section 3.19, it will be necessary to require such investigations of local structure, weight and balance, power plant installations and flight tests as are normally involved in a change of engine type. However, every effort will be made by reference to data already on hand to minimize the amount of testing and structural analysis required of the applicant.

(CAR, Supp. 10, 16 F. R. 3281, Apr. 1951. Redesignated and amended by Supp. 14, 17 F. R. 9065, Oct. 11, 1952.)

Airplane Categories

3.20 ***Airplane categories.***

(a) For the purpose of certification under this part, airplanes are divided upon the basis of their intended operation into the following categories:

(1) *Normal-suffix N.* Airplanes in this category are intended for nonacrobatic, nonscheduled passenger, and nonscheduled cargo operation.

(2) *Utility-suffix U.* Airplanes in this category are intended for normal operations and limited acrobatic maneuvers. These airplanes are not suited for use in snap or inverted maneuvers.

NOTE: The following interpretation of paragraph (a) (2) was issued May 15, 1947, 12 F. R. 3434: The phrase "limited acrobatic maneuvers" as used in section 3.6 (now sec. 3.20) is interpreted to include steep turns, spins, stalls (except whip stalls), lazy eights, and chandelles.

(3) *Acrobatic-suffix A.* Airplanes in this category will have no specific restrictions as to type of maneuver permitted unless the necessity therefor is disclosed by the required flight tests.

(b) An airplane may be certificated under the requirements of a particular category, or in more than one category, provided that all of the requirements of each such category are met. Sections of this part which apply to only one or more, but not all, categories are identified in this part by the appropriate suffixes added to the section number, as indicated in paragraph (a) of this section. All sections not identified by a suffix are applicable to all categories except as otherwise specified.

3.20-1 *Approved maneuvers for normal category aircraft (CAA interpretations which apply to sec. 3.20).* The phrase "nonacrobatic operation" as used in section 3.20 (a) (1) is interpreted to mean that type of operation in which the aircraft is limited to those maneuvers incidental to normal flying and including stalls (except whip stalls) and turns in which the angle of bank is not in excess of 60°.

(CAR, Supp. 10, 16 F. R. 3278, Apr. 14, 1951. Redesignated and amended by Supp. 14, 17 F. R. 9065, Oct. 11, 1952.)

3.20-2 *Approved limited acrobatic maneuvers for utility category aircraft (CAA interpretations which apply to sec. 3.20).* The phrase "limited acrobatic maneuvers" as used in section 3.20 (a) (2) is interpreted to include spins (where approved for the particular type airplane) lazy eights, chandelles and steep turns in which the angle of bank is in excess of 60°. It is recognized that aircraft in this category are also capable of performing all normal maneuvers listed in section 3.20–1 for normal category aircraft. Although it is possible in many airplanes to perform other acrobatic maneuvers, such as loops, without exceeding airspeed and strength limitations, inexperienced or uninstructed pilots are likely to get into difficulty. It is therefore considered unwise to label such maneuvers "approved" in the Airplane Flight Manual

(CAR, Supp. 10, 16 F. R. 3278, Apr. 14, 1951. Redesignated and amended by Supp. 14, 17 F. R. 9065, Oct. 11, 1952.)

Subpart B—Flight Requirements

General

3.61 *Policy re proof of compliance.* Compliance with the requirements specified in this subpart governing functional characteristics shall be demonstrated by suitable flight or other tests conducted upon an airplane of the type, or by calculations based upon the test data referred to above, provided that the results so obtained are substantially equal in accuracy to the results of direct testing. Compliance with each requirement must be provided at the critical combination of airplane weight and center of gravity position within the range of either for which certification is desired. Such compliance must be demonstrated by systematic investigation of all probable weight and center of gravity combinations or must be reasonably inferable from such as are investigated.

3.62 *Flight test pilot.* The applicant shall provide a person holding an appropriate pilot certificate to make the flight tests, but a designated representative of the Administrator may pilot the airplane insofar as that may be necessary for the determination of compliance with the airworthiness requirements.

3.63 *Noncompliance with test requirements.* Official type tests will be discontinued until corrective measures have been taken by the applicant when either:

(a) The applicant's test pilot is unable or unwilling to conduct any of the required flight tests; or

(b) Items of noncompliance with requirements are found which may render additional test data meaningless or are of such nature as to make further testing unduly hazardous.

3.64 *Emergency egress.* Adequate provisions shall be made for emergency egress and use of parachutes by members of the crew during the flight tests.

3.65 *Report.* The applicant shall submit to the representative of the Administrator a report covering all computations and tests required in connection with calibration of instruments used for test purposes and correction of test results to standard atmospheric conditions. The representative of the Administrator will conduct any flight tests which he finds to be necessary in order to check the calibration and correction report.

Weight Range and Center of Gravity

3.71 *Weight and balance.*

(a) There shall be established, as a part of the type inspection, ranges of weight and center of gravity within which the airplane may be safely operated.

(b) When low fuel adversely affects balance or stability, the airplane shall be so tested as to simulate the condition existing when the amount of usable fuel on board does not exceed 1 gallon for every 12 maximum continuous horsepower of the engine or engines installed.

3.71–1 *Weight and balance limitations for flight tests (CAA policies which apply to sec. 3.71 (a)).*

(a) Flight tests should be conducted at the maximum weight for which the airplane is to be certificated and at no time during the test should the weight exceed the following tolerances from the maximum weight:

Item	*Tolerance (percent)*
General	+5; −10.
Flight characteristics, general	+5; −10.
Flight characteristics, critical items affected by weight	+5; −1.

(b) The forward and rearward center of gravity during flight test loading should be within a tolerance of 7 percent of the total travel for which the airplane is to be certificated.

(a) the extremes selected by the applicant, or (b) the extremes for which the structure has been proven, or (c) the extremes for which compliance with all functional requirements were demonstrated, loading instructions shall be provided in the Airplane Flight Manual as specified in sections 3.777–3.780.

3.76–1 *Center of gravity position (CAA policies which apply to sec. 3.76).*

(a) It is suggested that as wide a range of c. g. as practicable be investigated (using ballast if necessary) in the flight tests to provide for future changes in empty weight c. g. without rerunning tests or structural analysis.

(b) Where practicable, the extreme c. g. positions should be investigated, both in structural design and flight tests in combination with maximum weight (using ballast if necessary) to make loading instructions as simple as possible, and also provide for future changes in empty weight c. g. and useful load.

(c) In cases where the permissible c. g., positions vary with maximum weight, it is suggested that a note be included in the loading instruction portion of the Airplane Flight Manual advising owners to contact the airplane manufacturer for new loading instructions when any change is made to the airplane which would appreciably affect the location of the empty weight c. g. or the useful load.

(Supp. 10, 16 F. R. 3283, Apr. 14, 1951.)

Performance Requirements

General

3.80 ***Alternate performance requirements.*** **The provisions of sections 3.84, 3.85, 3.86, and 3.112 (a) (2) (ii) shall not be applicable to airplanes having a maximum certificated take-off weight of 6,000 pounds or less. In lieu thereof, such airplanes shall comply with the provisions of sections 3.84a, 3.85a, 3.87, and 3.112 (c).**

3.81 ***Performance.*** **The following items of performance shall be determined and the airplane shall comply with the corresponding requirements in standard atmosphere and still air.**

3.82 ***Definition of stalling speeds.***

(a) V_{s_0} denotes the true indicated stalling speed, if obtainable, or the minimum steady flight speed at which the airplane is controllable, in miles per hour, with:

(1) Engines idling, throttles closed (or not more than sufficient power for zero thrust),

(2) Propellers in position normally used for take-off,

(3) Landing gear extended,

(4) Wing flaps in the landing position,

(5) Cowl flaps closed,

(6) Center of gravity in the most unfavorable position within the allowable landing range,

(7) The weight of the airplane equal to the weight in connection with which V_{s_0} is being used as a factor to determine a required performance.

(b) V_{s_1} denotes the true indicated stalling speed, if obtainable, otherwise the calculated value in miles per hour, with:

(1) Engines idling, throttles closed (or not more than sufficient power for zero thrust),

(2) Propellers in position normally used for take-off, the airplane in all other respects (flaps, landing gear, etc.) in the particular condition existing in the particular test in connection with which V_{s_1} is being used,

(3) The weight of the airplane equal to the weight in connection with which V_{s_1} is being used as a factor to determine a required performance.

(c) These speeds shall be determined by flight tests using the procedure outlined in section 3.120.

3.82–1 *"Zero thrust" (CAA interpretations which apply to sec. 3.82).* As used in section 3.82 (a) (1) and (b) (1) the term "zero thrust" contained in the phrase "engines idling, throttles closed (or not more than sufficient power for zero thrust)" is interpreted to permit "zero thrust at a speed not greater than 110 percent of the stalling speed."

(Supp. 1, 12 F. R. 3434, May 28, 1947, as amended by Amdt. 1, 14 F. R. 36, Jan. 5, 1949.)

3.83 ***Stalling speed.*** **V_{s_0} at maximum weight shall not exceed 70 miles per hour for (1) single-engine airplanes and (2) multiengine airplanes which do not have the rate of climb with critical engine inoperative specified in section 3.85 (b).**

Takeoff

3.84 ***Takeoff.***

(a) The distance required to take off and climb over a 50-foot obstacle shall be determined under the following conditions:

(1) Most unfavorable combination of weight and center of gravity location,

(2) Engines operating within the approved limitations,

(3) Cowl flaps in the position normally used for takeoff.

(b) Upon obtaining a height of 50 feet above the level takeoff surface, the airplane shall have attained a speed of not less than 1.3 V_{s_1} unless a lower speed of not less than V_x plus 5 can be shown to be safe under all conditions, including turbulence and complete engine failure.

(c) The distance so obtained, the type of surface from which made, and the pertinent information with respect to the cowl flap position, the use of flight-path control devices and landing gear retraction system shall be entered in the Airplane Flight Manual. The takeoff shall be made in such a manner that its reproduction shall not require an exceptional degree of skill on the part of the pilot or exceptionally favorable conditions.

3.84–1 *Takeoff performance (CAA policies which apply to sec. 3.84).* To meet the requirements of section 3.84 pertaining to certification of takeoff performance and to provide the Airplane Flight Manual performance data required in section 3.780 (a) (3) and (4), the following procedure may be used during official type tests:

(a) The ground and climb distances may be determined separately and the corrected data pieced together (as is now done in the transport category). Thus, for the simplest procedure, the airplane would be accelerated on (or near) the ground with gear extended to a speed not less than $1.3V_{s_1}$, and a climb segment to the 50-foot height point with gear extended would be determined by saw-tooth climb data. If it is desired to assume retraction of the landing gear at an earlier point, such point should be assumed to occur not earlier than that which would be used in normal takeoffs. The acceleration to $1.3V_{s_1}$ should then be measured as above, with gear retraction being initiated at the selected speed. If gear retraction is completed before reaching $1.3V_{s_1}$, only one climb segment, with gear retracted, need be determined. If retraction is not completed during acceleration to $1.3V_{s_1}$, two climb segments should be determined; one with gear extended for the time period necessary to complete retraction; the second with gear retracted. The acceleration segment should be determined photographically, and a minimum of three trials should be made up to speeds equal to or greater then $1.3V_{s_1}$.

NOTE: (CAA camera equipment may be obtained on a loan basis).

(b) Based upon the CAA's experience to date, the test method outlined in paragraph (a) of this section has given the desired accuracy of results. It also provides suitable means for showing the approximate calculated effect of temperature and altitude upon climb (up to 7,000 feet).

NOTE: It is permissible for other methods to be used in accomplishing these tests, providing that any method used is one which the average pilot may be reasonably expected to duplicate without use of unusual skill or experience, and one which produces equivalent accuracy. The operating procedure which must be followed to achieve the measured performance should in all cases be described in the Airplane Flight Manual.

(c) The takeoff and climb requirements of sections 3.84 and 3.85 were written to assure the airplane's ability to clear obstacles in the vicinity of the airport. Consequently, the wing flap used for the airborne portion of the takeoff to the 50-foot height should not exceed that used for the "normal climb condition" of section 3.85 (a). However, if the applicant so desires, he may enter additional takeoff data in the Airplane Flight Manual in which the flap setting specified in section 3.84 or section 3.85 (a) has been exceeded, provided the portion of the flight path beyond the 50-foot point which will cover the transition to normal climb configuration of section 3.85 (a), is also included.

(Supp. 10, 16 F. R. 3283, Apr. 14, 1951.)

3.84–2 *Measurement of seaplane takeoff distances (CAA interpretations which apply to sec. 3.84 (a)).* The standard starting point for the measurement of seaplane takeoff distances may be assumed to be the point at which the seaplane has attained an initial speed of three miles per hour during takeoff.

(Supp. 10, 16 F. R. 3283, Apr. 14, 1951.)

3.84–3 *Takeoff speed (CAA interpretations which apply to sec. 3.84 (b)).* $1.3 \times V_{s_1}$ or V_x+5 speed should be used for takeoff even if throttling back is necessary to prevent exceeding r. p. m. limits.

(Supp. 10, 16 F. R. 3283, Apr. 14, 1951.)

3.84a *Takeoff requirements; airplanes of 6,000 lbs. or less.* Airplanes having a maximum certificated takeoff weight of 6,000 lbs. or less shall comply with the provisions of this section.

(a) The elevator control for tail wheel type airplanes shall be sufficient to maintain at a speed equal to 0.8 V_{s_1} an airplane attitude which will permit holding the airplane on the runway until a safe takeoff speed is attained.

(b) The elevator control for nose wheel type airplanes shall be sufficient to raise the nose wheel clear of the takeoff surface at a speed equal to 0.85 V_{s_1}.

(c) The characteristics prescribed in paragraphs (a) and (b) of this section shall be demonstrated with:

(1) Takeoff power,

(2) Most unfavorable weight,

(3) Most unfavorable c. g. position.

(d) It shall be demonstrated that the airplane will take off safely without requiring an exceptional degree of piloting skill.

Climb

3.85 *Climb.*

(a) *Normal climb condition.* The steady rate of climb at sea level shall be at least 300 feet per minute, and the steady angle of climb at least 1:12 for landplanes or 1:15 for seaplanes with:

(1) Not more than maximum continuous power on all engines,

(2) Landing gear fully retracted,

(3) Wing flaps in take-off position,

(4) Cowl flaps in the position used in cooling tests specified in sections 3.581–3.596.

(b) *Climb with inoperative engine.* All multiengine airplanes having a stalling speed V_{s_0} greater than 70 miles per hour or a maximum weight greater than 6,000 pounds shall have a steady rate of climb of at least 0.02 $V_{s_0}^2$ *in feet per minute at an altitude of 5,000 feet* with the critical engine inoperative and:

(1) The remaining engines operating at not more than maximum continuous power,

(2) The inoperative propeller in the minimum drag position,

(3) *Landing gear retracted,*

(4) Wing flaps in the most favorable position,

(5) Cowl flaps *in the position used in cooling* tests specified in sections 3.581–3.596.

(c) *Balked landing conditions.* The steady *angle of climb at sea level shall be at least* 1:30 with:

(1) Take-off power on all engines,

(2) *Landing gear extended,*

(3) Wing flaps in landing position.

If rapid retraction is possible with safety without loss of altitude and without requiring sudden changes of angle of attack or exceptional skill on the part of the pilot, wing flaps may be retracted.

3.85–1 *Rate of climb (CAA policies which apply to sec. 3.85).* To meet the requirements of section 3.85 it is necessary that a suitable method be employed for the purpose of determining the rates of climb. The Administrator will accept the following procedure for this purpose:

This method of obtaining rates of climb is through the derivation of a polar curve obtained from a series of saw-tooth climbs at various speeds. When saw-tooth climbs are employed, a minimum of five different speeds is required. However, demonstration climbs to prove the article meets the minimum climb requirement may be made at one given air speed. In such cases, the minimum number of climbs at one air speed shall be not less than three. This may not be interpreted to mean the best three of a number of climbs. In the event additional climbs are made the average of the total shall be the value to be accepted. It shall be permissible, however, to discard any climbs which are obviously in error due to such factors as turbulent air.

(Supp. 1, 12 F. R. 3434, May 28, 1947, as amended by Amdt. 1, 14 F. R. 36, Jan. 5, 1949.)

3.85–2 *"Normal climb" and "cooling test procedure for single-engine airplanes" (CAA interpretations which apply to sec. 3.85).* In connection with any application to have an aircraft certified for airworthiness under a combination of the requirements of this part and Part 4a of this subchapter as authorized by the provisions of section 3.2, the items of "normal climb" (sec. 3.85 (a)) and "cooling test procedure for single-engine airplanes" (sec. 3.586), shall be construed by the Administrator as "related items."

(Supp. 1, 12 F. R. 3435, May 28, 1947, as amended by Amdt. 1, 14 F. R. 36, Jan. 5, 1949.)

3.85–3 *"Rapid retraction" (CAA interpretations which apply to sec. 3.85).* The Administrator will consider retraction of flaps in 2 seconds or less as compliance with the factor of "rapid retraction" as that phrase is used in section 3.85 (c).

(Supp. 1, 12 F. R. 3435, May 28, 1947, as amended by Amdt. 1, 14 F. R. 36, Jan. 5, 1949.)

3.85–4 *Weight for items of performance and flight characteristics (CAA interpretations which apply to sec. 3.85).* For multiengine airplanes in which the design landing weight (sec. 3.242) is less than the maximum weight (sec. 3.74) for which certification is desired, the weight for items of performance and flight characteristics shall be construed by the Administrator as the maximum weight defined in section 3.74. Such items of performance and flight characteristics shall consist of balked landing (climb) conditions (sec. 3.74), landing over 50-foot obstacles (sec. 3.86), and all flight characteristics tests in the landing configuration. The design weight covered in section 3.242 is intended for use for structural design purposes only.

(Supp. 1, 12 F. R. 3435, May 28, 1947, as amended by Amdt. 1, 14 F. R. 36, Jan. 5, 1949.)

3.85–5 *Low-pitch propeller setting in normal climb position (CAA interpretations which apply to sec. 3.85 (a)).*

(a) In the event an airplane has:

(1) An engine for which the takeoff and maximum continuous power ratings are identical, and

(2) A fixed-pitch, two-position or similar type propeller,

then the regulations provide that the best rate of climb speed specified in section 3.85 (a) for normal climb should be determined with the low-pitch propeller setting which would restrain the engine to an r. p. m. at full throttle not exceeding its permissible takeoff r. p. m. (see sec. 3.419 (a)).

(b) A relaxation of the propeller pitch setting requirement stipulated by section 3.419 (a) may be granted, however, for an airplane falling into the foregoing classification, when it shows a marginal item of performance as, for example, when it can meet the rate of climb requirement of section 3.85 (a) for normal climb, but may have difficulty in meeting the angle of climb requirements of section 3.85 (a) for normal climb and/or section 3.85 (c) for balked landing. In this case, it will be permissible to use a lower propeller pitch setting than specified in section 3.419 (a), in order to obtain rated engine r. p. m. at the best angle of climb speed: *Provided,* Acceptable engine cooling can be demonstrated at the lower speed associated with the best angle of climb. In employing this procedure, consideration should also be given to the following:

(1) That the best angle of climb speed for the balked landing condition may be considerably lower than the best angle of climb speed for the normal climb condition.

(2) That as a result of subparagraph (1) of this paragraph, the engine would normally have to be part throttled to avoid exceeding rated r. p. m. at the higher speeds, and would therefore develop less than rated power for showing compliance with the normal climb and takeoff requirements of sections 3.85 (a) and 3.84, respectively.

(Supp. 10, 16 F. R. 3283, Apr. 14, 1951.)

3.85a *Climb requirements; airplanes of 6,000 lbs. or less.* Airplanes having a maximum certificated take-off weight of 6,000 lbs. or less shall comply with the requirements of this section.

(a) *Climb; take-off climb condition.* The steady rate of climb at sea level shall not be less than 10 V_{s_1} or 300 feet per minute, whichever is the greater, with:

(1) Take-off power,

(2) Landing gear extended,

(3) Wing flaps in take-off position,

(4) Cowl flaps in the position used in cooling tests specified in sections 3.581 through 3.596.

(b) *Climb with inoperative engine.* All multiengine airplanes having a stalling speed

V_{s0} greater than 70 miles per hour shall have a steady rate of climb of at least 0.02 V_{s0}^2 in feet per minute at an altitude of 5,000 feet with the critical engine inoperative and:

(1) The remaining engines operating at not more than maximum continuous power,

(2) The inoperative propeller in the minimum drag position,

(3) Landing gear retracted,

(4) Wing flaps in the most favorable position,

(5) Cowl flaps in the position used in cooling tests specified in sections 3.581 through 3.596.

(c) *Climb; balked landing conditions.* The steady rate of climb at sea level shall not be less than 5 V_{s0} or 200 feet per minute, whichever is the greater, with:

(1) Take-off power,

(2) Landing gear extended,

(3) Wing flaps in the landing position. If rapid retraction is possible with safety, without loss of altitude and without requiring sudden changes of angle of attack or exceptional skill on the part of the pilot, wing flaps may be retracted.

Landing

3.86 *Landing.*

(a) The horizontal distance required to land and to come to a complete stop (to a speed of approximately 3 miles per hour for seaplanes or float planes) from a point at a height of 50 feet above the landing surface shall be determined as follows:

(1) Immediately prior to reaching the 50-foot altitude, a steady gliding approach shall have been maintained, with a true indicated air speed of at least 1.3 V_{s0}.

(2) The landing shall be made in such a manner that there is no excessive vertical acceleration, no tendency to bounce, nose over, ground loop, porpoise, or water loop, and in such a manner that its reproduction shall not require any exceptional degree of skill on the part of the pilot or exceptionally favorable conditions.

(b) The distance so obtained, the type of landing surface on which made and the pertinent information with respect to cowl flap position, and the use of flight path control devices shall be entered in the Airplane Flight Manual.

3.86-1 *Landing distances (CAA policies which apply to sec. 3.86).* The Administrator will not approve the use of landing distances obtainable with reverse-thrust propellers in establishing landing field lengths until such time as sufficient experience with their use is available for proper consideration of all related factors involved in the establishment of adequate airport lengths for routine landings.

(Supp. 1, 12 F. R. 3437, May 28, 1947, as amended by Amdt. 1, 14 F. R. 36, Jan. 5, 1949.)

3.86-2 *Use of camera equipment (CAA policies which apply to sec. 3.86).* The landing distance should be determined photographically. CAA camera equipment is available on a loan basis.

(Supp. 10, 16 F. R. 3284, Apr. 14, 1951.)

3.87 *Landing requirements; airplanes of 6,000 lbs. or less.* For an airplane having a maximum certificated take-off weight of 6,000 lbs. or less it shall be demonstrated that the airplane can be safely landed and brought to a stop without requiring an exceptional degree of piloting skill, and without excessive vertical acceleration, tendency to bounce, nose over, ground loop, porpoise, or water loop.

Flight Characteristics

3.105 *Requirements.* The airplane shall meet the requirements set forth in sections 3.106 to 3.124 at all normally expected operating altitudes under all critical loading conditions within the range of center of gravity and, except as otherwise specified, at the maximum weight for which certification is sought.

Controllability

3.106 *General.* The airplane shall be satisfactorily controllable and maneuverable during take-off, climb, level flight, dive, and landing with or without power. It shall be possible to make a smooth transition from one flight condition to another, including turns and slips, without requiring an exceptional degree of skill, alertness, or strength on the part of the pilot, and without danger of exceeding the limit load factor under all conditions of operation probable for the type, including for multiengine airplanes those conditions normally encountered in the event of sudden failure of any engine. Compliance with "strength of pilots" limits need not be demonstrated by quantitative tests unless the Administrator finds the condition to be marginal. In the latter case they shall not exceed maximum values found by the Administrator to be appropriate for the type but in no case shall they exceed the following limits:

	Pitch	Roll	Yaw
(a) For temporary application:			
Stick	60	30	150
Wheel [1]	75	60	150
(b) For prolonged application	10	5	20

[1] Applied to rim.

3.107–U *Approved acrobatic maneuvers.* It shall be demonstrated that the approved acrobatic maneuvers can be performed safely. Safe entry speeds shall be determined for these maneuvers.

3.108–A *Acrobatic maneuvers.*

It shall be demonstrated that acrobatic maneuvers can be performed readily and safely. Safe entry speeds shall be determined for these maneuvers.

3.109 *Longitudinal control.* The airplane shall be demonstrated to comply with the following requirements:

(a) It shall be possible at all speeds below V_x to pitch the nose downward so that the rate of increase in air speed is satisfactory for prompt acceleration to V_x with:

(1) Maximum continuous power on all engines, the airplane trimmed at V_x.

(2) Power off, airplanes of more than 6,000 pounds maximum weight trimmed at 1.4 V_{s_1}, and airplanes of 6,000 pounds or less maximum weight trimmed at 1.5 V_{s_1}.

(3) (i) Wing flaps and landing gear extended and

(ii) Wing flaps and landing gear retracted.

(b) During each of the controllability demonstrations outlined below it shall not require a change in the trim control or the exertion of more control force than can be readily applied with one hand for a short period. Each maneuver shall be performed witht he landing gear extended.

(1) With power off, flaps retracted, and the airplane trimmed as prescribed in paragraph (a) (2) of this section, the flaps shall be extended as rapidly as possible while maintaining the air speed at approximately 40 percent above the instantaneous value of the stalling speed.

(2) Same as subparagraph (1) of this paragraph, except the flaps shall be initially extended and the airplane trimmed as prescribed in paragraph (a) (2) of this section, then the flaps shall be retracted as rapidly as possible.

(3) Same as subparagraph (2) of this paragraph, except maximum continuous power shall be used.

(4) With power off, the flaps retracted, and the airplane trimmed as prescribed in paragraph (a) (2) of this section, take-off power shall be applied quickly while the same air speed is maintained.

(5) Same as subparagraph (4) of this paragraph, except with the flaps extended.

(6) With power off, flaps extended, and the airplane trimmed as prescribed in paragraph (a) (2) of this section, air speeds within the range of 1.1 V_{s_1} to 1.7 V_{s_1} or V_f, whichever is the lesser, shall be obtained and maintained.

(c) It shall be possible without the use of exceptional piloting skill to maintain essentially level flight when flap retraction from any position is initiated during steady horizontal flight at 1.1 V_{s_1} with simultaneous application of not more than maximum continuous power.

3.110 *Lateral and directional control.*

(a) It shall be possible with multiengine airplanes to execute 15-degree banked turns both with and against the inoperative engine from steady climb at 1.4 V_{s_1} or V_y for the condition with:

(1) Maximum continuous power on the operating engines,

(2) Rearmost center of gravity,

(3) (i) Landing gear retracted and

(ii) Landing gear extended.

(4) Wing flaps in most favorable climb position.

(5) Maximum weight.

(6) The inoperative propeller in its minimum drag condition.

(b) It shall be possible with multiengine airplanes, while holding the wings level laterally within 5 degrees, to execute sudden changes in heading in both directions without dangerous characteristics being encountered. This shall be demonstrated at $1.4V_{s_1}$ or V_y up to heading changes of 15 degrees, except that the heading change at which the rudder force corresponds to that specified in section 3.106 need not be exceeded, with:

(1) The critical engine inoperative,

(2) Maximum continuous power on the operating engine(s),

(3) (i) Landing gear retracted and

(ii) Landing gear extended,

(4) Wing flaps in the most favorable climb position,

(5) The inoperative propeller in its minimum drag condition,

(6) The airplane center of gravity at its rearmost position.

3.111 *Minimum control speed* (V_{mc}).

(a) A minimum speed shall be determined under the conditions specified below, such that when any one engine is suddenly made inoperative at that speed, it shall be possible to recover control of the airplane, with the one engine still inoperative, and to maintain it in straight flight at that speed, either with zero yaw or, at the option of the applicant, with a bank not in excess of 5 degrees. Such speed shall not exceed [1.2 V_{s_1}], with:

(1) Take-off or maximum available power on all engines,

(2) Rearmost center of gravity,

(3) Flaps in take-off position,

(4) Landing gear retracted.

(b) In demonstrating this minimum speed, the rudder force required to maintain it shall not exceed forces specified in section 3.106, nor shall it be necessary to throttle the remaining engines. During recovery the airplane shall not assume any dangerous attitude, nor shall it require exceptional skill, strength, or alertness on the part of the pilot to prevent a change of heading in excess of 20 degrees before recovery is complete.

Trim

3.112 *Requirements.*

(a) The means used for trimming the airplane shall be such that, after being trimmed and without further pressure upon or movement of either the primary control or its corresponding trim control by the pilot or the automatic pilot, the airplane will maintain:

(1) Lateral and directional trim in level flight at a speed of 0.9 V_h or at V_c, if lower, with the landing gear and wing flaps retracted;

(2) Longitudinal trim under the following conditions:

(i) During a climb with maximum continuous power at a speed between V_x and 1.4 V_{s_1}.

(*a*) With landing gear retracted and wing flaps retracted,

(*b*) With landing gear retracted and wing flaps in the take-off position.

(ii) During a glide with power off at a speed not in excess of 1.4 V_{s_1},

(*a*) With landing gear extended and wing flaps retracted,

(*b*) With landing gear extended and wing flaps extended under the forward center of gravity position approved with the maximum authorized weight.

(*c*) With landing gear extended and wing flaps extended under the most forward center of gravity position approved, regardless of weight.

(iii) During level flight at any speed from 0.9 V_h to V_x or 1.4 V_{s_1} with landing gear and wing flaps retracted.

(b) In addition to the above, multi-engine airplanes shall maintain longitudinal and directional trim at a speed between V_y and 1.4 V_{s_1} during climbing flight with the critical of two or more engines inoperative, with:

(1) The other engine(s) operating at maximum continuous power,

(2) The landing gear retracted,

(3) Wing flaps retracted,

(4) Bank not in excess of 5 degrees.

(c) For aircraft having a maximum certificated take-off weight of 6,000 lbs. or less,

the value specified in paragraph (a) (2) (ii) of this section shall be 1.5 V_{s_1} or, if the stalling speed V_{s_1} is not obtainable in the particular configuration, 1.5 times the minimum steady flight speed at which the airplane is controllable.

3.112–1 *Trim during a glide (CAA policies which apply to sec. 3.112).* The following performance standards will be used for the purpose of administering section 3.112 (a) (2) (ii):

(a) In the case of new airplane designs which, due to their being equipped with high lift devices, cannot meet the required trim at 1.4 times stall speed with the landing gear and flaps extended, the Administrator, as authorized in section 3.1, will accept, as being of equivalent safety, performance with the flaps extended based on the following standards:

(1) The flap-down power-off stalling speed shall not exceed 90 percent of the flap-retracted power-off stalling speed.

(2) The minimum trim speed with power off, flaps and landing gear extended, under the forward center of gravity position approved with the maximum authorized weight, and under the most forward center of gravity position approved, regardless of weight, shall not exceed 1.5 times the stall speed for that configuration.

(3) The force required to maintain steady flight in this configuration at 1.4 V_{s_1}, shall not exceed 10 pounds.

(4) It shall be possible, trimmed in this configuration, to execute a normal power-off landing without exceeding a stick force of 40 pounds.

(5) It shall be possible, with the stick free, to reduce the rate of descent to zero and simultaneously bring the airplane to an attitude suitable for landing, using not more than maximum continuous power. During this demonstration the flaps-extended speed shall not be exceeded.

(b) When the standards set forth above are relied upon to determine compliance with section 3.112, the Administrator will accept as equivalent safety a demonstration of the following items at 1.5 times stall speed instead of 1.4 times stall speed:

Longitudinal control (sec. 3.109 (a) and (b) (2), (5), and (6)).

Specific conditions (sec. 3.115 (a)).

(c) Either the requirement of section 3.112 (a) (2) (ii) or that of the alternate method outlined in paragraphs (a) and (b) of this section should be met in full; no interpolation between the 1.4 V_{s_1} and 1.5 V_{s_1} (for cases where the 90 percent factor cannot be met) may be permitted. For example, an airplane whose flaps-down stall speed is 95 percent of flaps-up stall speed is not to be permitted to demonstrated minimum trim at 1.45 V_{s_1}, but should comply with the original requirement in section 3.112 (a) (2) (ii).

(Supp. 1, 12 F. R. 3435, May 28, 1947, as amended by Amdt. 1, 14 F. R. 36, Jan. 5, 1949; Supp. 10, 16 F. R. 3284, Apr. 14, 1951.)

Stability

3.113 *General.* The airplane shall be longitudinally, directionally, and laterally stable in accordance with the following sections. Suitable stability and control "feel" (static stability) shall be required in other conditions normally encountered in service, if flight tests show such stability to be necessary for safe operation.

3.114 *Static longitudinal stability.* In the configurations outlined in section 3.115 and with the airplane trimmed as indicated, the characteristics of the elevator control forces and the friction within the control system shall be such that:

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

(b) The air speed shall return to within 10 percent of the original trim speed when the control force is slowly released from any speed within the limits defined in paragraph (a) of this section.

3.115 *Specific conditions.* In conditions set forth in this section, within the speeds specified, the stable slope of stick force versus speed curve shall be such that any substantial change

in speed is clearly perceptible to the pilot through a resulting change in stick force.

(a) *Landing.* The stick force curve shall have a stable slope and the stick force shall not exceed 40 lbs. at any speed between 1.1 V_{s_1} and 1.8 V_{s_1} with:

(1) Wing flaps in the landing position,

(2) The landing gear extended,

(3) Maximum weight,

(4) Throttles closed on all engines,

(5) Airplanes of more than 6,000 pounds maximum weight trimmed at 1.4V_{s_1}, and airplanes of 6,000 pounds or less maximum weight trimmed at 1.5V_{s_1}.

(b) *Climb.* The stick force curve shall have a stable slope at all speeds between 1.2 V_{s_1} and 1.6 V_{s_1} with:

(1) Wing flaps retracted,

(2) Landing gear retracted,

(3) Maximum weight,

(4) 75 percent of maximum continuous power,

(5) The airplane trimmed at 1.4 V_{s_1}.

(c) *Cruising.* (1) Between 1.3 V_{s_1} and the maximum permissible speed, the stick force curve shall have a stable slope at all speeds obtainable with a stick force not in excess of 40 pounds with:

(i) Landing gear retracted,

(ii) Wing flaps retracted,

(iii) Maximum weight,

(iv) 75 percent of maximum continuous power,

(v) The airplane trimmed for level flight with 75 percent of the maximum continuous power.

(2) Same as subparagraph (1) of this paragraph, except that the landing gear shall be extended and the level flight trim speed need not be exceeded.

3.116 *Instrumented stick force measurements.* Instrumented stick force measurements need not be made when changes in speed are clearly reflected by changes in stick forces and the maximum forces obtained in the above conditions are not excessive.

3.117 *Dynamic longitudinal stability.* Any short period oscillation occurring between stalling speed and maximum permissible speed shall be heavily damped with the primary controls (1) free, and (2) in a fixed position.

3.118 *Directional and lateral stability.*

(a) *Three-control airplanes.*

(1) The static directional stability, as shown by the tendency to recover from a skid with rudder free, shall be positive for all [landing gear and] flap positions and symmetrical power conditions, and for all speeds from 1.2 V_{s_1} up to the maximum permissible speed.

(2) The static lateral stability as shown by the tendency to raise the low wing in a sideslip, for all [landing gear and] flap positions and symmetrical power conditions, shall:

(i) Be positive at the maximum permissible speed.

(ii) Not be negative at a speed equal to 1.2 V_{s_1}.

(3) In straight steady sideslips (unaccelerated forward slips), the aileron and rudder control movements and forces shall increase steadily, but not necessarily in constant proportion, as the angle of sideslip is increased; the rate of increase of the movements and forces shall lie between satisfactory limits up to sideslip angles considered appropriate to the operation of the type. At greater angles, up to that at which the full rudder control is employed or a rudder pedal force of 150 pounds is obtained, the rudder pedal forces shall not reverse and increased rudder deflection shall produce increased angles of sideslip. Sufficient bank shall accompany sideslipping to indicate adequately any departure from steady unyawed flight.

(4) Any short-period oscillation occurring between stalling speed and maximum permissible speed shall be heavily damped with the primary controls (i) free and (ii) in a fixed position.

(b) *Two-control (or simplified) airplanes.*

(1) The directional stability shall be shown to be adequate by demonstrating that the airplane in all configurations can be rapidly rolled from a 45-degree bank to a 45-degree bank in the opposite direction without exhibiting dangerous skidding characteristics.

(2) Lateral stability shall be shown to be adequate by demonstrating that the airplane will not assume a dangerous attitude or speed

when all the controls are abandoned for a period of 2 minutes. This demonstration shall be made in moderately smooth air with the airplane trimmed for straight level flight at 0.9 V_h (or at V_c, if lower), flaps and gear retracted, and with rearward center of gravity loading.

(3) Any short period oscillation occurring between the stalling speed and the maximum permissible speed shall be heavily damped with the primary controls (i) free and (ii) in a fixed position.

3.118–1 *Test conditions (CAA policies which apply to sec. 3.118 (a) (3)).* The tests made necessary in section 3.118 (a) (3) may be conducted at speeds up to 1.2 times stall speed, flaps up and down, and with power up to 75 percent of maximum continuous rating.

(Supp. 1, 12 F. R. 3435, May 28, 1947, as amended by Amdt. 1, 14 F. R. 36, Jan. 5, 1949.)

3.118–2 *Large displacements of flight controls in directional and lateral stability tests (CAA policies which apply to sec. 3.118).*

(a) In performing flight tests to determine compliance with section 3.118, it should be borne in mind that the airplane structural requirements do not provide for large displacements of the flight controls at high speeds. Full application of rudder and aileron controls should be confined to speeds below the design maneuvering speed V_p. The following rules (approximations) will serve as a guide for the maximum permissible control surface deflections at speeds above V_p. (This does not imply that these maximum deflections must be used in the tests at high speeds).

(1) The permissible rudder angle decreases approximately according to the ratio $(V_p/V)^2$, where V is the speed of the test.

(2) The permissible aileron deflection decreases approximately at the ratio (V_p/V) up to the design cruising speed, V_c. Above V_c, the permissible aileron deflection decreases at a faster rate.

(b) Thus, in a typical case, assuming V_p is 141 mph, V_c is 200 mph, and V_{NE} is 250 mph:

	V_c	V_{NE}
Permissible rudder deflection	50%	32%
Permissible aileron deflection	70%	32%

where 100 percent is the deflection obtainable at V_p.

(c) Control movements should be made smoothly and sudden reversals avoided.

(Supp. 10, 16 F. R. 3284, Apr. 14, 1951.)

3.118–3 *Flight tests for adverse control force reversal or control locking (CAA policies which apply to sec. 3.118 (a) (3)).*

(a) Tests should be conducted in all critical configurations, weights and c. g. positions from power off to 75 percent M. C. P. for the following speeds and any higher speeds if considered more critical:

(1) *Normal category.*

(i) Over 4,000 pounds: $1.2V_{s1}$

(ii) Under 4,000 pounds: All speeds from $1.2V_{s_1}$ down to the lowest speed attainable in steady unstalled flight.

(2) *Utility and acrobatic categories (regardless of weight).* Same as subparagraph (1) (ii) of this paragraph.

(b) V_{s_1} is the stalling speed in the critical configuration as defined in section 3.82 (b).

(c) The rear c. g. is usually critical for these tests.

(Supp. 10, 16 F. R. 3284, Apr. 14, 1951.)

Stalls

3.120 *Stalling demonstration.*

(a) Stalls shall be demonstrated under two conditions:

(1) With power off, and

(2) With a power setting of not less than that required to show compliance with the provisions of section 3.85 (a) for airplanes of more than 6,000 pounds maximum weight, or with 90 percent of maximum continuous power for airplanes of 6,000 pounds or less maximum weight.

(b) In either condition required by paragraph (a) of this section it shall be possible, with flaps and landing gear in any position, with center of gravity in the position least favorable for recovery, and with appropriate airplane weights, to show compliance with the applicable requirements of paragraphs (c) through (f) of this section.

(c) For airplanes having independently controlled rolling and directional controls, it shall be possible to produce and to correct roll by unreversed use of the rolling control and to produce and correct yaw by unreversed use of

the directional control up until the time the airplane pitches in the maneuver prescribed in paragraph (g) of this section.

(d) For two-control airplanes having either interconnected lateral and directional controls or for airplanes having only one of these controls, it shall be possible to produce and to correct roll by unreversed use of the rolling control without producing excessive yaw up until the time the airplane pitches in the maneuver prescribed in paragraph (g) of this section.

(e) During the recovery portion of the maneuver, it shall be possible to prevent more than 15 degrees roll or yaw by the normal use of controls, and any loss of altitude in excess of 100 feet or any pitch in excess of 30 degrees below level shall be entered in the Airplane Flight Manual.

(f) A clear and distinctive stall warning shall precede the stalling of the airplane, with the flaps and landing gear in any position, both in straight and turning flight. The stall warning shall begin at a speed exceeding that of stalling by not less than 5 but not more than 10 miles per hour and shall continue until the stall occurs.

(g) In demonstrating the qualities required by paragraphs (c) through (f) of this section, the procedure set forth in subparagraphs (1) and (2) of this paragraph shall be followed.

(1) With trim controls adjusted for straight flight at a speed of approximately 1.4 V_{s_1} for airplanes of more than 6,000 pounds maximum weight, or approximately 1.5 V_{s_1} for airplanes of 6,000 pounds or less maximum weight, the speed shall be reduced by means of the elevator control until the speed is slightly above the stalling speed; then

(2) The elevator control shall be pulled back at a rate such that the airplane speed reduction does not exceed 1 mile per hour per second until a stall is produced as evidenced by an uncontrollable downward pitching motion of the airplane, or until the control reaches the stop. Normal use of the elevator control for recovery shall be allowed after such pitching motion has unmistakably developed.

3.120–1 *Measuring loss of altitude during stall (CAA policies which apply to sec. 3.120).* To meet the requirements of section 3.120, pertaining to the maximum loss of altitude permitted during the stall, it is necessary that a suitable method be used for the purpose of measuring such loss during the investigation of stalls. Unless special features of an individual type being investigated render the following instructions inapplicable, the procedure described shall be used for this purpose:

(a) The standard procedure for approaching a stall shall be used as specified in section 3.120.

(b) The loss of altitude encountered in the stall (power on or power off) shall be the distance as observed on the sensitive altimeter testing installation from the moment the airplane pitches to the observed altitude reading at which horizontal flight has been regained.

(c) Power used during the recovery portions of a stall maneuver may be that which, at the discretion of the inspector, would be likely used by a pilot under normal operating conditions when executing this particular maneuver. However, the power used to regain level flight shall not be applied until the airplane has regained flying control at a speed of approximately 1.2 V_{s_1}. This means that in the investigation of stalls with the critical engine inoperative, the power may be reduced on the operating engine(s) before reapplying power on the operating engine or engines for the purpose of regaining level flight.

(Supp. 1, 12 F. R. 3435, May 28, 1947, as amended by Amdt. 1, 14 F. R. 36, Jan. 5, 1949.)

3.120–2 *Indications of stall warnings (CAA policies which apply to sec. 3.120).*

(a) No precise and complete description of the various warnings that would comply with section 3.120 can be given at this time, but the following lists of items may be used as a guide:

(1) Satisfactory items include:

(i) Buffeting, which may be defined as general shaking or vibration of the airplane, elevator nibble, aileron nibble, rudder nibble, audible indications such as oil canning of structural members or covering roughness in riding qualities of the airplane due to aerodynamic disturbances, etc.

(ii) Stall warning instrument, either visual or aural. A visual instrument could be either a light or a dial.

(iii) Stick force, defined as heavy.

(iv) Stick travel to hold attitude.

(v) Stick position.

(2) Unsatisfactory items include:
(i) Airplane attitude.
(ii) Inability to hold heading.
(iii) Inability to hold wing level.

(Supp. 10, 16 F. R. 3284, Apr. 14, 1951.)

3.121 *Climbing stalls.* When stalled from an excessive climb attitude it shall be possible to recover from this maneuver without exceeding the limiting air speed or the allowable acceleration limit.

3.121-1 *Climbing stall flight tests for limited control airplanes (CAA interpretations which apply to sec. 3.121).*

(a) This requirement is intended to draw particular attention to any stall recovery characteristics that might be encountered when a limited control airplane is completely stalled from an extremely nose high attitude, either intentionally or inadvertently. In practice it is possible that the elevator control travel could be limited to such an extent that stalls could not be obtained at the normal rate of deceleration used in testing. However, if the airplane was pulled up into a very steep climbing attitude from reasonably high speed flight either power on or power off, and held in this attitude, excessive pitching may occur. At the same time, the limited elevator travel may retard recovery from the pitched attitude until excessively high speeds are obtained. These characteristics would normally be considered under section 3.106; however, it appears wise to call particular attention to the control characteristics that might result from these flight configurations on limited control airplanes.

(b) Although Form ACA-283-03, item A, (3), (a), indicates that take-off power should be used for these tests, this is not a mandatory requirement. In this regard it is to be noted that although section 3.121 is entitled "Climbing Stalls", it specifically states: ". . . when stalled from an excessive climb attitude", thus a specified application of power is not required. For example, flight tests recently conducted on several aircraft have indicated that the power-off configuration was critical since the stall resulted in greater pitch and less elevator control. The technique used for inducing such stalls consisted of stalling the airplane (power off) in as steep a climbing attitude as possible without falling into a whip stall, or other flight maneuver that might overstress the structure. (Form ACA-283-03 will be revised at the next printing, so that the power found to be critical can be recorded in a space that will be provided for this purpose.)

(Supp. 10, 16 F. R. 3284, Apr. 14, 1951.)

3.122 *Turning flight stalls.* When stalled during a coordinated 30-degree banked turn with 75 percent maximum continuous power on all engines, flaps and landing gear retracted, it shall be possible to recover to normal level flight without encountering excessive loss of altitude, uncontrollable rolling characteristics, or uncontrollable spinning tendencies. These qualities shall be demonstrated by performing the following maneuver: After a steady curvilinear level coordinated flight condition in a 30-degree bank is established and while maintaining the 30-degree bank, the airplane shall be stalled by steadily and progressively tightening the turn with the elevator control until the airplane is stalled or until the elevator has reached its stop. When the stall has fully developed, recovery to level flight shall be made with normal use of the controls.

3.123 *One-engine-inoperative stalls.* Multiengine airplanes shall not display any undue spinning tendency and shall be safely recoverable without applying power to the inoperative engine when stalled with:

(a) The critical engine inoperative,

(b) Flaps and landing gear retracted,

(c) The remaining engines operating at up to 75 percent of maximum continuous power, except that the power need not be greater than that at which the use of maximum control travel just holds the wings laterally level in approaching the stall. The operating engines may be throttled back during the recovery from the stall.

Spinning

3.124 *Spinning.*

(a) *Category N.* All airplanes of 4,000 lbs. or less maximum weight shall recover from a one-turn spin with the controls applied normally for recovery in not more than one additional turn and without exceeding either the limiting air speed or the limit positive maneuvering load factor for the airplane. In addition, there shall be no excessive back pressure either

during the spin or in the recovery. It shall not be possible to obtain uncontrollable spins by means of any possible use of the controls. Compliance with these requirements shall be demonstrated at any permissible combination of weight and center of gravity positions obtainable with all or any part of the designed useful load. All airplanes in category N, regardless of weight, shall be placarded against spins or demonstrated to be "characteristically incapable of spinning" in which case they shall be so designated. (See paragraph (d) of this section.)

(b) *Category U.* Airplanes in this category shall comply with either the entire requirements of paragraph (a) of this section or the entire requirements of paragraph (c) of this section.

(c) *Category A.* All airplanes in this category shall be capable of spinning and shall comply with the following:

(1) At any permissible combination of weight and center of gravity position obtainable with all or part of the design useful load, the airplane shall recover from a six-turn spin, or from any point in a six-turn spin, in not more than 1½ additional turns after the application of the controls in the manner normally used for recovery.

(2) It shall be possible to recover from the maneuver prescribed in subparagraph (1) of this paragraph without exceeding either the limiting air speed or the limit positive maneuvering load factor of the airplane.

(3) It shall not be possible to obtain uncontrollable spins by means of any possible use of the controls.

(4) A placard shall be placed in the cockpit of the airplane setting forth the use of the controls required for recovery from spinning maneuvers.

(d) *Category NU.* When it is desired to designate an airplane as a type "characteristically incapable of spinning," the flight tests to demonstrate this characteristic shall also be conducted with:

(1) A maximum weight 5 percent in excess of the weight for which approval is desired,

(2) A center of gravity at least 3 percent aft of the rearmost position for which approval is desired.

(3) An available up-elevator travel 4 degrees in excess of that to which the elevator travel is to be limited by appropriate stops.

(4) An available rudder travel 7 degrees, in both directions, in excess of that to which the rudder travel is to be limited by appropriate stops.

3.124–1 *Spin tests for category N airplanes (CAA interpretations which apply to sec. 3.124 (a)).* If during recovery from a one-turn flaps-down spin the airplane exceeds the placard flap speed or limit load factor, it is permissible to retract the flaps during recovery to avoid exceeding these limits.

(Supp. 10, 16 F. R. 3284, Apr. 14, 1951.)

3.124–2 *Spin tests for category A airplanes (CAA interpretations which apply to sec. 3.124 (c)).* If during recovery from a one-turn flaps-down spin the airplane exceeds the placard flap speed or limit load factor, it is permissible to retract the flaps during recovery to avoid exceeding these limits. In addition the airplane is to be placarded "Intentional spins with flaps down prohibited."

(Supp. 10, 16 F. R. 3284, Apr. 14, 1951.)

Ground and Water Characteristics

3.143 *Requirements.* All airplanes shall comply with the requirements of sections 3.144 to 3.147.

3.144 *Longitudinal stability and control.* There shall be no uncontrollable tendency for landplanes to nose over in any operating condition reasonably expected for the type, or when rebound occurs during landing or take-off. Wheel brakes shall operate smoothly and shall exhibit no undue tendency to induce nosing over. Seaplanes shall exhibit no dangerous or uncontrollable porpoising at any speed at which the airplane is normally operated on the water.

3.145 *Directional stability and control.*

(a) There shall be no uncontrollable looping tendency in 90-degree cross winds up to a velocity equal to 0.2 V_{s_0} at any speed at which the aircraft may be expected to be operated upon the ground or water.

(b) All landplanes shall be demonstrated to be satisfactory controllable with no exceptional degree of skill or alertness on the part of the

pilot in power-off landings at normal landing speed and during which brakes or engine power are not used to maintain a straight path.

(c) Means shall be provided for adequate directional control during taxying.

3.146 *Shock absorption.* The shock-absorbing mechanism shall not produce damage to the structure when the airplane is taxied on the roughest ground which it is reasonable to expect the airplane to encounter in normal operation.

3.147 *Spray characteristics.* For seaplanes, spray during taxying, take-off, and landing shall at no time dangerously obscure the vision of the pilots nor produce damage to the propeller or other parts of the airplane.

Flutter and Vibration

3.159 *Flutter and vibration.* All parts of the airplane shall be demonstrated to be free from flutter and excessive vibration under all speed and power conditions appropriate to the operation of the airplane up to at least the minimum value permitted for V_a in section 3.184. There shall also be no buffeting condition in any normal flight condition severe enough to interfere with the satisfactory control of the airplane or to cause excessive fatigue to the crew or result in structural damage. However, buffeting as stall warning is considered desirable and discouragement of this type of buffeting is not intended.

Subpart C—Strength Requirements

General

3.171 *Loads.*

(a) Strength requirements are specified in terms of limit and ultimate loads. Limit loads are the maximum loads anticipated in service. Ultimate loads are equal to the limit loads multiplied by the factor of safety. Unless otherwise described, loads specified are limit loads.

(b) Unless otherwise provided, the specified air, ground, and water loads shall be placed in equilibrium with inertia forces, considering all items of mass in the airplane. All such loads shall be distributed in a manner conservatively approximating or closely representing actual conditions. If deflections under load would change significantly the distribution of external or internal loads, such redistribution shall be taken into account.

(c) Simplified structural design criteria shall be acceptable if the Administrator finds that they result in design loads not less than those prescribed in sections 3.181 through 3.265.

3.171–1 *Design criteria (CAA policies which apply to sec. 3.171 (c)).* The Administrator finds that the simplified structural design criteria contained in Appendix A to Civil Aeronautics Manual 3, result in design loads not less than those prescribed in sections 3.181 through 3.265.

(Supp. 16, effective Jan. 31, 1953, 17 F. R. 11786, Dec. 30, 1952.)

[3.171–2 *Design loads and load distributions (CAA policies which apply to sec. 3.171 (b)).* The simplified method in appendix D to Civil Aeronautics Manual 3 may be used to determine the air loads and air load distributions resulting from the use of tip stores for low speed, low altitude (design Mach number less than 0.4; design altitude less than 15,000 ft.) airplanes with small amounts of sweep (i. e., midchord angles of sweep less than 15 degrees).]

(22 F. R. 10016, Dec. 13, 1957, effective Jan. 15, 1958.)

3.172 *Factor of safety.* The factor of safety shall be 1.5 unless otherwise specified.

3.173 *Strength and deformations.* The structure shall be capable of supporting limit loads without suffering detrimental permanent deformations. At all loads up to limit loads, the deformation shall be such as not to interfere with safe operation of the airplane. The structure shall be capable of supporting ultimate loads without failure for at least 3 seconds, except that when proof of strength is demonstrated by dynamic tests simulating actual conditions of load application, the 3-second limit does not apply.

3.173–1 *Dynamic tests (CAA policies which apply to sec. 3.173).*

(a) Section 3.173 permits dynamic testing in lieu of stress analysis or static testing in the proof of compliance of the structure with strength and deformation requirements. In demonstrating, by dynamic tests, proof of strength of landing gears for the stipulated landing conditions contained in sections 3.245, 3.246, and 3.247, it is necessary to employ a procedure which will not result in the accepting of landing gears weaker than those qualified for acceptance under present procedures, i. e., stress analysis or static testing.

(b) The Administrator will accept, as an adequate procedure for this purpose, the following dynamic tests:

The structure shall be dropped a minimum of 10 times from the limit drop height, and at least one time from the ultimate drop height, for each basic design condition for which proof of strength is being made by drop tests.

(c) With regard to the extent to which the structure can be proved by dynamic tests, such dynamic tests shall be accepted as proof of strength for only those elements of the structure for which it can be shown that the critical limit and ultimate loads have been reproduced.

(Supp. 1, 12 F. R. 3435, May 28, 1947, as amended by Amdt. 1, 14 F. R. 36, Jan. 5, 1949.)

3.174 *Proof of structure.* Proof of compliance of the structure with the strength and deformation requirements of section 3.173 shall be made for all critical loading conditions. Proof of compliance by means of structural analysis will be accepted only when the structure conforms with types for which experience has shown such methods to be reliable. In all other cases substantiating load tests are required. Dynamic tests including structural flight tests shall be acceptable, provided that it is demonstrated that the design load conditions have been simulated. In all cases certain portions of the structure must be subjected to tests as specified in Subpart D of this part.

3.174-1 *Material correction factors (CAA policies which apply to sec. 3.174).*

(a) In tests conducted for the purpose of establishing allowable strengths of structural elements such as sheet, sheet stringer combinations, riveted joints, etc., test results should be reduced to values which would be met by elements of the structure if constructed of materials having properties equal to design allowable values. Material correction factors in this case may be omitted, however, if sufficient test data are obtained to permit a probability analysis showing that 90 percent or more of the elements will either equal or exceed in strength the selected design allowable values. The number of individual test specimens needed to form a basis of "probability values" cannot be definitely stated but must be decided on the basis of consistency of results; i. e., "spread of results", deviations from mean value, and range of sizes, dimensions of specimens, etc., to be covered. This item should therefore be a matter for decision between the manufacturer and the CAA. (Secs. 1.654 and 1.655 of ANC-5a 1949 edition [7] outline two means of accomplishing material corrections in element tests; these methods, however, are by no means considered the only methods available.)

(b) In cases of static or dynamic tests of structural components, no material correction factor is required. The manufacturer, however, should use care to see that the strength of the component tested conservatively represents the strength of subsequent similar components to be used on aircraft to be presented for certification. The manufacturer should, in addition, include in his report of tests of major structural components, a statement substantially as follows:

The strength properties of materials and dimensions of parts used in the structural component(s) tested are such that subsequent components of these types used in aircraft presented for certification will have strengths substantially equal to or exceeding the strengths of the components tested.

(Supp. 6, 15 F. R. 619, Feb. 4, 9050.)

3.174-2 *Structural testing of new projects (CAA policies which apply to sec. 3.174).*

(a) The following is a general procedure that may be followed for determining the extent of required structural testing of a new project:

(1) As the initial step to determine the structural testing of a new project, a meeting between representatives of the manufacturer, the Civil Aeronautics Administration project engineer, and (if practicable) the pertinent Branch Chief of the Aircraft Division should be arranged. The question of minimum tests should be reviewed first. This will include generally such tests as proof and operation tests of control surfaces and systems, drop

[7] ANC-5a, "Strength of Aircraft Elements" is published by the Army-Navy-Civil Committee on Aircraft Design Criteria and may be obtained from the Superintendent of Documents, Government Printing Office, Washington 25, D. C.

that continuous readings of deflections of the structure are measured at an adequate number of points, and also provided that a close examination of the structure is maintained throughout the tests with particular emphasis being placed upon close observation of the

dynamic loading is usually critical is the landing gear and landing gear structure of an aircraft. (See 3.174-2.) The same yield criteria apply to dynamic tests as to static tests.

(Supp. 10, 16 F. R. 3285, Apr. 14, 1951.)

tests of landing gear, vibration tests, and wing torsional stiffness tests.

(2) If the structure is of a type on which the manufacturer has a thorough background of experience, analysis and proof tests can usually be considered acceptable. If, in addition, the analysis has a high degree of conservatism, proof tests other than those specifically required by regulation may be omitted at the discretion of the CAA.

(b) If the structure or parts thereof are definitely outside the manufacturer's previous

3.306 need not be included in test loads in which the actual stress conditions are simulated in the fitting and the surrounding structure. Also, these factors are considered to be included in and covered by the other special factors specified in section 3.302.

(e) *Casting factors.* Casting factors should be included in all tests in the substantiation of castings. (See sec. 3.304–1.)

(f) *Hinge and bearing factors.* Hinge and bearing factors specified shall be included in tests unless the appropriate portions of the parts are substantiated otherwise.

(g) *Other factors.* Test factors for rib, wing, and wing-covering are as follows:

(1) No additional factors of safety need be applied when rational chordwise upper and lower surface pressure distribution is used, provided that the test includes a complete wing or a section of a wing with end conditions and loadings applied in a manner closely simulating the actual wing conditions.

(2) When a rib alone, a section of wing, or small section of the airplane covering is tested without employing a completely rational load analysis and distribution, a factor of 1.25 should be included in the test loads. In an intermediate case, a factor between 1.0 and 1.25 may be employed in wing section tests if it is suitably established that a reduction from 1.25 is warranted by the particular conditions of the test.

(Supp. 10, 16 F. R. 3285, Apr. 14, 1951.)

3.174–7 *Establishment of material strength properties and design values by static test (CAA policies which apply to sec. 3.174).*

(a) There are several types of material design allowables, all of which are derived from test data. These are:

(1) Minimum acceptable values based on a minimum value already in an applicable materials procurement specification.

(2) Minimum non-specification values derived from tests of a series of standard specimens.

(3) Ninety percent probability values which are the lowest strength values expected in 90 percent of the specimens tested.

(4) Values based on "premium selection" of the material.

(b) Where testing is used to determine any of these types of allowables, procedures outlined in existing Government or industry specifications, e. g. QQ–M–151, ASTM's, etc., should be used although other procedures if approved by the CAA may be used. No clear-cut rules as to the extent of testing to be done can be established in this section, as this usually varies with the case. It is therefore a matter for joint discussion between the manufacturer and the CAA. The results, however, should be based on a sufficiently large number of tests of the material to establish minimum acceptable or probability values on a statistical basis.

(c) Design values pertinent to the items in paragraphs (a) (1), (2) and (3) of this section are presented in ANC–5 and ANC–18 for commonly used materials.

(d) With reference to paragraph (a) (4) of this section, some manufacturers have indicated a desire to use values greater than the established minimum acceptable values even in cases where only the use of minimum acceptable values is indicated. Such increases will be acceptable provided that specimens of each individual item of basic material as obtained are tested prior to use, to ascertain that the strength properties of that particular item will equal or exceed the properties to be used in design.

(Supp. 10, 16 F. R. 3285, Apr. 14, 1951, as amended by Supp. 14, 17 F. R. 9066, Oct. 11, 1952.)

3.174–8 *Unusual test situations (CAA policies which apply to sec. 3.174).* It should be borne in mind that in any unusual or different situations a conference between the CAA and the manufacturer should be held to determine if the testing program as proposed by the manufacturer is sufficient to substantiate the structural strength of the aircraft or its component.

(Supp. 10, 16 F. R. 3285, Apr. 14, 1951.)

Flight Loads

3.181 *General.* Flight load requirements shall be complied with at critical altitudes within the range in which the airplane may be expected to operate and at all weights between the minimum design weight and the maximum design weight, with any practicable distribution of disposable load within prescribed operating limitations stated in sections 3.777–3.780.

3.182 *Definition of flight load factors.*

The flight load factors specified represent the acceleration component (in terms of the gravitational constant g) normal to the assumed longitudinal axis of the airplane, and equal in magnitude and opposite in direction to the airplane inertia load factor at the center of gravity.

Symmetrical Flight Conditions (Flaps Retracted)

3.183 *General.* The strength requirements shall be met at all combinations of air speed and load factor on and within the boundaries of a pertinent V–n diagram, constructed similarly to the one shown in Figure 3–1, which represents the envelope of the flight loading conditions specified by the maneuvering and gust criteria of sections 3.185 and 3.187. This diagram will also be used in determining the airplane structural operating limitations as specified in Subpart G of this part.

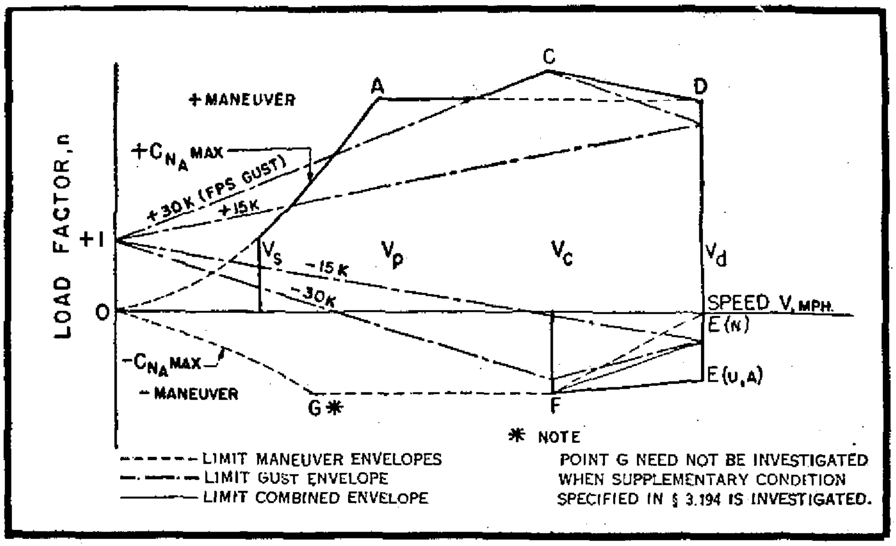


Figure 3–1.—(V-n) Diagram (flight envelope).

3.184 *Design air speeds.* The design air speeds shall be chosen by the designer except that they shall not be less than the following values:

V_c (design cruising speed)

$=38\sqrt{W/S}$ (NU)

$=42\sqrt{W/S}$ (A)

except that for values of W/S greater than 20, the above numerical multiplying factors shall be decreased linearly with W/S to a value of 33 at $W/S=100$: *And further provided,* That the required minimum value need be no greater than 0.9 V_h actually obtained at sea level.

V_d (design dive speed)

$=1.40\ V_{c\,\min}$ (N)

$=1.50\ V_{c\,\min}$ (U)

$=1.55\ V_{c\,\min}$ (A)

except that for values of W/S greater than 20, the above numerical multiplying factors shall be decreased linearly with W/S to a value of 1.35 at $W/S=100$. ($V_{c\,\min}$ is the required minimum value of design cruising speed specified above.)

V_p (design maneuvering speed)

$=V_s\sqrt{n}$ where:

V_s=a computed stalling speed with flaps fully retracted at the design weight, normally based on the maximum airplane normal force coefficient, C_{NA}.

n=limit maneuvering load factor used in design,

except that the value of V_p need not exceed the value of V_c used in design.

3.185 *Maneuvering envelope.* The airplane shall be assumed to be subjected to

symmetrical maneuvers resulting in the following limit load factors, except where limited by maximum (static) lift coefficients:

(a) The positive maneuvering load factor specified in section 3.186 at all speeds up to V_d,

(b) The negative maneuvering load factor specified in section 3.186 at speed V_c; and factors varying linearly with speed from the specified value at V_c to 0.0 at V_d for the N category and −1.0 at V_d for the A and U categories.

3.186 ***Maneuvering load factors.***

(a) The positive limit maneuvering load factors shall not be less than the following values:

$$n=2.1+\frac{24{,}000}{W+10{,}000} \text{ .... Category N}$$

except that n need not be greater than 3.8 and shall not be less than 2.5.

$n=4.4$ .... **Category U**

$n=6.0$ .... **Category A**

(b) The negative limit maneuvering load factors shall not be less than −0.4 times the positive load factor for the N and U categories, and shall not be less than −0.5 times the positive load factor for the A category.

(c) Lower values of maneuvering load factor may be employed only if it be proven that the airplane embodies features of design which make it impossible to exceed such values in flight. (See also sec. 3.106.)

3.187 ***Gust envelope.*** **The airplane shall be assumed to encounter symmetrical vertical gusts as specified below while in level flight and the resulting loads shall be considered limit loads:**

(a) Positive (up) and negative (down) gusts of 30 feet per second nominal intensity at all speeds up to V_c,

(b) Positive and negative 15 feet per second gusts at V_d. Gust load factors shall be assumed to vary linearly between V_c and V_d.

3.188 ***Gust load factors.*** **In applying the gust requirements, the gust load factors shall be computed by the following formula:**

$$n=1+\frac{KUVm}{575\ (W/S)}$$

where: $K=\frac{1}{2}(W/S)^{1/4}$ **(for $W/S<16$ p. s. f.)**

$$=1.33-\frac{2.67}{(W/S)^{3/4}} \text{ (for } W/S>16 \text{ p. s. f.)}$$

U= nominal gust velocity, f. p. s. (Note that the "effective sharp-edged gust" equals KU.)

V= airplane speed, m. p. h.

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

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

3.188–1 *"Slope of lift curve" (CAA interpretations which apply to sec. 3.188).* For purposes of gust load computations as required in section 3.188 the slope of the lift curve may be assumed equal to that of the wing alone.

(Supp. 1, 12 F. R. 3435, May 28, 1947, as amended by Amdt. 1, 14 F. R. 36, Jan. 5, 1949.)

3.189 ***Airplane equilibrium.*** **In determining the wing loads and linear inertia loads corresponding to any of the above specified flight conditions, the appropriate balancing horizontal tail load (see sec. 3.215) shall be taken into account in a rational or conservative manner.**

Incremental horizontal tail loads due to maneuvering and gusts (see sects. 3.216 and 3.217) shall be reacted by angular inertia of the complete airplane in a rational or conservative manner.

Flaps Extended Flight Conditions

3.190 ***Flaps extended flight conditions.***

(a) When flaps or similar high lift devices intended for use at the relatively low air speeds of approach, landing, and takeoff are installed, the airplane shall be assumed to be subjected to symmetrical maneuvers and gusts with the flaps fully deflected at the design flap speed V_f resulting in limit load factors within the range determined by the following conditions:

(1) Maneuvering, to a positive limit load factor of 2.0.

(2) Positive and negative 15-feet-per-second gusts acting normal to the flight path in level flight. The gust load factors shall be computed by the formula of section 3.188.

V_f shall be assumed not less than 1.4 V_s or 1.8 V_{sf}, whichever is greater, where:

V_s = the computed stalling speed with flaps fully retracted at the design weight

V_{sf}=the computed stalling speed with flaps fully extended at the design weight

except that when an automatic flap load limiting device is employed, the airplane may be designed for critical combinations of air speed and flap position permitted by the device. (See also sec. 3.338.)

(b) In designing the flaps and supporting structure, slipstream effects shall be taken into account as specified in section 3.223.

NOTE: In determining the external loads on the airplane as a whole, the thrust, slipstream, and pitching acceleration may be assumed equal to zero.

3.190-1 *Design flap speed V_f (CAA interpretations which apply to sec. 3.190 (a)).*

(a) The minimum permissible speed of 1.8 V_{sf} is specified in order to cover power-off flight tests as required by section 3.115 (a). Section 3.223 requires that slipstream effects be considered in the design of the flaps and operating mechanism up to a speed of at least 1.4 V_s in order to cover the power on flight tests of section 3.109 (b) (5).

(b) The designer may treat the foregoing conditions as two separate cases, or he may combine them if he so desires.

(Supp. 10, 16 F. R. 3285, Apr. 14, 1951).

Unsymmetrical Flight Conditions

3.191 *Unsymmetrical flight conditions.* The airplane shall be assumed to be subjected to rolling and yawing maneuvers as described in the following conditions. Unbalanced aerodynamic moments about the center of gravity shall be reacted in a rational or conservative manner considering the principal masses furnishing the reacting inertia forces.

(a) *Rolling conditions.* The airplane shall be designed for (1) unsymmetrical wing loads appropriate to the category and (2) the loads resulting from the aileron deflections and speeds specified in section 3.222, in combination with an airplane load factor of at least two-thirds of the positive maneuvering factor used in the design of the airplane. Only the wing and wing bracing need be investigated for this condition.

NOTE: [Unless the Administrator finds such data result in unrealistic loads, these conditions may be covered as follows:]

(a) Rolling accelerations may be obtained by modifying the symmetrical flight conditions shown in Figure 3-1 as follows:

(1) *Acrobatic category.* In conditions A and F assume 100 percent of the wing air load acting on one side of the plane of symmetry and 60 percent on the other.

(2) *Normal and utility categories.* In condition A, assume 100 percent of the wing air load acting on one side of the airplane and 70 percent on the other. For airplanes over 1,000 pounds design weight, the latter percentage may be increased linearly with weight up to 80 percent at 25,000 pounds.

(b) The effect of aileron displacement on wing torsion may be accounted for by adding the following increment to the basic airfoil moment coefficient over the aileron portion of the span in the critical condition as determined by the note under section 3.222:

$$\Delta cm = -.01\delta$$

where:

Δcm=moment coefficient increment

δ=down aileron deflection in degrees in critical condition

(b) *Yawing conditions.* The airplane shall be designed for the yawing loads resulting from the vertical surface loads specified in sections 3.219 to 3.221.

3.191-1 *Aileron rolling conditions (CAA policies which apply to sec. 3.191 (a)).* In determining whether airplanes of small to medium size and speed comply with section 3.191 (a), the Administrator will accept the following simplified procedure:

(a) *Steady roll.* Determine the C_n value, corresponding to ⅔ the symmetrical maneuvering load factor. The C_n distribution over the span may be assumed the same as that for the symmetrical flight conditions. Modify the wing movement coefficient over the aileron portions of the span, as described in the "Note" under section 3.191 (a), corresponding to the required aileron deflections. The wing may be critical in torsion on the up, as well as on the down aileron side, depending upon airfoil section, elastic axis location, aileron differential, etc. (For the up aileron, the moment coefficient increment will be positive.)

The above assumption concerning C_n distribution implies that the aerodynamic damping forces have exactly the same distribution as the rolling forces, which is not strictly correct. However, since the load factor in the rolling conditions is only ⅔ of that in the symmetrical conditions, the errors involved in this assumption are not likely to be significant.

(b) *Maximum angular acceleration.* This condition need be investigated only for wings carrying large mass items outboard. In such cases instantaneous aileron deflection (zero rolling velocity) may be assumed and the local value of C_n and C_m over the aileron portions of the span modified accordingly to obtain the spanwise airload distribution. The average C_n of the entire wing should correspond to ⅔ of the symmetrical maneuvering load factor. The resulting rolling moment should be resisted by the rolling inertia of the entire airplane. This procedure is, in general, conservative, and a more rational investigation based on the time history of the control movement and response of the airplane may be used if desired.

(Supp. 10, 16 F. R. 3285, Apr. 14, 1951.)

Supplementary Conditions

3.194 ***Special condition for rear lift truss.*** **When a rear lift truss is employed, it shall be designed for conditions of reversed airflow at a design speed of:**

$$V=10\sqrt{W/S}+10 \text{ (m. p. h.)}$$

NOTE: It may be assumed that the value of C_L is equal to −0.8 and the chordwise distributor is triangular between a peak at the trailing edge and zero at the leading edge.

3.195 ***Engine torque effects.***

(a) Engine mounts and their supporting structures shall be designed for engine torque effects combined with certain basic flight conditions as described in subparagraphs (1) and (2) of this paragraph. Engine torque may be neglected in the other flight conditions.

(1) The limit torque corresponding to take-off power and propeller speed acting simultaneously with 75 percent of the limit loads from flight condition A. (See Fig. 3–1.)

(2) The limit torque corresponding to maximum continuous power and propeller speed, acting simultaneously with the limit loads from flight condition A. (See Fig. 3–1.)

(b) The limit torque shall be obtained by multiplying the mean torque by a factor of 1.33 in the case of engines having 5 or more cylinders. For 4-, 3-, and 2-cylinder engines, the factors shall be 2, 3, and 4, respectively.

3.196 ***Side load on engine mount.*** **The limit load factor in a lateral direction for this condition shall be at least equal to one-third of the limit load factor for flight condition A (see Fig. 3–1) except that it shall not be less than 1.33. Engine mounts and their supporting structure shall be designed for this condition which may be assumed independent of other flight conditions.**

3.197 ***Pressurized cabin loads.*** **The provisions of paragraphs (a) through (d) of this section shall apply to pressurized compartments.**

(a) The airplane structure shall have sufficient strength to withstand the flight loads combined with pressure differential loads from zero up to the maximum relief valve setting. Account shall be taken of the external pressure distribution in flight. Stress concentrations shall be taken into account in the design of the pressurized structure. (See sec. 3.270.)

(b) If landings are to be permitted with the cabin pressurized, landing loads shall be combined with pressure differential loads from zero up to the maximum permitted during landing.

(c) The airplane structure shall have sufficient strength to withstand the pressure differential loads corresponding with the maximum relief valve setting multiplied by a factor of 1.33. It shall be acceptable to eliminate all other loads in this case.

(d) Where a pressurized cabin is separated into 2 or more compartments by bulkheads or floor, the primary structure shall be designed for the effects of sudden release of pressure in any compartment having external doors or windows. This condition shall be investigated for the effects resulting from the failure of the largest opening in a compartment. Where intercompartment venting is provided, it shall be acceptable to take into account the effects of such venting.

Control Surface Loads

3.211 ***General.*** **The control surface loads specified in the following sections shall be assumed to occur in the symmetrical and unsymmetrical flight conditions as described in sections 3.189–3.191. See Figures 3–3 to 3–10 for acceptable values of control surface loadings which are considered as conforming to the following detailed rational requirements.**

[NOTE: For a seaplane version of a landplane, it is normally acceptable to use the wing loading of the landplane in determining the limit maneuvering control surface loadings from Figure 3-3 (b) provided: the power of the engines and the placard maneuver speed of the seaplane do not exceed those established for the landplane; the maximum certificated weight of the seaplane does not exceed the corresponding weight of the landplane by more than 10 percent; and service experience with the landplane is such that no evidence of any serious control-surface load problems is indicated and is such that the service experience is of sufficient scope to deduce with reasonable accuracy that no serious control-surface load problems will develop on the seaplane.]

Acceptable values of limit average maneuvering control surface loadings can be obtained from Figure 3-3 (b) as follows:

Horizontal Tail Surfaces

(1) Condition section 3.216 (a):
Obtain $\overline{w}$ as function of W/S and surface deflection;
Use Curve C for deflection 10° or less;
Use Curve B for deflection 20°;
Use Curve A for deflection 30° or more;
(Interpolate for other deflections);
Use distribution of Figure 3-8.

(2) Condition section 3.216 (b):
Obtain $\overline{w}$ from Curve B. Use distribution of Figure 3-8.

Vertical Tail Surfaces

(3) Condition section 3.219 (a):
Obtain $\overline{w}$ as function of W/S and surface deflection in same manner as outlined in (1) above, use distribution of Figure 3-8;

(4) Condition section 3.219 (b):
Obtain $\overline{w}$ from Curve C, use distribution of Figure 3-7;

(5) Condition section 3.219 (c):
Obtain $\overline{w}$ from Curve A, use distribution of Figure 3-9. (Note that condition section 3.220 generally will be more critical than this condition.)

Ailerons

(6) In lieu of conditions section 3.222 (b):
Obtain $\overline{w}$ from Curve B, acting in both up and down directions.
Use distribution of Figure 3-10.

Figure 3-3 (a).—Limit average maneuvering control surface loadings.

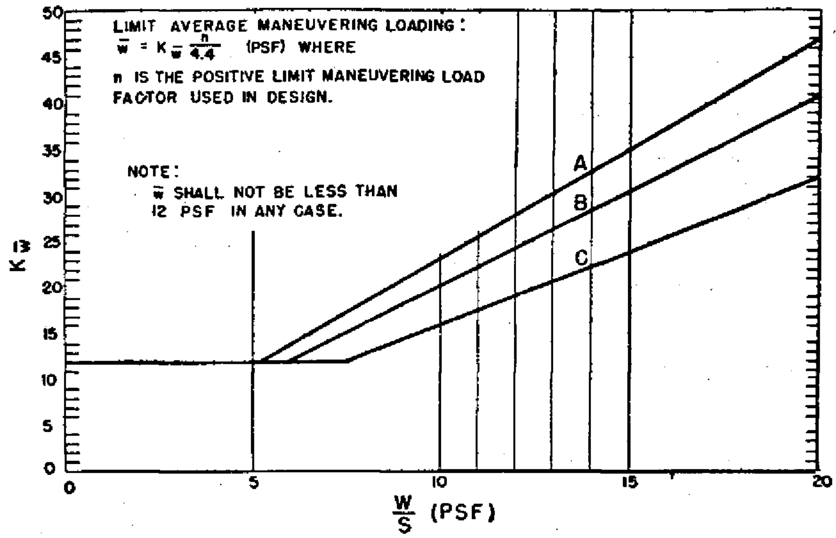


Figure 3–3 (b).—Limit average maneuvering control surface loading.

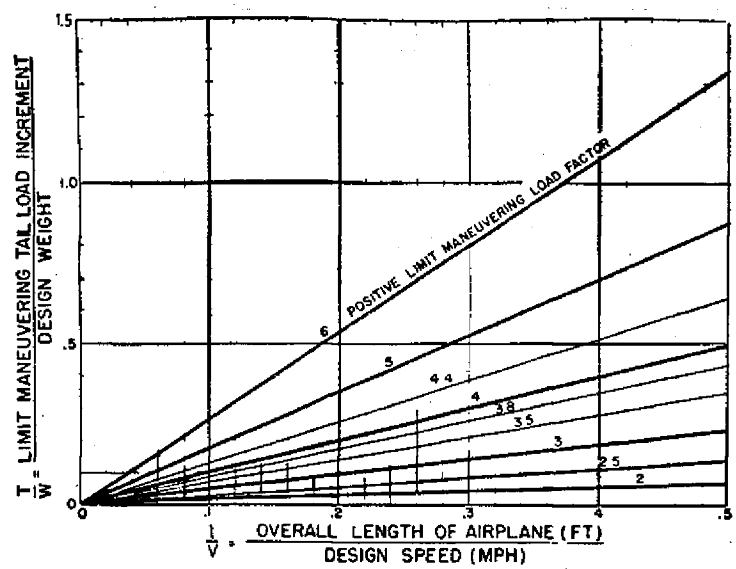


Figure 3–4.—Maneuvering tail load increment (up or down).

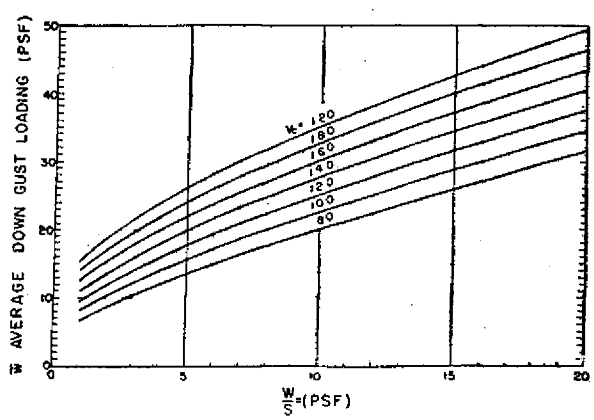


Figure 3–5 (a).—Down gust loading on horizontal tail surface.

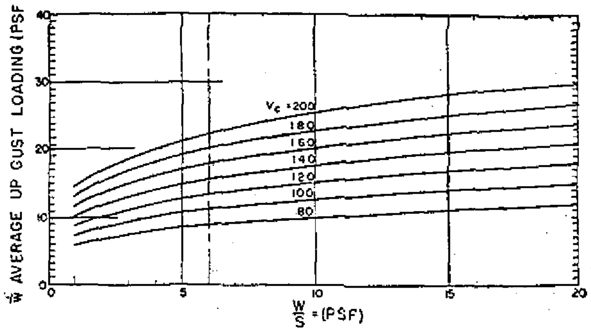


Figure 3–5 (b).—Up gust loading on horizontal tail surface.

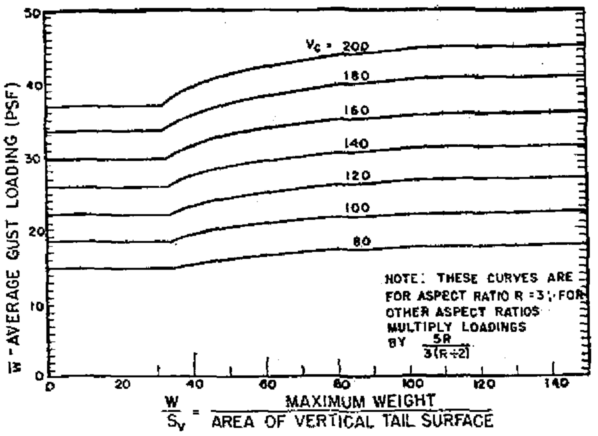


Figure 3–6.—Gust loading on vertical tail surface.

3.211–1 *Control surface loads for design of "Vee" type tail assemblies (CAA policies which apply to sec. 3.211).*

(a) "Vee" type tail assemblies require special design criteria in order to show "the same level of safety" under section 3.10. Thus, for "Vee" type tail assemblies, all the tail road requirements as set forth in this part are considered acceptable to this type tail design. It will be necessary, however, to increase the unit loads on each side of the tail surface to account for the tail surface dihedral, since air loads act normal to the surface only. Thus the unit loads, based on the projected area, on each side of the tail surface due to vertical loads on the tail assembly should be increased by a factor equal to 1/cos Ø, while the unit horizontal loads on the tail assembly should be increased by a factor equal to 1/sin θ, where θ is the dihedral angle, or the angle between each side of the tail surface and the horizontal.

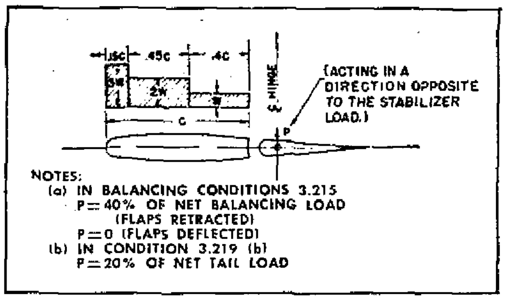


Figure 3–7.—Tail surface load distribution.

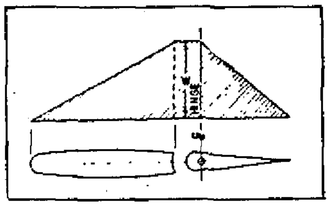


Figure 3–8.—Tail surface load distribution.

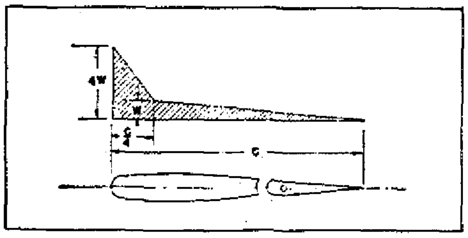


Figure 3–9.—Tail surface load distribution.

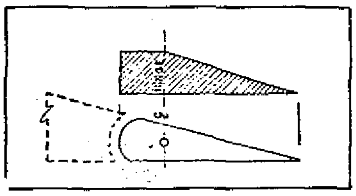


Figure 3–10.—Aileron load distribution.

(b) The following supplementary conditions should also be investigated:

(1) A ±30 fps gust, acting normal to the chord plane of one side of the tail surface at V_c, should be combined with a one "g" balancing tail load. Reduction for downwash is acceptable. It is evident that this condition will be unsymmetrical, since one side of the "Vee" tail will not be highly loaded by the gust.

(2) Combined rudder and elevator maneuvering condition.

(i) In order to obtain the full one way travel of the ruddervator, it is desirable to have full elevator travel in conjunction with full rudder travel. The limiting factor for this configuration is ⅔ elevator load for one pilot, and ⅔ rudder load for one pilot applied simultaneously.

(ii) When it can be shown that the lateral gust condition (reference: sec. 3.220) is less critical than the condition in subparagraph (1) of this paragraph, no analysis for the lateral gust need be made.

(Supp. 10, 16 F. R. 3286, Apr. 14, 1951.)

3.212 *Pilot effort.* In the control surface loading conditions described, the airloads on the movable surfaces and the corresponding deflections need not exceed those which could be obtained in flight by employing the maximum pilot control forces specified in Figure 3–11. In applying this criterion, proper consideration shall be given to the effects of control system boost and servo mechanisms, tabs, and automatic pilot systems in assisting the pilot.

LIMIT PILOT LOADS

Control	Maximum loads for design weight W equal to or less than 5,000 lbs.[1]	Minimum loads[2]
Aileron:		
Stick	67 pounds	40 pounds.
Wheel[3]	53 D in-pounds[4]	40 D in-pounds.[4]
Elevator:		
Stick	167 pounds	100 pounds.
Wheel	200 pounds	100 pounds.
Rudder	200 pounds	130 pounds.

[1] For design weight W greater than 5,000 pounds the above specified maximum values shall be increased linearly with weight to 1.5 times the specified values at a design weight of 25,000 pounds.

[2] If the design of any individual set of control systems or surfaces is such as to make these specified minimum loads inapplicable, values corresponding to the pertinent hinge moments obtained according to section 3.233 may be used instead except that in any case values less than 0.6 of the specified minimum loads shall not be employed.

[3] The critical portions of the aileron control system shall also be designed for a single tangenital force having a limit value equal to 1.25 times the couple force determined from the above criteria.

[4] D=wheel diameter.

Figure 3–11.—Pilot control force limits.

3.212–1 *Automatic pilot systems (CAA policies which apply to sec. 3.212).* The Administrator will accept the following procedure as giving proper consideration of automatic pilot systems in assisting the pilot under section 3.212: The autopilot effort need not be added to human pilot effort but the autopilot effort shall be used for design if it alone can produce greater control surface loads than the human pilot.

(Supp. 1, 12 F. R. 3435, May 28, 1947, as amended by Amdt. 1, 14 F. R. 36, Jan. 5, 1949.)

3.213 *Trim tab effects.* The effects of trim tabs on the control surface design conditions need be taken into account only in cases where the surface loads are limited on the basis of maximum pilot effort. In such cases the tabs shall be considered to be deflected in the direction which would assist the pilot and the deflection shall correspond to the maximum expected degree of "out of trim" at the speed for the condition under consideration.

Horizontal Tail Surfaces

3.214 *Horizontal tail surfaces.* The horizontal tail surfaces shall be designed for the conditions set forth in sections 3.215–3.218.

3.215 *Balancing loads.* A horizontal tail balancing load is defined as that necessary to maintain the airplane in equilibrium in a specified flight condition with zero pitching acceleration. The horizontal tail surfaces shall be designed for the balancing loads occurring at any point on the limit maneuvering envelope, Figure 3–1, and in the flap conditions. (See sec. 3.190.)

[NOTE: The distribution of Figure 3–7 may be used unless the Administrator finds it results in unrealistic loads.]

3.216 *Maneuvering loads.*

(a) At maneuvering speed V_p assume a sudden deflection of the elevator control to the maximum upward deflection as limited by the control stops or pilot effort, whichever is critical.

NOTE: The average loading of Figure 3–3 and the distribution of Figure 3–8 may be used [unless the Administrator finds it results in unrealistic loads.] In determining the resultant normal force coefficient for the tail under these conditions, it will be permissible to assume that the angle of attack of the stabilizer with respect to the resultant direction of air flow is equal to that which occurs when the airplane is in steady unaccelerated flight at a flight speed equal to V_P. The maximum elevator deflection can then be determined from the above criteria and the tail normal force coefficient can be obtained from the data given in NACA Report No. 688, "Aerodynamic Characteristics of Horizontal Tail Surfaces," or other applicable NACA reports.

(b) Same as case (a) except that the elevator deflection is downward.

NOTE: The average loading of Figure 3–3 and the distribution of Figure 3–8 may be used [unless the Administrator finds it results in unrealistic loads.]

(c) At all speeds above V_p the horizontal tail shall be designed for the maneuvering loads resulting from a sudden upward deflection of the elevator, followed by a downward deflection of the elevator such that the following combinations of normal acceleration and angular acceleration are obtained:

Condition	Airplane normal acceleration n	Angular acceleration radian/sec.^{2}
Down load	1.0	$+\frac{45}{V}n_m(n_m-1.5)$
Up load	n_m	$-\frac{45}{V}n_m(n_m-1.5)$

where:

n_m=positive limit maneuvering load factor used in the design of the airplane.

V=initial speed in miles per hour.

(d) The total tail load for the conditions specified in (c) shall be the sum of: (1) The balancing tail load corresponding with the condition at speed V and the specified value of the normal load factor n, plus (2) the maneuvering load increment due to the specified value of the angular acceleration.

NOTE: The maneuvering load increment of Figure 3–4 and the distributions of Figure 3–8 (for down loads) and Figure 3–9 (for up loads) may be used [unless the Administrator finds it results in unrealistic loads.] *These distributions apply to the total tail load.*

3.216–1 *Time histories of pull-up maneuvers (CAA policies which apply to sec. 3.216).*

3.216–2 *Unchecked pull-up maneuver (CAA policies which apply to sec. 3.216 (a)).*

(a) The condition given in section 3.216 (a) represents what may occur in an "unchecked" pull-up maneuver. The basic assumption is that while the airplane is flying in steady level flight at the speed V_p, the pilot suddenly pulls the elevator control back and holds it in the full back position.

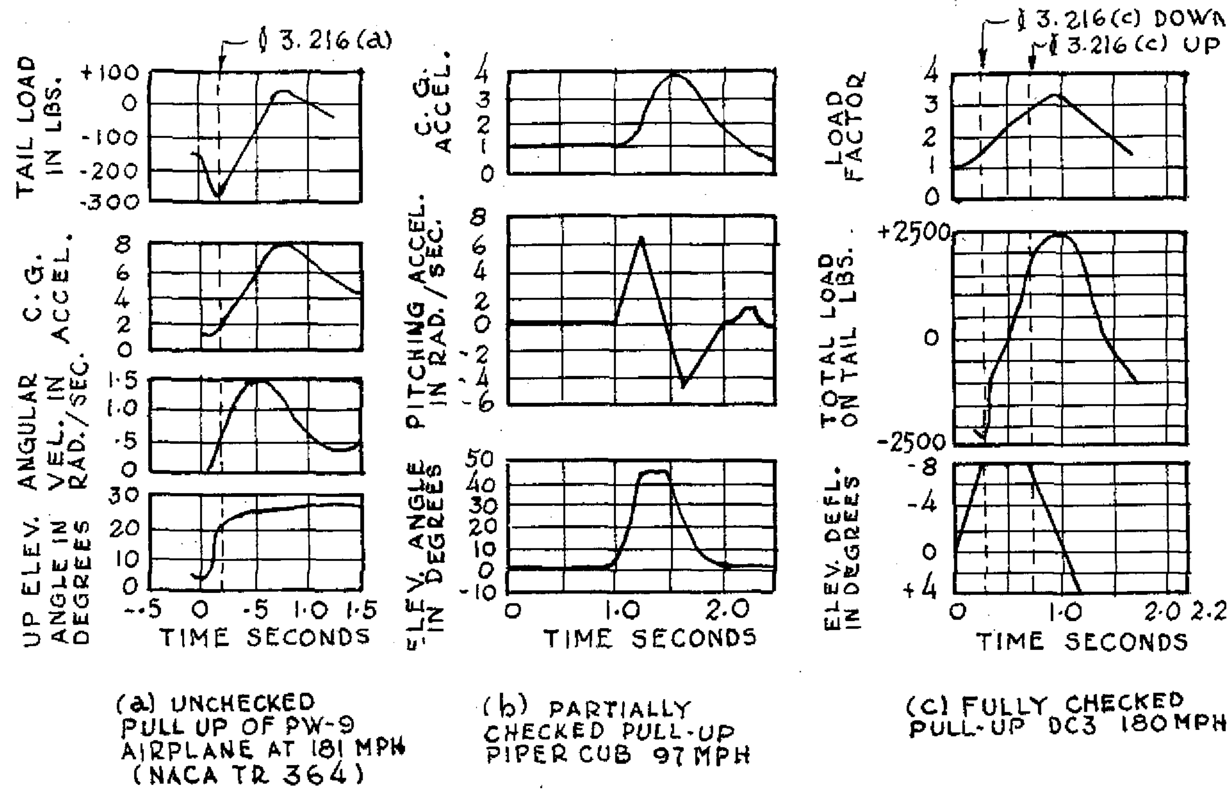


Figure 1.—Time histories of pull-up maneuvers.
(Supp. 10, 16 F. R. 3286, Apr. 14, 1951)

(1) An example of the time history of a typical case of an unchecked pull-up maneuver is shown in Figure 1 (a) (see sec. 3.216–1). It will be noted from this figure that approximately full elevator deflection was applied in roughly 0.1 second and that the elevator was held in the full up position until after the peak c. g. acceleration was obtained. It will also be noted that the maximum down tail load was attained before the airplane had a chance to pitch appreciably since the c. g. acceleration corresponding to maximum down tail load was approximately 1.5.

(2) This condition is intended to represent the condition obtained at the instant of maximum down tail load in an unchecked pull-up as shown on the Figure 1 (a) (see sec. 3.216–1) at the time of approximately 0.15 second.

(b) For purposes of simplifying analysis procedure the download applied to the horizontal tail surface may be carried forward to the wing attachment points, assuming that the fuselage load factor is equal to zero. The moment at the wing due to the above described loads need not be balanced out as a couple at the wing attachment points. However, the linear and angular inertia forces may be taken into account if desired.

(Supp. 10, 16 F. R. 3287, Apr. 14, 1951.)

3.216–3 *Unchecked push-down maneuvering load (CAA policies which apply to sec. 3.216 (b)).* The condition given in section 3.216 (b) represents and "unchecked" push-down and is identical to section 3.216 (a) in principle, except that sudden application of full forward stick is assumed. To simplify the analysis the up load applied to the horizontal tail surfaces may be carried through the attachment of the horizontal tail surfaces to the fuselage, and local fuselage members. No other structure need be investigated for this condition.

(Supp. 10, 16 F. R. 3287, Apr. 14, 1951.)

3.216–4 *Checked maneuvering load condition (CAA policies which apply to sec. 3.216 (c)).*

(a) The condition given in section 3.216 (c) involves a down load and up load corresponding to what may occur in a "checked maneuver."

(b) A "checked maneuver" is defined as one in which the pitching control is suddenly displaced in one direction and then suddenly moved in the opposite direction, the deflections and timing being such as to avoid exceeding the limit maneuvering load factor.

(c) A typical case of a fully checked pull-up maneuver is shown for the DC–3 airplane in Figure 1 (c) (see 3.216–1). This figure will be briefly reviewed as it contains all of the information essential to explaining the down load and up load cases required by section 3.216 (c).

(1) It will be noted that 8 degrees of up elevator was obtained in approximately 0.2 second. This 0.2 second time is the time at which the critical down load case occurs. It will be noted that a maximum down tail load of approximately 2,500 pounds is obtained at this point; further, that the airplane load factor is only slightly over 1 g. (The requirements specify a load factor of 1.0 for simplicity.) As time increases, it will be noted that the load factor begins to build up but that, when the load factor had been built up to approximately 2.7 g, the pilot started to push forward rapidly on the elevator control. This pushing forward is called "checking" and at speeds above the maneuvering speed such "checking" is required in order to prevent the airplane from exceeding the limit maneuvering factor. It will be noted that at the end of one second, the elevator has been completely "checked" back to zero deflection and that the maximum up tail load was obtained at this point concurrent with the maximum load factor at 3.2 g. The condition occurring at this time (1.0 second) represents the critical up tail load condition of section 3.216 (c).

(Supp. 10, 16 F. R. 3287, Apr. 14, 1951.)

3.216–5 *Principles applicable to detailed analysis of conditions given in section 3.216 (CAA policies which apply to sec. 3.216).*

(a) The basic principles underlying detailed analysis for the conditions covered in section 3.216 (a), (b) and (c) are described below:

(1) For the down load case, a normal acceleration of 1.0 is specified, concurrent with a specified positive value of angular acceleration. The forces acting on the airplane should therefore satisfy the following conditions:

(i) The algebraic sum of the upload on the wing and down load on the tail should equal the weight of the airplane. (For analysis purposes, a reasonable approximation to this condition is satisfactory.)

(ii) The summation of wing, fuselage and tail moments about the center of gravity of the airplane should be equal to the pitching moment of inertia of the airplane multiplied by the specified angular acceleration.

(2) The analysis of the upload condition may be carried out in the same manner, except that "n_m" times the weight of the airplane is used in subparagraph (1) (i) of this paragraph.

(b) In all of the conditions covered in section 3.216 (c), the thrust may be assumed zero for simplicity. There are many computation procedures by which these conditions can be satisfied. An example of a typical method is that given in Navy Specification SS-1A. In Figure 3-4 of this part, the maneuvering tail load increment has been based on average values of the ratio of airplane pitching inertia to overall length.

(c) Conditions specified by this requirement are likely to be critical only at speeds V_p and V_d. Investigation has shown that at V_p the specified down load condition is adequately taken care of by section 3.216 (a) and that the specified upload condition is adequately taken care of by section 3.216 (b). For these reasons, the conditions of section 3.216 (c) need not be investigated at the speed V_p.

(Supp. 10, 16 F. R. 3287, Apr. 14, 1951.)

3.216-6 *Maneuvering control surface loading figure 3-3 (b) in this part (CAA policies which apply to sec. 3.216).*

(a) The curves on Figure 3-3 (b) in section 3.216 were derived as follows:

(1) The three curves A, B, and C of Figure 3-3 (b) giving control surfaces loading vs. W/S correspond to normal force coefficients of 0.80, 0.70, and 0.55 respectively. These curves represent psf loading obtained with the above normal force coefficients acting at a design speed of V_p based on the assumption of $C_{L_{\max}}$ equals 1.5.

(2) The basic computations for these curves were as follows:

$$V_p = V_s\sqrt{n}$$

$$q_p = 0.00256\ V_p^2 = 0.00256\ n\ V_s^2$$

$$V_s^2 = \frac{(W/S)}{0.00256 C_{L_{\max}}}$$

$$q_p = \frac{n(W/S)}{C_{L_{\max}}} = \frac{(W/S)}{1.5}$$

$$\overline{w} = C_n q_p = \frac{C_n}{1.5}\, n(W/S)$$

$$4.4\, w = \frac{4.4.\ C_n}{1.5}\,(W/S)$$

(3) These curves are all straight line curves and can be extended as straight lines to give the correct pounds per square foot loadings on the surface on the same basis as given above.

(Supp. 10, 16 F. R. 3287, Apr. 14, 1951.)

3.217 *Gust loads.* The horizontal tail surfaces shall be designed for loads occurring in the conditions specified in paragraphs (a) and (b) of this section.

(a) Positive and negative gusts of 30 feet per second nominal intensity at speed V_c corresponding with the flight condition specified in section 3.187 (a) with flaps retracted.

NOTE: The average loadings of Figures 3-5 (a) and (b) and the distribution of Figure 3-9 may be used for the total tail loading in this condition [unless the Administrator finds it results in unrealistic loads.]

(b) Positive and negative gusts of 15 feet per second nominal intensity at speed V_d corresponding with the flight condition specified in section 3.190 (b) with flaps extended and at speed V_d corresponding with the flight condition specified in section 3.187 (b) with flaps retracted.

(c) In determining the total load on the horizontal tail for the conditions specified in paragraphs (a) and (b) of this section, the initial balancing tail loads shall first be determined for steady unaccelerated flight at the pertinent design speeds V_f, V_c, V_d. The incremental tail load resulting from the gust shall be added to the initial balancing tail load to obtain the total tail load.

NOTE: The incremental tail load due to the gust may be computed by the following formula:

$$\Delta t = 0.1KUVS_t a_t \left[1 - \frac{36 a_w}{R_w}\right]$$

where:

Δt = the limit gust load increment on the tail in pounds.

K=gust coefficient K in section 3.188.
U=nominal gust intensity in feet per second.
V=airplane speed in mines per hour.
S_t=tail surface area in square feet,
a_t=slope of lift curve of tail surface, C_L per degree, corrected for aspect ratio,
a_w=slope of lift curve of wing, C_L per degree, and
R_w=aspect ratio of the wing.

3.217–1 *Gust loads: horizontal tail surfaces (CAA policies which apply to sec. 3.217).* The specified up gust and down gust load may be carried through the fuselage structure to the wing attachment points, assuming that the fuselage load factor is equal to that given by positive and negative gusts of 30 fps at V_c respectively. The angular inertia forces in general produce reliving loads and may be taken into account if desired. The attachments of concentrated mass items in the rear portion of the fuselage may be critically loaded by pitching acceleration forces.

(Supp. 10, 16 F. R. 3292, Apr. 14, 1951.)

3.218 *Unsymmetrical loads.* The maximum horizontal tail surface loading (load per unit area), as determined by the preceding sections, shall be applied to the horizontal surfaces on one side of the plane of symmetry and the following percentage of that loading shall be applied on the opposite side:

$\%=100-10\ (n-1)$ where:
n is the specified positive maneuvering load factor.

In any case the above value shall not be greater than 80 percent.

Vertical Tail Surfaces

3.219 *Maneuvering loads.* At all speeds up to V_p:

(a) With the airplane in unaccelerated flight at zero yaw, a sudden displacement of the rudder control to the maximum deflection as limited by the control stops or pilot effort, whichever is critical, shall be assumed.

NOTE: The average loading of Figure 3–3 and the distribution of Figure 3–8 may be used [unless the Administrator finds it results in unrealistic loads.]

(b) The airplane shall be assumed to be yawed to a sideslip angle of 15 degrees while the rudder control is maintained at full deflection (except as limited by pilot effort) in the direction tending to increase the sideslip.

NOTE: The average loading of Figure 3–3 and the distribution of Figure 3–7 may be used [unless the Administrator finds it results in unrealistic loads.]

(c) The airplane shall be assumed to be yawed to a sideslip angle of 15 degrees while the rudder control is maintained in the neutral position (except as limited by pilot effort). The assumed sideslip angles may be reduced if it is shown that the value chosen for a particular speed cannot be exceeded in the cases of steady slips, uncoordinated rolls from a steep bank, and sudden failure of the critical engine with delayed corrective action.

NOTE: The average loading of Figure 3–3 and the distribution of Figure 3–9 may be used [unless the Administrator finds it results in unrealistic loads.]

3.219–1 *Vertical surface maneuvering loads (CAA policies which apply to sec. 3.219).*

(a) The specified maneuvering loads may be applied to the vertical surfaces and carried through the fuselage structure to the wing attachment points, assuming the lateral inertia load factor along the fuselage structure as zero. The wing drag bracing through the fuselage should be analyzed for this condition since the wings will furnish a large part of the resisting angular inertia. Angular inertia forces on the fuselage may be included if desired.

(Supp. 10, 16 F. R. 3287, Apr. 14, 1951.)

(1) When the Figures 3–3, 3–7, 3–8, and 3–9 are used to compute the specified maneuvering loads on the vertical tail surfaces, it is not necessary to include the lg balancing load for unaccelerated flight which acts on the horizontal tail surfaces in considering the effects of the vertical tail loads on the fuselage.

(2) When rational methods are used, the maneuvering loads on the vertical tail surfaces and the lg horizontal balancing tail load should be applied simultaneously for the structural loading condition.

(20 F. R. 6246, August 26, 1955. Effective September 19, 1955.)

3.220 *Gust loads.*

(a) The airplane shall be assumed to encounter a gust of 30 feet per second nominal

intensity, normal to the plane of symmetry while in unaccelerated flight at speed V_c.

(b) The gust loading shall be computed by the following formula:

$$\bar{w}=\frac{KUVm}{575}$$

where:

$\bar{w}$=average limit unit pressure in pounds per square foot.

$K=1.33-\frac{4.5}{(W/S_v)^{3/4}}$, except that K shall not be less than 1.0. A value of K obtained by rational determination may be used.

U=nominal gust intensity in feet per second.

V=airplane speed in miles per hour.

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

W=design weight in pounds.

S_v=vertical surface area in square feet.

(c) This loading applies only to that portion of the vertical surfaces having a well-defined leading edge.

NOTE: The average loading of Figure 3–6 and the distribution of Figure 3–9 may be used [unless the Administrator finds it results in unrealistic loads.]

3.220–1 *Gust loads; vertical tail surfaces* (*CAA policies which apply to sec. 3.220*).

(a) The K factor specified in section 3.220 was derived from the K factor for vertical gusts (§ 3.188) on the assumption that the effective area of the airplane for lateral gusts is twice the vertical surface area. Substituting $2S_v$ in place of S in the formula of section 3.188, we obtain:

$$K=1.33-\frac{2.67}{\left(\frac{W}{2S_v}\right)^{3/4}}$$

$$=1.33-\frac{4.50}{\left(\frac{W}{S_v}\right)^{3/4}}$$

(b) The specified gust loads may be applied to the vertical surfaces and carried through the fuselage structure to the wing attachment points as described in section 3.219–1.

(Supp. 10, 16 F. R. 3287, Apr. 14, 1951.)

3.221 *Outboard fins.* When outboard fins are carried on the horizontal tail surface, the tail surfaces shall be designed for the maximum horizontal surface load in combination with the corresponding loads induced on the vertical surfaces by end plate effects. Such induced effects need not be combined with other vertical surface loads. When outboard fins extend above and below the horizontal surface, the critical vertical surface loading (load per unit area) as determined by sections 3.219 and 3.220 shall be applied:

(a) To the portion of the vertical surfaces above the horizontal surface, and 80 percent of that loading applied to the portion below the horizontal surface,

(b) To the portion of the vertical surfaces below the horizontal surface, and 80 percent of that loading applied to the portion above the horizontal surface.

Ailerons, Wing Flaps, Tabs, Etc.

3.222 *Ailerons.* **[Paragraphs (c) through (e) and the note following paragraph (b) (3) of this section shall not be applicable to airplanes for which the Administrator finds them to result in unrealistic loads.]**

(a) In the symmetrical flight conditions (see sections 3.183–3.189), the ailerons shall be designed for all loads to which they are subjected while in the neutral position.

(b) In unsymmetrical flight conditions (see sec. 3.191 (a)), the ailerons shall be designed for the loads resulting from the following deflections except as limited by pilot effort:

(1) At speed V_p it shall be assumed that there occurs a sudden maximum displacement of the aileron control. (Suitable allowance may be made for control system deflections.)

(2) When V_c is greater than V_p, the aileron deflection at V_c shall be that required to produce a rate of roll not less than that obtained in condition (1).

(3) At speed V_d the aileron deflection shall be that required to produce a rate of roll not less than one-third of that which would be obtained at the speed and aileron deflection specified in condition (1).

NOTE: For conventional ailerons, the deflections for conditions 2 and 3 may be computed from:

$$\delta_2=\frac{V_p}{V_c}\,\delta_1; \qquad \text{and } \delta_3=\frac{0.5V_p}{V_d}\,\delta_1;$$

where:

δ_1=total aileron deflection (sum of both aileron deflections) in condition (1).

δ_2=total aileron deflection in condition (2).

δ_3=total deflection in condition (3). In the equation for δ_3, the 0.5 factor is used instead of 0.33 to allow for wing torsional flexibility.

(c) The critical loading on the ailerons should occur in condition (2) if V_d is less than $2V_c$ and the wing meets the torsional stiffness criteria. The normal force coefficient C_N for the ailerons may be taken as 0.04δ, where δ is the deflection of the individual aileron in degrees. The critical condition for wing torsional loads will depend upon the basic airfoil moment coefficient as well as the speed, and may be determined as follows:

$$\frac{T_3}{T_2}=\frac{(C_m-.01\delta_{3_1})\,V_d^2}{(C_m-.01\delta_{2_1})\,V_c^2}$$

where:

T_3/T_2 is the ratio of wing torsion in condition (b) (3) to that in condition (b) (2).

δ_{2_1} and δ_{3_1} are the down deflections of the individual aileron in conditions (b) (2) and (3) respectively.

(d) When T_3/T_2 is greater than 1.0 condition (b) (3) is critical; when T_3/T_2 is less than 1.0 condition (b) (2) is critical.

(e) In lieu of the above rational conditions the average loading of Figure 3–3 and the distribution of Figure 3–10 may be used.

3.223 *Wing flaps.* Wing flaps, their operating mechanism, and supporting structure shall be designed for critical loads occurring in the flap-extended flight conditions (see sec. 3.190) with the flaps extended to any position from fully retracted to fully extended; except that when an automatic flap load limiting device is employed these parts may be designed for critical combinations of air speed and flap position permitted by the device. (Also see secs. 3.338 and 3.339.) The effects of propeller slipstream corresponding to take-off power shall be taken into account at an airplane speed of not less than $1.4V_s$ where V_s is the computed stalling speed with flaps fully retracted at the design weight. For investigation of the slipstream condition the airplane load factor may be assumed to be 1.0.

3.223–1 *Wing flap load distribution (CAA policies which apply to sec. 3.223).* A trapezoidal chord load distribution with the leading edge twice the trailing edge loading is acceptable. (Note that these loadings apply in the up direction only; however, it is recommended that the supporting structure also be designed to withstand a down load equal to 25 percent of the up load.)

(Supp. 10, 16 F. R. 3288, Apr. 14, 1951.)

3.224 *Tabs.* Control surface tabs shall be designed for the most severe combination of air speed and tab deflection likely to be obtained within the limit *V-n* diagram (Fig. 3–1) for any usable loading condition of the airplane.

3.224–1 *Trim tab design (CAA policies which apply to sec. 3.224).*

(a) To provide ruggedness and for emergency use of tabs, it is recommended that trim tabs, their attachments and actuating mechanism be designed for loads corresponding to full tab deflection at speed V_c with main surface neutral; except that the tab deflection need not exceed that which would produce a hinge moment on the main surface corresponding to maximum pilot effort.

(b) A trapezoidal chord load distribution with the loading of the leading edge twice that of the trailing edge is acceptable.

(Supp. 10, 16 F. R. 3288, Apr. 14, 1951.)

3.225 *Special devices.* The loading for special devices employing aerodynamic surfaces, such as slots and spoilers, shall be based on test data.

Control System Loads

3.231 *Primary flight controls and systems.*

(a) Flight control systems and supporting structures shall be designed for loads corresponding to 125 percent of the computed hinge moments of the movable control surface in the conditions prescribed in sections 3.211 to 3.225, subject to the following maxima and minima:

(1) The system limit loads need not exceed those which can be produced by the pilot and automatic devices operating the controls.

(2) The loads shall in any case be sufficient to provide a rugged system for service use, including consideration of jamming, ground gusts, taxying tail to wind, control inertia, and friction.

(b) Acceptable maximum and minimum pilot loads for elevator, aileron, and rudder controls are shown in Figure 3–11. These pilot loads shall be assumed to act at the appropriate control grips or pads in a manner simulating flight conditions and to be reacted at the attachments of the control system to the control surface horn.

3.231–1 *Hinge moments* (*CAA policies which apply to sec. 3.231* (*a*)). The 125 percent factor on computed hinge moments provided in section 3.231 (a) need be applied only to elevator, aileron and rudder systems. A factor as low as 1.0 may be used when hinge moments are based on test data; however, the exact reduction will depend to an extent upon the accuracy and reliability of the data. Small scale wind tunnel data are generally not reliable enough to warrant elimination of the factor. If accurate flight test data are used, the factor may be reduced to 1.0.

(Supp. 10, 16 F. R. 3288, Apr. 14, 1951.)

3.231–2 *System limit loads* (*CAA policies which apply to sec. 3.231* (*a*) (*1*)).

(a) When the autopilot is acting in conjunction with the human pilot, the autopilot effort need not be added to human pilot effort, but the autopilot effort should be used for design if it alone can produce greater control system loads than the human pilot.

(b) When the human pilot acts in opposition to the autopilot, that portion of the system between them should be designed for the maximum effort of human pilot or autopilot, whichever is the lesser.

(Supp. 10, 16 F. R. 3288, Apr. 14, 1951.)

3.231–3 *Interconnected control systems on two-control airplanes* (*CAA policies which apply to sec. 3.231*).

(a) With respect to interconnected control systems such as in two control airplanes, the following is recommended in showing the "same level of safety" specified in section 3.10.

(1) If, in the case of two or more interconnected control systems, the control wheel or stick forces due to combined control system loads resulting from air loads on the control surfaces are less than the minimum prescribed in Figure 3–11 of this part, each control system from the interconnection to the control surface should be designed for minimum pilot effort on the control wheel or stick in order that sufficient ruggedness be incorporated into the system.

(2) If the control wheel or stick forces due to combined control system loads resulting from air loads on the control surfaces are in excess of the maximum forces prescribed in Figure 3–11 of this part, it is considered permissible to divide the maximum pilot effort loads in the control systems from the point of interconnection to the control surfaces in proportion to the control surface air loads. However, the load in each such control system should be increased 25 percent to allow for any error in the determination of the control surface loads, but in no case need the resulting loads in any one system exceed the total pilot effort, if the pilot effort were applied to that system alone. In any case, the minimum load in any one system should be no less than that specified in subparagraph (1) of this paragraph.

(Supp. 10, 16 F. R. 3288, Apr. 14, 1951.)

3.232 *Dual controls.* When dual controls are provided, the systems shall be designed for the pilots operating in opposition, using individual pilot loads equal to 75 percent of those obtained in accordance with section 3.231, except that the individual pilot loads shall not be less than the minimum loads specified in Figure 3–11.

3.233 *Ground gust conditions.*

(a) The following ground gust conditions shall be investigated in cases where a deviation from the specific values for minimum control forces listed in Figure 3–11 is applicable. The following conditions are intended to simulate the loadings on control surfaces due to ground gusts and when taxying with the wind.

(b) The limit hinge moment H shall be obtained from the following formula:

$$H = K_c S q$$

where:

H=limit hinge moment (foot-pounds).

c=mean chord of the control surface aft of the hinge line (feet).

S=area of control surface aft of the hinge line (square feet).

q=dynamic pressure (pounds per square foot) to be based on a design speed not less than $10\sqrt{W/S}+10$ miles per hour, except that the design speed need not exceed 60 miles per hour.

K=factor as specified below:

Surface	***K***
(a) Aileron	**+0.75**
Control column locked or lashed in mid-position.	
(b) Aileron	**±0.50**
Ailerons at full throw; + moment on one aileron, − moment on the other.	
(c) (d) Elevator	**±0.75**
Elevator (c) full up (−), and (d) full down (+).	
(e) (f) Rudder	**±0.75**
Rudder (e) in neutral, and (f) at full throw.	

(c) As used in paragraph (b) in connection with ailerons and elevators, a positive value of K indicates a moment tending to depress the surface while a negative value of K indicates a moment tending to raise the surface.

3.233–1 *Ground gust loads (CAA policies which apply to sec. 3.233).* Section 3.233 requires ground gust loads to be investigated when a reduction in minimum pilot effort loads is desired. In such cases the entire system shall be investigated for ground gust loads. However, in instances where the designer desires to investigate ground gust loads without intending to reduce pilot effort loads, the ground gust load need be carried only from the control surface horn to the nearest stops or gust locks, including the stops or locks and their supporting structures.

(Supp. 1, 12 F. R. 3436, May 28, 1947, as amended by Amdt. 1, 14 F. R. 36, Jan. 5, 1949.)

3.234 *Secondary controls and systems.* Secondary controls, such as wheel brakes, spoilers, and tab controls, shall be designed for the loads based on the maximum which a pilot is likely to apply to the control in question.

Ground Loads

3.241 *Ground loads.* The loads specified in the following conditions shall be considered as the external loads and the inertia forces which occur in an airplane structure. In each of the ground load conditions specified the external reactions shall be placed in equilibrium with the linear and angular inertia forces in a rational or conservative manner.

3.241–1 *Four-wheel type alighting gears (CAA policies which apply to sec. 3.241).* At present, little operational data or other information are available on which to base requirements for airplanes equipped with four wheel type alighting gears. The following is suggested for applying the requirements of this part to aircraft equipped with four wheel type alighting gears.

(a) The provisions of sections 3.241 through 3.256, except for the following, should be considered applicable: Sections 3.245 (a), 3.246 (a), and 3.250 through 3.252.

(b) The conditions as specified in sections 3.245 (b) (2), 3.246 (b), 3.247 and 3.249 should be considered applicable to four wheel type gear without modification, the rear wheels being considered the main gear.

(c) The landing conditions specified in section 3.245 (b) (1) should be modified by dividing the total required load on the forward gear between the two wheels, 60 percent to one wheel and 40 percent to the other.

(d) The requirements of section 3.253 should be modified by applying the required loads simultaneously to the two front wheels, 120 percent to one wheel and 80 percent to the other. (Note that this gives an 80–40 percent distribution of the total load on the front gear.)

(e) It is believed that the method of applying the requirements of this part for single nose wheel type alighting gear to four wheel type alighting gear should result in a satisfactory design. It is suggested, however, that sufficient landing and taxiing tests be conducted to determine the suitability of the landing gear design and configuration. Since higher speed turns should be possible with a four wheel aircraft than with one having a conventional tricycle gear, it is believed that provision should be made to include high speed turns in the taxiing test programs of all four wheel aircraft.

(f) If an aircraft with four wheel type alighting gear is also designed for roadability, i. e. for use as an automobile, which is usually the case, the design of the alighting gear in accordance with applicable motor vehicle design requirements is acceptable, provided it can be shown that these requirements fully cover the airworthiness requirements of the regulations in this subchapter.

(Supp. 10, 16 F. R. 3288, Apr. 14, 1951.)

3.241–2 *Ground load evaluation for aircraft with wing tip-tanks (CAA policies which apply to sec. 3.241).* The assumption of the aircraft structure as a rigid body in applying the ground load conditions of sections 3.241 through 3.243 is not considered directly applicable to aircraft which incorporate tip-tanks. When such a design feature is present, the dynamic response of the wing to the short-period landing load impulse may induce inertia loadings of the wing which are significantly higher than the rigid body inertia loading and which create critical wing loadings greater than all other wing design conditions. Accordingly, neglect of the wing inertia loadings due to dynamic response of the wing structure under the landing loads may render the aircraft unsafe. Therefore, the dynamic inertia loading of the airplane wing structure should be considered in evaluating the design loads under the ground load conditions when tip-tanks are present in accordance with the following:

(a) Only the two-wheel level landing condition, section 3.245 (a) or (b) (2), need be considered in substantiating the structural strength of the wing, wing tip-tank and wing fuselage attaching structure for dynamic loads.

(b) The spanwise inertia loading should be determined by either of the following methods:

(1) An engineering evaluation which conservatively provides for the effect of dynamic response. One acceptable method of dynamic landing load analysis is given in CAA Engineering Report No. 52, entitled "Outline of an Acceptable Method of Determining Dynamic Landing Loads."

(2) Dynamic tests of the complete aircraft consisting of static drop tests or in-flight landing tests wherein suitable test instrumentation is used to evaluate the design variation of the vertical inertia load factor from the aircraft centerline to the wing tip under the landing impact.

(19 F. R. 8653, Dec. 17, 1954. Effective Jan. 15, 1955.)

3.242 *Design weight.* The design landing weight shall not be less than the maximum weight for which the airplane is to be certificated, except as provided in paragraph (a) or (b) of this section.

(a) A design landing weight equal to not less than 95 percent of the maximum weight shall be acceptable if it is demonstrated that the structural limit load values at the maximum weight are not exceeded when the airplane is operated over terrain having the degree of roughness to be expected in service at all speeds up to the take-off speed. In addition, the following shall apply:

(1) The minimum fuel capacity shall not be less than the total of the capacity prescribed in section 3.440 and of the capacity equivalent to the weight of fuel equal in amount to that by which the maximum weight exceeds the design landing weight.

(2) The operating limitations shall limit the take-off weight in such a manner as to assure that landings in normal operation would not exceed the design landing weight.

(b) A design landing weight equal to less than 95 percent of the maximum weight shall be acceptable for multi-engine airplanes, meeting the one-engine-inoperative climb requirement of section 3.85 (b) or section 3.85a (b) if compliance is shown with the following sections of Part 4b of this subchapter in lieu of the corresponding requirement of this part: The ground load requirements of section 4b.230, the landing gear requirements of sections 4b.331 through 4b.336, and the fuel jettisoning system requirements of section 4b.437.

3.243 *Load factor for landing conditions.* In the following landing conditions the limit vertical inertia load factor at the center of gravity of the airplane shall be chosen by the designer but shall not be less than the value which would be obtained when landing the airplane with a descent velocity, in feet per second, equal to the following value:

$$V=4.4\,(W/S)^{1/4}$$

except that the descent velocity need not exceed 10 feet per second and shall not be less than 7 feet per second. Wing lift not exceeding two-thirds of the weight of the airplane

may be assumed to exist throughout the landing impact and may be assumed to act through the airplane center of gravity. When such wing lift is assumed, the ground reaction load factor may be taken equal to the inertia load factor minus the ratio of the assumed wing lift to the airplane weight. (See sec. 3.354 for requirements concerning the energy absorption tests which determine the limit load factor corresponding to the required limit descent velocities.) In no case, however, shall the inertia load factor used for design purposes be less than 2.67, nor shall the limit ground reaction load factor be less than 2.0, unless it is demonstrated that lower values of limit load factor will not be exceeded in taxying the airplane over terrain having the maximum degree of roughness to be expected under intended service use at all speeds up to take-off speed.

Landing Cases and Attitudes

3.244 *Landing cases and attitudes.* For conventional arrangements of main and nose, or main and tail wheels, the airplane shall be assumed to contact the ground at the specified limit vertical velocity in the attitudes described in sections 3.245–3.247. (See Figs. 3–12 (a) and 3–12 (b) for acceptable landing conditions which are considered to conform with secs. 3.245–3.247.)

3.244–1 *Landing cases and attitudes (CAA policies which apply to sec. 3.244).* The supporting structure as well as the landing gear itself should be capable of withstanding the loads occurring at the critical extension of the shock struts in accordance with Note (2) of Figure 3–12 (a) in section 3.245–1.

(Supp. 10, 16 F. R. 3288, Apr. 14, 1951.)

3.245 *Level landing.*

(a) *Tail wheel type.* Normal level flight attitude.

(b) *Nose wheel type.* Two cases shall be considered:

(1) Nose and main wheels contacting the ground simultaneously,

(2) Main wheels contacting the ground, nose wheel just clear of the ground. (The angular attitude may be assumed the same as in subparagraph (1) of this paragraph for purposes of analysis.)

(c) *Drag components.* In this condition, drag components simulating the forces required to accelerate the tires and wheels up to the landing speed shall be properly combined with the corresponding instantaneous vertical ground reactions. The wheel spin-up drag loads may be based on vertical ground reactions, assuming wing lift and a tiresliding coefficient of friction of 0.8, but in any case the

Condition		Tail wheel type		Nose wheel type		
		Level landing	Tail-down landing	Level landing with inclined reactions	Level landing with nose wheel just clear of ground	Tail-down landing
Reference section		3. 245 (a)	3. 246 (a)	3. 245 (b) (1)	3. 245 (b) (2)	3. 246 (b) (c)
Vertical component at c. g.		nW	nW	nW	nW	nW
Fore and aft component at c. g.		KnW	0	KnW	KnW	0
Lateral component in either direction at c. g.		0	0	0	0	0
Shock absorber extension (hydraulic shock absorber)		Note (2)	Note (2)	Note (2)	Note (2)	Note (2)
Shock absorber deflection (rubber or spring shock absorber)		100%	100%	100%	100%	100%
Tire deflection		Static	Static	Static	Static	Static
Main wheel loads (both wheels)	V_r	$(n-L)W$	$(n-L)W$b/d	$(n-L)W$a′/d′	$(n-L)W$	$(n-L)W$
	D_r	KnW	0	KnWa′/d′	KnW	0
Tail (nose) wheel loads	V_f	0	$(n-L)W$a/d	$(n-L)W$b′/d′	0	0
	D_f	0	0	KnWb′/d′	0	0
Notes		(1) and (3)	-------------	(1)	(1) and (3)	(3)

NOTE (1). K may be determined as follows: $K=0.25$ for $W=3{,}000$ pounds or less; $K=0.33$ for $W=6{,}000$ pounds or greater, with linear variation of K between these weights.

NOTE (2). For the purpose of design, the maximum load factor shall be assumed to occur throughout the shock absorber stroke from 25 percent deflection to 100 percent deflection unless demonstrated otherwise, and the load factor shall be used with whatever shock absorber extension is most critical for each element of the landing gear.

NOTE (3). Unbalanced moments shall be balanced by a rational or conservative method.

NOTE (4). L is defined in section 3.353.

Figure 3–12 (a).—Basic landing condtions.

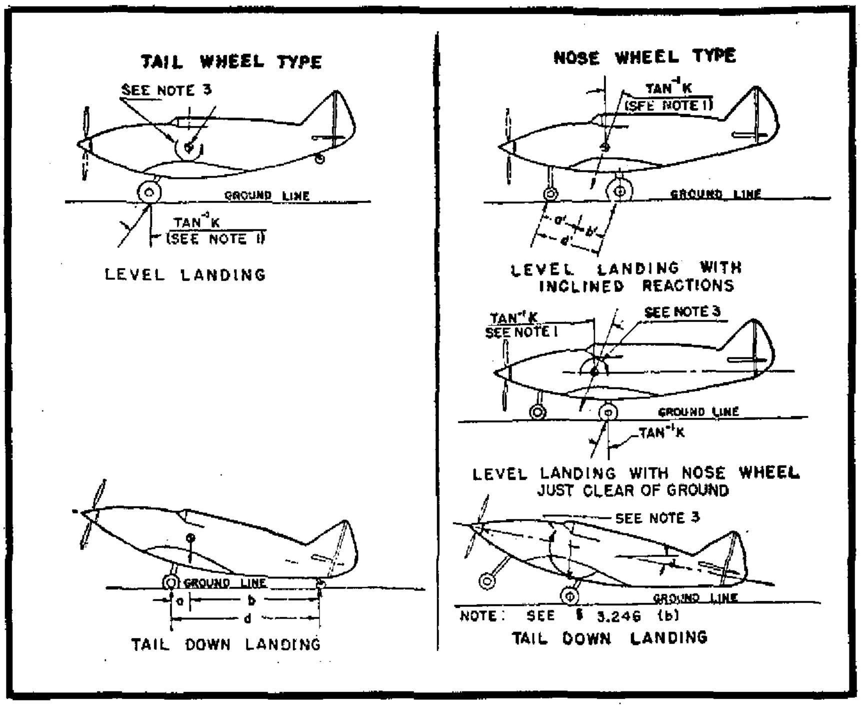


Figure 3–12 (b).—Basic landing conditions.

drag loads shall not be less than 25 percent of the maximum vertical ground reactions neglecting wing lift.

3.245–1 *Wheel spin-up loads (CAA policies which apply to sec. 3.245).*

(a) Section 3.245 requires that spin-up loads be taken into account in structural designs. Section 3.244 permits the use of arbitrary drag loads for this purpose.

(b) If it is desired to use a method more rational than the arbitrary drag components referred to in section 3.244 in determining the wheel spin-up loads for landing conditions, the Administrator will accept the following methods from NACA T. N. 863 for this purpose (however, the minimum drag component of 0.25 times the vertical component will still apply):

$$F_{H\max}=\frac{1}{r_e}\sqrt{\frac{2I_w(V_H-V_c)nF_{V\max}}{t_z}}$$

where

$F_{H\max}$=maximum rearward horizontal force acting on the wheel-pounds.

r_e=effective rolling radius of wheel under impact-feet based on recommended operating tire pressure (may be assumed equal to the rolling radius under a static load of n_jW_e).

I_w=rotational mass moment of inertia of rolling assembly slug feet required.

V_H=linear velocity of airplane parallel to ground at instant of contact, assumed $1.2V_{s_0}$, in feet per second.

V_c=peripheral speed of tire if prerotation is used (feet per second)—a positive means of pre-rotation should be provided before pre-rotation can be considered.

n=effective coefficient of friction; 0.80 is acceptable.

$F_{V\max}$=maximum vertical force on wheel (pounds)=n_jW_e, where W_e and n_j are defined in sections 3.353 and 3.354.

t_z=time interval between ground contact and attainment of maximum vertical force on wheel (seconds). If the value of $F_{H\max}$ from the above equation exceeds 0.8 $F_{V\max}$, the latter value should be used for $F_{H\max}$.

NOTE: This equation assumes a linear variation of load factor with time until the peak load is reached and under this assumption determines the drag force at the time that the wheel peripheral velocity at radius r_e equals the airplane velocity. Most shock absorbers do not exactly follow a linear variation of load factor with time. Hence, rational or conservative allowances should be made to compensate for these variations. On most landing gears the time for wheel spin-up will be less than the time required to develop maximum vertical load factor for the specified rate of descent and forward velocity. However, for exceptionally large wheels, a wheel peripheral velocity equal to the ground speed may not have been attained at time of maximum vertical gear load. This case is covered by the statement above that the drag spin-up load need not exceed 0.8 of the maximum vertical load.

(c) Dynamic spring-back of the landing gear and adjacent structure at the instant just after the wheels come up to speed may result in dynamic forward acting loads of considerable magnitude. This effect may be simulated in the level landing condition by assuming that the wheel spin-up loads are reversed. Dynamic spring-back is likely to be critical only for landing gear units having wheels of large mass supported by relatively flexible cantilever struts.

(d) The arbitrary drag loads referred to in section 3.244 (Fig. 3–12) are usually sufficient to provide for wheel spin-up except for airplanes having large diameter wheels or high stalling speeds. For the latter, it is recommended that a more rational investigation, such as that described above, be made.

(Supp. 1, 12 F. R. 3436, May 28, 1947, as amended by Amdt. 1, 14 F. R. 36, Jan. 5, 1949.)

3.245–2 *Level landing inclined reaction resultant (CAA policies which apply to sec. 3.245).* In Figure 3–12 (b) in section 3.245–1 the level landing inclined reaction resultant for both tail wheel and nose wheel type landing gears is assumed to pass through the wheel axles.

(Supp. 10, 16 F. R. 3288, Apr. 14, 1951.)

3.246 *Tail down.*

(a) *Tail wheel type.* Main and tail wheels contacting ground simultaneously.

(b) *Nose wheel type.* Stalling attitude or the maximum angle permitting clearance of the ground by all parts of the airplane, whichever is the lesser.

(c) *Vertical ground reactions.* In this condition, it shall be assumed that the ground reactions are vertical, the wheels having been brought up to speed before the maximum vertical load is attained.

3.247 *One-wheel landing.* One side of the main gear shall contact the ground with the airplane in the level attitude. The ground reactions shall be the same as those obtained on the one side in the level attitude. (See sec. 3.245.)

Ground Roll Conditions

3.248 *Braked roll.* The limit vertical load factor shall be 1.33. The attitude and ground contacts shall be those described for level landings in sec. 3.245, with the shock absorbers and tires deflected to their static positions. A drag reaction equal to the vertical reaction at the wheel multiplied by a coefficient of friction of 0.8 shall be applied at the ground contact point of each wheel having brakes, except that the drag reaction need not exceed the maximum value based on limiting brake torque.

3.249 *Side load.* Level attitude with main wheels only contacting the ground, with the shock absorbers and tires deflected to their static positions. The limit vertical load factor shall be 1.33 with the vertical ground reaction divided equally between main wheels. The limit side inertia factor shall be 0.83 with the side ground reaction divided between main wheels as follows:

$0.5W$ acting inboard on one side.

$0.33W$ acting outboard on the other side.

Tail Wheels

3.250 *Supplementary conditions for tail wheels.* The conditions in sections 3.251 and 3.252 apply to tail wheels and affected supporting structure.

3.251 *Obstruction load.* The limit ground reaction obtained in the tail down landing condition shall be assumed to act up and aft through the axle at 45 degrees. The shock absorber and tire may be assumed deflected to their static positions.

3.252 *Side load.* A limit vertical ground reaction equal to the static load on the tail wheel, in combination with a side component of equal magnitude. When a swivel is provided, the tail wheel shall be assumed swiveled 90 degrees to the airplane longitudinal axis, the resultant ground load passing through the axle. When a lock steering device or shimmy damper is provided, the tail wheel shall also be assumed in the trailing position with the

side load acting at the ground contact point. The shock absorber and tire shall be assumed deflected to their static positions.

Nose Wheels

3.253 *Supplementary conditions for nose wheels.* The conditions set forth in sections 3.254–3.256 apply to nose wheels and affected supporting structure. The shock absorbers and tires shall be assumed deflected to their static positions.

3.254 *Aft load.* Limit force components at axle:

Vertical, 2.25 times static load on wheel,
Drag. 0.8 times vertical load.

3.255 *Forward load.* Limit force components at axle:

Vertical, 2.25 times static load on wheel,
Forward, 0.4 times vertical load.

3.256 *Side load.* Limit force components at ground contact:

Vertical, 2.25 times static load on wheel,
Side, 0.7 times vertical load.

Skiplanes

3.257 *Supplementary conditions for skiplanes.* The airplane shall be assumed resting on the ground with one main ski frozen in the snow and the other main ski and the tail ski free to slide. A limit side force equal to *P*/3 shall be applied at the most convenient point near the tail assembly, where *P* is the static ground reaction on the tail ski. For this condition the factor of safety shall be assumed equal to 1.0.

3.257–1 *Type certification of skis (CAA policies which apply to sec. 3.257).* Type certification of skis is not contingent upon compliance with section 3.257 which applies to skiplanes only.

(Supp. 10, 16 F. R. 3288, Apr. 14, 1951.)

3.257–2 *Supplementary conditions for skiplanes (CAA policies which apply to sec. 3.257).*

(a) The following material outlines acceptable supplementary structural conditions for skiplanes with a tricycle type gear, in order to show "the same level of safety" under section 3.10.

(1) To provide adequate strength for normal landing, taxiing and ground handling conditions for skiplanes equipped with a tricycle gear, a limit torque equal to $0.667W$ foot pounds should be separately applied about the vertical axis through the centerline of each main pedestal bearing of each main gear, W being the maximum design weight of the airplane in pounds.

(2) For the nose gear, a limit torque equal to $1.333WK$ foot pounds should be separately applied about the vertical axis through the centerline of the nose gear pedestal bearing, where K is the ratio of the nose gear ground reaction (total of both sides), as determined from section 3.245 (b) (1), proper account being taken of the increase of load on the nose gear due to pitching of the airplane.

(3) In the case of a steerable nose gear, the limit torque on the nose gear need not exceed the pilot effort.

(b) An ultimate factor of safety of 1.5 should be applied to the limit torques specified in paragraph (a) (1), (2) and (3) of this section.

(Supp. 10, 16 F. R. 3288, Apr. 14, 1951.)

3.257–3 *Factor of safety of 1.0 (CAA policies which apply to sec. 3.257).*

(a) The load $P/3$ in section 3.257 is considered an ultimate loading. Therefore the factor of 1.0 is considered an ultimate factor.

(Supp. 10, 16 F. R. 3288, Apr. 14, 1951.)

Water Loads

3.265 *Water load conditions.* The structure of boat and float type seaplanes shall be designed for water loads developed during take-off and landing with the seaplane in any attitude likely to occur in normal operation at appropriate forward and sinking velocities under the most severe sea conditions likely to be encountered. Unless a more rational analysis of the water loads is performed, the requirements of sections 4b.251 through 4b.258 of this subchapter shall apply.

3.265–1 *Float loads (CAA policies which apply to sec. 3.265).*

(a) Floats which are presently certificated on the basis of Part 4a of this subchapter in effect prior to November 9, 1945, are considered satisfactory structurally for installation on airplanes which are designed in accordance with this part.

(b) New float designs which are submitted for approval should be investigated for the structural design requirements of this part.

(Supp. 10, 16 F. R. 3288, Apr. 14, 1951, as amended by Supp. 14, 17 F. R. 9066, Oct. 11, 1952.)

3.265–2 *Water loads; alternate standards (CAA policies which apply to secs. 3.10 and 3.265).* ANC–3 provides a level of safety equivalent to, and may be applied in lieu of section 3.265.

(Supp. 14, 17 F. R. 9066, Oct. 11, 1952.)

[Fatigue Evaluation

[3.270 *Pressurized cabins.* The strength, detail design, and fabrication of the pressure cabin structure shall be evaluated in accordance with the provisions of either paragraph (a) or paragraph (b) of this section.

[(a) *Fatigue strength.* The structure shall be shown by analysis and/or tests to be capable of withstanding the repeated loads of variable magnitude expected in service.

[(b) *Fail safe strength.* It shall be shown by analysis and/or tests that catastrophic failure is not probable after fatigue failure or obvious partial failure of a principal structural element. After such failure the remaining structure shall be capable of withstanding a static ultimate load factor of 75 percent of the limit load factor at V_c, taking into account the combined effect of normal operating pressures, the expected external aerodynamic pressures, and flight loads. These loads shall be multiplied by a factor of 1.15 unless the dynamic effects of failure under static load are otherwise taken into consideration.]

Subpart D—Design and Construction

General

3.291 *General.* The suitability of all questionable design details or parts having an important bearing on safety in operation shall be established by tests.

3.292 *Materials and workmanship.* The suitability and durability of all materials used in the airplane structure shall be established on the basis of experience or tests. All materials used in the airplane structure shall conform to approved specifications which will insure their having the strength and other properties assumed in the design data. All workmanship shall be of a high standard.

3.293 *Fabrication methods.* The methods of fabrication employed in constructing the airplane structure shall be such as to produce consistently sound structure. When a fabrication process such as gluing, spot welding, or heat-treating requires close control to attain this objective, the process shall be performed in accordance with an approved process specification.

3.294 *Standard fastenings.* All bolts, pins, screws, and rivets used in the structure shall be of an approved type. The use of an approved locking device or method is required for all such bolts, pins, and screws. Self-locking nuts shall not be used on bolts subject to rotation during the operation of the airplane.

3.295 *Protection.* All members of the structure shall be suitably protected against deterioration or loss of strength in service due to weathering, corrosion, abrasion, or other causes. In seaplanes, special precaution shall be taken against corrosion from salt water, particularly where parts made from different metals are in close proximity. Adequate provisions for ventilation and drainage of all parts of the structure shall be made.

3.296 *Inspection provisions.* Adequate means shall be provided to permit the close examination of such parts of the airplane as require periodic inspection, adjustments for proper alignment and functioning, and lubrication of moving parts.

Structural Parts

3.301 *Material strength properties and design values.* Material strength properties shall be based on a sufficient number of tests of material conforming to specifications to establish design values on a statistical basis. The design values shall be so chosen that the probability of any structure being understrength because of material variations is extremely remote. Values contained in ANC–5, ANC–18, and ANC–23, Part II shall be used unless shown to be inapplicable in a particular case.

NOTE: ANC–5, "Strength of Metal Aircraft Elements," ANC–18, "Design of Wood Aircraft Structures," and ANC–23, "Sandwich Construction for

Aircraft," are published by the Subcommittee on Air Force-Navy-Civil Aircraft Design Criteria, and may be obtained from the Superintendent of Documents, Government Printing Office, Washington 25, D. C.

3.301–1 *Design properties (CAA policies which apply to sec. 3.301).*

(a) With reference to section 5.00 of ANC–5, Amendment No. 1, allowable design property columns headed "Army-Navy" represent design properties which will be equalled or exceeded by the properties possessed by approximately 90 percent of the material. All other allowable design property columns relate to the minimum guaranteed properties and are based on values given in the various material specifications. The Administrator will permit uses of these design properties as outlined in subparagraphs (1) and (2) of this section, based on the objectives of section 3.301.

(1) In the case of structures where the applied loads are eventually distributed through single members within an assembly, the failure of which would result in the loss of the structural integrity of the component involved, the guaranteed minimum design mechanical properties listed in ANC–5 shall be used.

NOTE: Typical examples of such items are:
1. Wing lift struts.
2. Spars in two-spar wings.
3. Sparcaps in regions such as wing cutouts and wing center sections where loads are transmitted through caps only.
4. Primary attachment fittings dependent on single bolts for load transfer.

(2) Redundant structures wherein partial failure of individual elements would result in the applied load being safely distributed to other load carrying members, may be designed on the basis of the "90 percent probability" allowable.

NOTE: Typical examples of such items are:
1. Sheet-stiffener combinations.
2. Multi-rivet or multiple bolt connections.

(b) Certain manufacturers have indicated a desire to use design value greater than the guaranteed minimums even in applications where only guaranteed minimum values would be permitted under paragraph (a) of this section, and have advocated that such allowables be based on "premium selection" of the material. Such increased design allowables will be acceptable to the Administrator: *Provided*, That a specimen or specimens of each individual item are tested prior to its use, to determine that the actual strength properties of that particular item will equal or exceed the properties used in design. This, in effect, results in the airplane or materials manufacturer guaranteeing higher minimum properties than those given in the basic procurement specifications.

(c) See section 3.174–1 (a).

3.301–2 *Substitution of seam-welded for seamless steel tubing (CAA policies which apply to sec. 3.301).* Seam welded tubing may be substituted for seamless steel tubing as follows:

(a) SAE 4130 welded tubing as per Specification AN–T–3, may be substituted for SAE 4130 seamless tubing conforming to Specification AN–WW–T–850a, and vice versa.

(b) SAE 1025 welded tubing as per Specification AN–T–4, may be substituted for SAE 1025 seamless tubing conforming to Specification AN–WW–T–846, and vice versa.

(c) SAE 8630 welded tubing conforming to Specification AN–T–33a may be substituted for SAE 8630 seamless tubing conforming to Specification AN–T–15 and vice versa.

(Supp. 10, 16 F. R. 3289, Apr. 14, 1951.)

3.302 *Special factors.* Where there may be uncertainty concerning the actual strength of particular parts of the structure or where the strength is likely to deteriorate in service prior to normal replacement, increased factors of safety shall be provided to insure that the reliability of such parts is not less than the rest of the structure as specified in sections 3.303–3.306.

3.303 *Variability factor.* For parts whose strength is subject to appreciable variability due to uncertainties in manufacturing processes and inspection methods, the factor of safety shall be increased sufficiently to make the probability of any part being under-strength from this cause extremely remote. Minimum variability factors (only the highest pertinent variability factor need be considered) are set forth in sections 3.304–3.306.

3.304 *Castings.*

(a) Where visual inspection only is to be employed, the variability factor shall be 2.0.

(b) The variability factor may be reduced to 1.25 for ultimate loads and 1.15 for limit loads when at least three sample castings are tested to show compliance with these factors, and all

sample and production castings are visually and radiographically inspected in accordance with an approved inspection specification.

(c) Other inspection procedures and variability factors may be used if found satisfactory by the Administrator.

3.304–1 *Casting factors (CAA policies which apply to sec. 3.304).* With reference to paragraphs (b) and (c) of section 3.304, the Administrator has approved specific proposals which permit the use of lower casting factors as specified in (b), with 100 percent radiographic inspection on initial runs, but with radiographic inspection gradually reduced on production lots as it becomes evident that adequate quality control has been established. All such procedures require the submittal and execution of a satisfactory process specification and statistical proof that adequate quality control has been achieved.

(Supp. 1, 12 F. R. 3437, May 28, 1947, as amended by Amdt. 1, 14 F. R. 36, Jan. 5, 1949.)

3.305 *Bearing factors.*

(a) The factor of safety in bearing at bolted or pinned joints shall be suitably increased to provide for the following conditions:

(1) Relative motion in operation (control surface and system joints are covered in sections 3.327–3.347).

(2) Joints with clearance (free fit) subject to pounding or vibration.

(b) Bearing factors need not be applied when covered by other special factors.

3.306 *Fitting factor.* Fittings are defined as parts such as end terminals used to join one structural member to another. A multiplying factor of safety of at least 1.15 shall be used in the analysis of all fittings the strength of which is not proved by limit and ultimate load tests in which the actual stress conditions are simulated in the fitting and the surrounding structure. This factor applies to all portions of the fitting, the means of attachment, and bearing on the members joined. In the case of integral fittings, the part shall be treated as a fitting up to the point where the section properties become typical of the member. The fitting factor need not be applied where a type of joint design based on comprehensive test data is used. The following are examples: continuous joints in metal plating, welded joints, and scarf joints in wood, all made in accordance with approved practices.

3.307 *Fatigue strength.* The structure shall be designed, insofar as practicable, to avoid points of stress concentration where variable stresses above the fatigue limit are likely to occur in normal service.

Flutter and Vibration

3.311 *Flutter and vibration prevention measures.* Wings, tail, and control surfaces shall be free from flutter, airfoil divergence, and control reversal from lack of rigidity, for all conditions of operation within the limit *V–n* envelope, and the following detail requirements shall apply:

(a) Adequate wing torsional rigidity shall be demonstrated by tests or other methods found suitable by the Administrator.

(b) The mass balance of surfaces shall be such as to preclude flutter.

(c) The natural frequencies of all main structural components shall be determined by vibration tests or other methods found satisfactory by the Administrator.

3.311–1 *Simplified flutter prevention criteria (CAA policies which apply to secs. 3.311 (a) and (b)).* Compliance with the rigidity and mass balance criteria presented on pages 4 through 12 of CAA Airframe and Equipment Engineering Report No. 45, as corrected February 1952, "Simplified Flutter Prevention Criteria for Personal Type Aircraft", is considered to be an acceptable means of meeting the flutter prevention requirements of secs. 3.311 (a) and (b) with the following limitations:

(a) The wing and aileron flutter prevention criteria as represented by the wing torsional stiffness and aileron balance criteria are limited to aircraft which do not have large mass concentrations located along their wing span. For example, these criteria are not applicable for wings carrying engines, floats, fuel in outer-panels, etc.

(b) The elevator and rudder balance criteria are limited in application to tail surface configurations which include a fixed fin surface and a fixed stabilizer surface.

(Supp. 13, 17 F. R. 2161, Mar. 13, 1952.)

[Discussion of Policies Relating to Flight Flutter Testing in Section 3.311–2

[Although demonstration of freedom from flutter by suitable flight tests in lieu of other flutter prevention substantiation is an acceptable procedure for complying with the requirements of section 3.311 (a) and (b), the acceptance of flight tests is not intended as a recommendation that such tests be used as a general procedure for substantiating freedom from flutter. The application of the simplified criteria of section 3.311–1 when applicable, or substantiation by a flutter analysis are recommended as preferred procedures because of the hazards involved in flight flutter tests.

[Flutter in flight can occur suddenly without forewarning and develop to disastrous amplitudes within a very short interval of time. Therefore, applicants to whom flight flutter testing may appear to be the most expedient procedure are cautioned that such tests may be hazardous with respect to the pilot's life and to the integrity of the prototype airplane. It is recommended that flight flutter tests be performed only when allied investigations or engineering evaluations give some assurance that the tests may be performed safely.

[It is possible for an aircraft to fly within a flutter speed range without indications of actual flutter if a disturbance of sufficient strength and proper character is not experienced to initiate the flutter. Therefore, it is essential to satisfactory completion of the flight flutter test to demonstrate that proper and adequate attempts to induce flutter were performed.

[Two principal types of excitation means have been used in flutter tests:

[1. Continuous variable frequency excitation.

[2. Transient, or impulse excitation.

[The first method is considered to be the more reliable procedure with respect to indicating the possible flutter mode and consequently more reliable in providing a warning of approach to a flutter condition. Hence, from the safety point of view the first method is to be preferred. However, the method involves the installation of special equipment to supply and control the exciting forces. Because of the complexity, weight and volume of such installations, most applicants desiring to substantiate freedom from flutter by flight test select the transient method of excitation. For this reason, the policies set forth in section 3.311–2 describe acceptable procedures only for tests in which transient excitation is used. In its simplest form, this is one in which a rapid displacement of a control surface supplies the aerodynamic disturbing force to excite oscillations of the aircraft structure. It is essential that control surface deflection be large enough to provide an adequate disturbing force and the rapidity of application as sudden as possible in order to excite the higher structural frequencies.

[The flutter stability of the airplane at each airspeed is measured by the character of the vibratory response of the structure to the suddenly applied force. Although simpler, this method of excitation is less reliable in providing a warning of approach to flutter since some latent flutter modes may not be evident in the response of the structure until the actual flutter speed is attained.

[In order to determine that adequate and proper excitation to induce flutter was applied in the tests, the installation of flight recording instrumentation is necessary to provide a permanent record of the control surface deflections applied and the structural response to the applied flutter exciting forces. The rotational motion of the rudder and/or elevator surfaces following their sudden deflection is considered to be an adequate index of the flutter stability of the aircraft tail structure. Therefore, the degree of excitation applied and the structural response can be evaluated from a record of the control surface deflections. However, wing flutter can consist of essentially bending and torsional oscillations of the wing surface with very little associated oscillations of the aileron. Therefore, a record of only the aileron deflection is not sufficient and additional instrumentation is needed to evaluate the response of the wing surface during flight flutter tests of the wing and aileron.

[It is recommended that the records of control surface oscillations and wing motion resulting from the test excitation be examined at each test speed before proceeding to the next higher test speed, particularly as V_d is approached. Flutter stability at each airspeed is indicated by the fact that the resultant motion of the control surface and the associated fixed surface in response to the excitation is "dead beat" or well damped. Above the critical flutter speed, these resultant motions are not damped but diverge—build up to large destructive deflections. As a flutter speed is approached in flight, it is expected that the motion of the surfaces will become less damped. Hence, examination of the control surface and wing motion records provides one possible source of warning that a flutter speed is being approached.

[A flight flutter test program may involve flutter substantiation of two or more control surfaces whose static unbalance values are in excess of the allowable values in the simplified flutter criteria of section 3.311-1. As a precautionary procedure, it is recommended that tests be performed for one surface at a time with the remaining unsubstantiated control surfaces balanced to at least the degree indicated necessary by the simplified flutter criteria.]

[3.311-2 *Substantiation of freedom from flutter by flight flutter tests (CAA policies which apply to sec. 3.311).* Various procedures are available for demonstrating freedom from flutter in complying with the requirements of section 3.311 (a) and (b), namely: performance of a flutter analysis, application of the simplified criteria of section 3.311-1 when applicable, and performance of a flight flutter test by the applicant. It is not recommended that flight flutter tests be used as a general procedure for substantiating freedom from flutter. The performance of a flutter analysis or the application of the simplified flutter criteria are considered preferred procedures due to the hazards involved in flight flutter tests.

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

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

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

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

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

[(i) Aileron control to induce wing and aileron flutter.

[(ii) Rudder control to induce rudder and vertical tail flutter.

[(iii) Elevator control to induce elevator and horizontal tail flutter, and symmetric wing flutter.

[(iv) Control surface to which the tab is attached to induce tab flutter.

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

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

[(5) A permanent record [7a] at each test speed should be obtained as follows:

[(i) In flutter tests of the rudder, elevator and tab surfaces; a time history of the control surface rotational deflection, and the associated airspeed.

[(ii) In flutter tests of the wing and aileron; a time history of the aileron rotational deflection, a time history of the wing vibratory response, and the associated test airspeed.

[(6) The tests should consider significant variations in mass and rigidity values which might be expected in service. The aileron and tab control systems should be freed to the extent necessary to be representative of what might be expected in service. (See Airframe and Equipment Engineering Report No. 45.) In tests of wings incorporating wing-tip fuel tanks, the tip-tank weight during the tests should include the weight which results in the most adverse wing flutter characteristics. An engineering evaluation of the ground vibration survey results may be made to determine which value of tip-tank weight is most critical. In some instances, such an engineering evaluation will not permit selection of any single weight as being most critical and flight flutter tests at several tip-tank weights should be performed.

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

[(8) The vibration survey required under section 3.311 (c) should be conducted prior to performing the flight flutter tests. The structural frequency and vibration mode data should be evaluated to determine what structural modes are most likely to be flutter critical. An engineering evaluation of the wing bending and torsion frequencies as well as the ratios of wing bending to wing torsion frequencies should be made for various wing tip-tank weight conditions to determine the critical tip-tank weight(s) which should be considered in the flight flutter tests.]

(21 F. R. 5188, July 12, 1956, effective Aug. 15, 1956.)

[7a These data can be obtained by installing a control surface position indicating device at the control surface, a vibration pickup in the vicinity of one wing tip to detect the wing response, and a suitable pressure transducer connected to the aircraft's pitot-static system to measure the airspeed with the electrical signals from these instruments connected to a recording oscillograph.

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

Wings

3.317 *Proof of strength.* The strength of stressed-skin wings shall be substantiated by load tests or by combined structural analysis and tests.

3.318 *Ribs.* Rib tests shall simulate conditions in the airplane with respect to torsional rigidity of spars, fixity conditions, lateral support, and attachment to spars. The effects of ailerons and high lift devices shall be properly accounted for.

3.318–1 *Rib tests (CAA policies which apply to sec. 3.318).* See section 3.174–6 (g).

(Supp. 10, 16 F. R. 3289, Apr. 14, 1951.)

Control Surfaces (Fixed and Movable)

3.327 *Proof of strength.* Limit load tests of control surfaces are required. Such tests shall include the horn or fitting to which the control system is attached. In structural analyses, rigging loads due to wire bracing shall be taken into account in a rational or conservative manner.

3.328 *Installation.* Movable tail surfaces shall be so installed that there is no interference between the surfaces or their bracing when each is held in its extreme position and all others are operated through their full angular movement. When an adjustable stabilizer is used, stops shall be provided which, in the event of failure of the adjusting mechanism, will limit its travel to a range permitting safe flight and landing.

3.328–1 *Bonding of control surfaces (CAA policies which apply to sec. 3.328).* In order to avoid possible "freezing" of control surface bearings caused by electrical discharges (as for example when flying through thunderstorms), it is recommended that all control surfaces be bonded to the airframe.

(Supp. 10, 16 F. R. 3289, Apr. 14, 1951.)

3.329 *Hinges.* Control surface hinges, excepting ball and roller bearings, shall incorporate a multiplying factor of safety of not less than 6.67 with respect to the ultimate bearing strength of the softest material used as a bearing. For hinges incorporating ball or roller bearings, the approved rating of the bearing shall not be exceeded. Hinges shall provide sufficient strength and rigidity for loads parallel to the hinge line.

3.330 *Mass balance weights.* The supporting structure and the attachment of concentrated mass balance weights which are incorporated on control surfaces shall be designed for the following limit accelerations: 24g normal to the plane of the control surface, 12g fore and aft, and 12g parallel to the hinge line.

Control Systems

3.335 *General.* All controls shall operate with sufficient ease, smoothness, and positiveness to permit the proper performance of their function and shall be so arranged and identified as to provide convenience in operation and prevent the possibility of confusion and subsequent inadvertent operation. (See sec. 3.384 for cockpit controls.)

3.336 *Primary flight controls.*

(a) Primary flight controls are defined as those used by the pilot for the immediate control of the pitching, rolling, and yawing of the airplane.

(b) For two-control airplanes the design shall be such as to minimize the likelihood of complete loss of the lateral directional control in the event of failure of any connecting or transmitting element in the control system.

3.336–1 *Aileron controls for two-control airplanes (CAA interpretations which apply to sec. 3.336 (b)).* In the case of two-control airplanes having side by side control wheels, the aileron controls in the right wing should be independent of those in the left wing; however, they may be connected to a common bell-crank or lever in the fuselage. From the point of common connection to the control wheels, the margin of safety should be large and the detail design adequate to minimize possibility of failure of any part.

(Supp. 10, 16 F. R. 3289, Apr. 14, 1951.)

3.337 *Trimming controls.* Proper precautions shall be taken against the possibility of inadvertent, improper, or abrupt tab operations. Means shall be provided to indicate to the pilot the direction of control movement relative to airplane motion and the position of the trim device with respect to the range of adjustment. The means used to indicate the direction of the control movement shall be adjacent to the control, and the means used to indicate the position of the trim device shall be easily visible to the pilot and so located and operated as to preclude the possibility of confusion. Longitudinal trimming devices for single-engine airplanes and longitudinal and directional trimming devices for multiengine airplanes shall be capable of continued normal operation notwithstanding the failure of any one connecting or transmitting element in the primary flight control system. Tab controls shall be irreversible unless the tab is properly balanced and possesses no unsafe flutter characteristics. Irreversible tab systems shall provide adequate rigidity and reliability in the portion of the system from the tab to the attachment of the irreversible unit to the airplane structure.

3.337–1 *Independence of bungee trim system from primary control system (CAA interpretations which apply to sec. 3.337).* The sentence "Trimming devices shall be capable of continued normal operation notwithstanding the failure of any one connecting or transmitting element in the primary flight control system" permits a bungee system acting directly on the control surface horn in such a manner that a failure of the primary control system will not adversely affect operation of the trimming device. A bungee system which actuates a control surface through elements of the primary control system does not meet these requirements. Each trim control system will be reviewed on the basis of its individual merits.

(16 F. R. 3289, Apr. 14, 1951, effective Apr. 15, 1951, revised 22 F. R. 135, Jan. 5, 1957, effective Jan. 31, 1957.)

3.337–2 *Electrical trim tab systems (CAA policies which apply to secs. 3.337 and 3.681).*

proval as such for pulleys for general use on aircraft. Approval is limited to its use as a part of a specific airplane design. Conformance with the Army-Navy-Aeronautical Standards, National Aircraft Standards, or with standards established for pulleys previously approved for use in civil aircraft, or adequate substantiation of the manufacturer's own design through stress analysis or tests are the procedures utilized in complying with the "approved specifications" of CAR 3.345.

(20 F. R. 8809, Dec. 1, 1955. Effective Dec. 30, 1955.)

3.346 *Joints.* Control system joints subject to angular motion in push-pull systems, excepting ball and roller bearing systems, shall incorporate a multiplying factor of safety of not less than 3.33 with respect to the ultimate bearing strength of the softest material used as a bearing. This factor may be reduced to 2.0 for such joints in cable control systems. For ball or roller bearings the approved rating of the bearing shall not be exceeded.

3.347 *Spring devices.* The reliability of any spring devices used in the control system shall be established by tests simulating service conditions, unless it is demonstrated that failure of the spring will not cause flutter or unsafe flight characteristics.

Landing Gear

Shock Absorbers

3.351 *Tests.* Shock absorbing elements in main, nose, and tail wheel units shall be substantiated by the tests specified in the following section. In addition, the shock absorbing ability of the landing gear in taxying must be demonstrated in the operational tests of section 3.146.

3.352 *Shock absorption tests.*

(a) It shall be demonstrated by energy absorption tests that the limit load factors selected for design in accordance with section 3.243 will not be exceeded in landings with the limit descent velocity specified in that section.

(b) In addition, a reserve of energy absorption shall be demonstrated by a test in which the descent velocity is at least 1.2 times the limit descent velocity. In this test there shall be no failure of the shock absorbing unit, although yielding of the unit will be permitted. Wing lift equal to the weight of the airplane may be assumed for purposes of this test.

3.352-1 *Landing gear drop tests (CAA policies which apply to sec. 3.352).*

(a) The following method has been approved by the Administrator for determining the effective mass to be dropped in drop tests of nose wheel landing gear assemblies pursuant to section 3.352 (a): For aircraft with nose wheel type gear, the effective mass to be used in free drop test of the nose wheel shall be determined from the formula for W_e (secs. 3.353 and 3.355) using $W=W_n$ where W_n is equal to the vertical components of the resultant force acting on the nose wheel, computed under the following assumptions: (1) the mass of the airplane concentrated at the center of gravity and exerting a force of 1.0 g downward and 0.33 g forward, (2) the nose and the main gears and tires in static position, and (3) the resultant reactions at the main and nose gears acting through the axles and parallel to the resultant force at the airplane center of gravity.

NOTE: By way of explanation, the use of an inclined reactions condition as the basis for determining the mass to be dropped with a nose wheel unit is based on rational dynamic investigation of the landing condition, assuming the landing is made with simultaneous three-point contact, zero pitching velocity, and a drag component representing the average wheel spin-up reactions during the landing impact. Although spin-up loads on small airplanes may be less than the value implied by the formula, such airplanes are more likely to be landed with a nosing down pitching velocity, or in soft ground. The vertical component of the ground reaction is specified above because the method of defining the direction of the inertia force at the center of gravity gives a resultant effective mass greater than that of the airplane.

(b) The following procedure has been approved by the Administrator for determining the attitude in which the landing gear unit should be dropped pursuant to section 3.352 (a): The attitude in which a landing gear unit is dropped shall be that which simulates the airplane landing condition which is critical from the standpoint of energy to be absorbed by the particular unit, thus: (1) For nose wheel type landing gear, the nose wheel gear shall be drop tested in an attitude which simulates the three point landing inclined reaction condition; (2) the attitude selected for main gear drop tests shall be that which simulates the two-wheel level landing with inclined reactions condition.

NOTE: In addition, it is recommended that the main gear be dropped in an attitude simulating the tail-down landing with vertical reactions condition if the geometry of the gear is such that this condition is likely to result in shock strut action appreciably different from that obtained in level attitude drop tests; for example, when a cantilever shock strut has a large inclination with respect to the direction of the ground reaction.

(3) Tail wheel units shall be tested in such a manner as to simulate the tail-down landing condition (three-point contact). Drag components may be covered separately by the tail wheel "obstruction" condition.

(c) The Administrator has accepted the following procedure for determining slopes of inclined platforms when such are used in drop tests: When the arbitrary drag components given on Fig. 3–12 (a) of this part are used for the design of the landing gear in the level landing conditions, the drag loads in the drop tests for these conditions may be simulated by dropping the units onto inclined platforms so arranged as to obtain the proper direction of the resultant ground reactions in relation to the landing gear. (If wheel spin-up loads for these conditions are determined by rational methods and found to be more severe than the arbitrary drag loads, it is suggested that the spin-up loads be simulated by dropping the gear onto a level platform with wheel spinning.) In at least one limit drop test the platform should simulate the friction characteristics of paved runways and the rotational speed of the wheel just prior to contact should correspond to an airplane ground speed of 1.2 V_{s_0}. It is suggested that additional limit drops be made onto surfaces of lower friction coefficient and at several wheel rotational speeds; coefficients for example, corresponding to 0.6, 0.8 and 1.0 V_{s_0}. The direction of wheel rotation in the drop test should be opposite to that which would occur in landing the airplane. Spin-up loads which are slightly greater than the arbitrary drag loads can probably be simulated satisfactorily by inclined platforms, but platforms having greater inclinations may not simulate spin-up loads correctly and are not recommended.

(Supp. 1, 12 F. R. 3437, May 28, 1947, as amended by Amdt. 1, 14 F. R. 36, Jan 5, 1949.)

3.353 *Limit drop tests.*

(a) If compliance with the specified limit landing conditions of section 3.352 (a) is demonstrated by free drop tests, these shall be conducted on the complete airplane, or on units consisting of wheel, tire, and shock absorber in their proper relation, from free drop heights not less than the following:

$$h \text{ (inches)} = 3.6(W/S)^{0.5}$$

except that the free drop height shall not be less than 9.2 inches and need not be greater than 18.7 inches.

(b) In simulating the permissible wing lift in free drop tests, the landing gear unit shall be dropped with an effective mass equal to:

$$W_e = W\left[\frac{h+(1-L)d}{h+d}\right]$$

where:

W_e=the effective weight to be used in the drop test.

h=specified height of drop in inches.

d=deflection under impact of the tire (at the approved inflation pressure) plus the vertical component of the axle travel relative to the drop mass. The value of d used in the computation of W_e) shall not exceed the valuc actually obtained in the drop tests.

W=W_M for main gear units, and shall be equal to the static weight on the particular unit with the airplane in the level attitude (with the nose wheel clear, in the case of nose wheel type airplanes).

W=W_T for tail gear units, and shall be equal to the static weight on the tail unit with the airplane in the tail down attitude.

W=W_N for nose wheel units, and shall be equal to the static reaction which will exist at the nose wheel when the mass of the airplane is concentrated at the center of gravity and exerts a force of 1.0g downward and 0.33g forward.

L=ratio of assumed wing lift to airplane weight, not greater than 0.667.

The attitude in which the landing gear unit is drop tested shall be such as to simulate the airplane landing condition which is critical from the standpoint of energy to be absorbed by the particular unit.

3.354 *Limit load factor determination.* In determining the limit airplane inertia load factor n from the free drop test described above, the following formula shall be used:

$$n=n_j \frac{W_e}{W}+L$$

where:

n_j= the load factor developed in the drop test, i. e., the acceleration (d_v/dt) in g's recorded in the drop test, plus 1.0.

The value of n so determined shall not be greater than the limit inertia load factor used in the landing conditions, section 3.243.

3.355 *Reserve energy absorption drop tests.* If compliance with the reserve energy absorption condition specified in section 3.352 (b) is demonstrated by free drop tests, the drop height shall be not less than 1.44 times the drop height specified in section 3.353. In simulating wing lift equal to the airplane weight, the units shall be dropped with an effective mass equal to

$$W_e=W \frac{h}{h+d}$$

where the symbols and other details are the same as in section 3.353.

Retracting Mechanism

3.356 *General.* The landing gear retracting mechanism and supporting structure shall be designed for the maximum load factors in the flight conditions when the gear is in the retracted position. It shall also be designed for the combination of friction, inertia, brake torque, and air loads occurring during retraction at any air speed up to 1.6 V_{s_1}, flaps retracted and any load factors up to those specified for the flaps extended condition, section 3.190. The landing gear and retracting mechanism, including the wheel well doors, shall withstand flight loads with the landing gear extended at any speed up to at least 1.6 V_{s_1} flaps retracted. Positive means shall be provided for the purpose of maintaining the wheels in the extended position.

3.356–1 *Retracting mechanism (CAA policies which apply to sec. 3.356).*

(a) In order to provide for adequate strength of the landing gear doors, landing gear, etc., in yawed attitude, it will be satisfactory to show compliance with section 3.356 at the maximum yaw angle as determined by the flight characteristic requirements of section 3.105 and at speeds up to 1.6V_{s_1} flaps retracted.

(b) To meet the requirement that a positive means be provided for maintaining wheels in the extended position, a positive mechanized lock or latch should be provided that can be released directly or sequentially only by some specific manual actuation by the pilot. In this regard, the use of hydraulic pressure is not considered a positive means of down lock.

(Supp. 10, 16 F. R. 3289, Apr. 14, 1951.)

3.357 *Emergency operation.* When other than manual power for the operation of the landing gear is employed, an auxiliary means of extending the landing gear shall be provided.

3.358 *Operation test.* Proper functioning of the landing gear retracting mechanism shall be demonstrated by operation tests.

3.359 *Position indicator and warning device.* When retractable landing wheels are used, means shall be provided for indicating to the pilot when the wheels are secured in the extreme positions. In addition, landplanes shall be provided with an aural or equally effective warning device which shall function continuously after the throttle is closed until the gear is down and locked.

3.359–1 *Wheel position indicators (CAA policies which apply to sec. 3.359).*

(a) The means to be provided for in section 3.359 to indicate to the pilot when the wheels are secured in the extreme positions may consist of lights. For example, a green light for the gear down and locked position is considered satisfactory, provided a placard indicates that this is the down position. "All lights out" is considered satisfactory for intermediate gear positions. However, there should then be another light indicating gear up and locked. "All lights out" is not considered desirable for either extreme gear locked position, since such a system would not "fail safe" if a lamp burned out.

(b) The regulations do not require an aural warning device for amphibian aircraft. A two-light warning system similar to the following would be considered sufficient and satisfactory:

Gear Up	Light #1	Water
Gear Down	Light #2	Land

When light #1 is on, the gear would be in the extreme up position and locked and when light #2 is on, the gear would be in the extreme down position and locked.

(Supp. 10, 16 F. R. 3289, Apr. 14, 1951.)

3.359-2 *Position indicator and warning device (CAA policies which apply to sec. 3.359).* A throttle stop is not considered an acceptable alternative to an aural landing gear warning device.

(Supp. 10, 16 F. R. 3289, Apr. 14, 1951.)

3.359-3 *Landing gear position indicator switches (CAA interpretations which apply to sec. 3.359).* The phrase "means shall be provided for indicating to the pilot" includes a landing gear position indicator as well as the switches necessary to actuate such indicator. The switches must be so located and coupled to the landing gear mechanical system as to preclude the possibility of an erroneous indication of "down and locked" if the landing gear is not in a fully extended position, or "up and locked" if the landing gear is not in the completely retracted position. Location of the switches so that they are operated by the actual landing gear locking latch or device is an acceptable method of compliance with the requirements of this section.

(21 F. R. 3002, May 5, 1956, effective May 25, 1956.)

3.360 *Control.* See section 3.384.

Wheels and Tires

3.361 *Wheels.* Main wheels and nose wheels shall be of an approved type. The maximum static load rating of each main wheel and nose wheel shall not be less than the corresponding static ground reaction under the design maximum weight of the airplane and the critical center of gravity position. The maximum limit load rating of each main wheel and nose wheel shall not be less than the maximum radial limit load determined in accordance with the applicable ground load requirements of this part. (See secs. 3.241 through 3.256.)

3.362 *Tires.* A landing gear wheel may be equipped with any make or type of tire, provided that the tire is a proper fit on the rim of the wheel and provided that the approved tire rating is not exceeded under the following conditions:

(a) Load on each main wheel tire equal to the corresponding static ground reaction under the design maximum weight of the airplane and the critical center of gravity position.

(b) Load on nose wheel tires (to be compared with the dynamic rating established for such tires) equal to the reaction obtained at the nose wheel, assuming the mass of the airplane concentrated at the most critical center of gravity and exerting a force of 1.0*g* downward and 0.31*g* forward, the reactions being distributed to the nose and main wheels by the principle of statics with the drag reaction at the ground applied only at those wheels having brakes. When specially constructed tires are used to support an airplane, the wheels shall be plainly and conspicuously marked to that effect. Such markings shall include the make, size, number of plies, and identification marking of the proper tire.

3.362-1 *Approved tire rating (CAA interpretations which apply to sec. 3.362).* An approved tire rating is a rating assigned by the Tire and Rim Association or by the Administrator.

(Supp. 14, 17 F. R. 9066, Oct. 11, 1952.)

3.362-2 *Tire rating standards (CAA policies which apply to sec. 3.362).* Approved tire ratings or experimental tire ratings assigned by the Tire and Rim Association may be used in determining whether a tire is satisfactory for use on civil aircraft.

(Supp. 14, 17 F. R. 9066, Oct. 11, 1952.)

Brakes

3.363 *Brakes.* Brakes shall be installed which are adequate to prevent the airplane from rolling on a paved runway while applying take-off power to the critical engine, and of sufficient capacity to provide adequate speed control during taxying without the use of excessive pedal or hand forces.

Skis

3.364 *Skis.* Skis shall be of an approved type. The maximum limit load rating of each ski shall not be less than the maximum limit

load determined in accordance with the applicable ground load requirements of this part. (See secs. 3.241 through 3.257.)

3.364–1 *Tail skis* (*CAA interpretations which apply to sec. 3.364*). Type certification of tail skis is not required under the regulations in this subchapter. Such skis should therefore be approved as a part of the airplane.

(Supp. 10, 16 F. R. 3289, Apr. 14, 1951.)

Hulls and Floats

3.371 *Seaplane main floats.* Seaplane main floats shall be of an approved type and shall comply with the provisions of section 3.265. In addition, the following shall apply:

(a) *Buoyancy.* Each seaplane main float shall have a bouyancy of 80 percent in excess of that required to support the maximum weight of the seaplane in fresh water.

[(b) *Compartmentation.* Each seaplane main float shall contain a sufficient number of watertight compartments to provide reasonable assurance that the airplane will stay afloat in the event that any 2 compartments of the main floats are flooded. In any case, a main float shall contain at least 4 watertight compartments. The compartments shall have approximately equal volumes.]

3.372 *Buoyancy (boat seaplanes).* The hulls of boat seaplanes and amphibians shall be divided into watertight compartments in accordance with the following requirements:

(a) In airplanes of 5,000 pounds or more maximum weight, the compartments shall be so arranged that, with any two adjacent compartments flooded, the hull and auxiliary floats (and tires, if used) will retain sufficient buoyancy to support the maximum weight of the airplane in fresh water.

(b) In airplanes of 1,500 to 5,000 pounds maximum weight, the compartments shall be so arranged that, with any one compartment flooded, the hull and auxiliary floats (and tires, if used) will retain sufficient buoyancy to support the maximum weight of the airplane in fresh water.

(c) In airplanes of less than 1,500 pounds maximum weight, watertight subdivision of the hull is not required.

(d) Bulkheads may have watertight doors for the purpose of communication between compartments.

3.373 *Water stability.* Auxiliary floats shall be so arranged that when completely submerged in fresh water, they will provide a righting moment which is at least 1.5 times the upsetting moment caused by the airplane being tilted. A greater degree of stability may be required by the Administrator in the case of large flying boats, depending on the height of the center of gravity above the water level, the area and location of wings and tail surfaces, and other considerations.

Fuselage

Pilot Compartment

3.381 *General.*

(a) The arrangement of the pilot compartment and its appurtenances shall provide a satisfactory degree of safety and assurance that the pilot will be able to perform all his duties and operate the controls in the correct manner without unreasonable concentration and fatigue.

(b) The primary flight control units listed on Figure 3–14, excluding cables and control rods, shall be so located with respect to the propellers that no portion of the pilot or controls lies in the region between the plane of rotation of any inboard propeller and the surface generated by a line passing through the center of the propeller hub and making an angle of 5° forward or aft of the plane of rotation of the propeller.

3.382 *Vision.* The pilot compartment shall be arranged to afford the pilot a sufficiently extensive, clear, and undistorted view for the safe operation of the airplane. During flight in a moderate rain condition, the pilot shall have an adequate view of the flight path in normal flight and landing, and have sufficient protection from the elements so that his vision is not unduly impaired. This may be accomplished by providing an openable window or by a means for maintaining a portion of the windshield in a clear condition without continuous attention by the pilot. The pilot compartment shall be free of glare and reflections which would interfere with the pilot's vision. For airplanes intended for night operation, the demonstration of these qualities shall include night flight tests.

3.382–1 *Openable window or openable portion of the windshield (CAA interpretations which apply to sec. 3.382).*

(a) The third sentence of section 3.382 is interpreted to mean that an openable window, or an openable portion of the windshield is required only when the windshield does not remain, or is not maintained (by means of windshield wipers or other devices) in a clean condition during a moderate rain.

(b) If deflectors or other means are provided, so that the elements do not fully impair the pilot's ability to see when an openable window, or a movable portion of the windshield is open, then the pilot should have an adequate view during the rain condition of the flight path in normal flight and landing with these deflectors or other devices installed (and, if applicable, in any position within the limits of adjustability).

(Supp. 10, 16 F. R. 3289, Apr. 14, 1951.)

3.382–2 *Pilot vision in rain conditions (CAA interpretations which apply to sec. 3.382).* The means for providing vision during flight in rain conditions should permit the pilot to view both the normal flight path and the instrument panel without difficulty or excessive head movement.

3.383 *Windshields, windows, and canopies.*

(a) All internal glass panes shall be of a nonsplintering safety type.

(b) The design of windshields, windows, and canopies on pressurized airplanes shall be based on factors peculiar to high altitude operation. (See also sec. 3.394.)

NOTE: **Factors peculiar to high altitude operation normally include the effect of continuous and cyclic pressurization loadings, the inherent characteristics of the material used, the effects of temperatures and temperature gradients, etc.**

(c) On pressurized airplanes, an enclosure canopy including a representative portion of the installation shall be subjected to special tests to account for the combined effects of continuous and cyclic pressurization loadings and flight loads.

3.383–1 *Plexiglas windshields and windows (CAA policies which apply to sec. 3.383).* A plastic material such as plexiglas is considered to be a nonsplintering material and can be used in windshields and windows.

(Supp. 10, 16 F. R. 3290, Apr. 14, 1951.)

3.384 *Cockpit controls.*

(a) All cockpit controls shall be so located and, except for those the function of which is obvious, identified as to provide convenience in operation including provisions to prevent the possibility of confusion and consequent inadvertent operation. (See Fig. 3–14 for required sense of motion of cockpit controls.) The controls shall be so located and arranged that when seated it will be readily possible for the pilot to obtain full and unrestricted movement of each control without interference from either his clothing or the cockpit structure.

(b) Identical power-plant controls for the several engines in the case of multi-engine airplanes shall be so located as to prevent any misleading impression as to the engines to which they relate.

Controls	***Movement and actuation***
Primary:	
Aileron	**Right (clockwise) for right wing down.**
Elevator	**Rearward for nose up.**
Rudder	**Right pedal forward for nose right.**
Power plant:	
Throttle	**Forward to open.**

Figure 3–14.—Cockpit controls.

3.385 *Instruments and markings.* See section 3.661 relative to instrument arrangement. The operational markings, instructions, and placards required for the instruments and controls are specified in sections 3.756 to 3.765.

Emergency Provisions

3.386 *Protection.* The fuselage shall be designed to give reasonable assurance that each occupant, if he makes proper use of belts or harness for which provisions are made in the design, will not suffer serious injury during minor crash conditions as a result of contact of any vulnerable part of his body with any penetrating or relatively solid object, although it is

accepted that parts of the airplane may be damaged.

(a) The ultimate accelerations to which occupants are assumed to be subjected shall be as follows:

	N, U	A
Upward	3.0*g*	4.5*g*
Forward	9.0*g*	9.0*g*
Sideward	1.5*g*	1.5*g*

(b) For airplanes having retractable landing gear, the fuselage in combination with other portions of the structure shall be designed to afford protection of the occupants in a wheels-up landing with moderate descent velocity.

(c) If the characteristics of an airplane are such as to make a turnover reasonably probable, the fuselage of such an airplane in combination with other portions of the structure shall be designed to afford protection of the occupants in a complete turnover.

NOTE: In section 3.386 (b) and (c), a vertical ultimate acceleration of 3*g* and a friction coefficient of 0.5 at the ground may be assumed.

[(d) The inertia forces specified for N, U, and A category airplanes in paragraph (a) of this section shall be applied to all items of mass which would be apt to injure the passengers or crew if such items became loose in the event of a minor crash landing, and the supporting structure shall be designed to restrain these items.]

3.386–1 *Crash protection (CAA interpretations which apply to sec. 3.386).*

(a) Cockpit arrangements and collapse of cabin structure have been found to cause excessive injuries in crashes. Close study of crash results shows that the human body, when properly supported, can tolerate crash forces which exceed those necessary to demolish contemporary aircraft structure.

(b) The following points are of general significance:

(1) Many survivable accidents are "fatal" because of insufficient design consideration when mocking up the cabin and its installation.

(2) The torso is rarely exposed to dangerous injury when the safety belts hold and control wheels provide reasonable support for the chest.

(3) Fractures of the extremities occur in severe crashes but are not normally regarded as dangerous injuries.

(4) Head injuries are the principal cause of crash fatalities. Increased protection for the head can be provided by elimination, shielding, or redesigning of elements of the cabin which permit solid head blows in a crack-up, such as turnovers during a bad landing.

(c) In view of the fact that injuries and fatalities in many moderate and severe accidents are purely mechanical results of poor cockpit design, the following guide rules for design are suggested:

(1) Typical injurious objects, from the standpoint of crash injury, are listed as follows:

(i) Those which present a hard surface and are so attached or have sufficient mass to produce a severe impact when struck by the head or other vulnerable part of the body as it swings forward under the specified inertia forces.

(ii) Those which present corners, knobs, or similar projections which are likely to penetrate a vulnerable part of the body, even when the impact forces are not as high as in paragraph (a) of this section.

(2) A flat or curved sheet metal panel which will dent upon impact by the head is not considered dangerous, whereas a magnetic compass case having appreciable mass and a rigid mounting might cause fatal head injuries.

(3) Heavy transverse braces or other structures immediately behind a light instrument panel have changed many accident reports from "Instrument panel depressed six inches by pilot's head" to "Fatal head injury; depressed fracture of the skull." Pilot's chances can be greatly improved by spacing solid braces several inches behind the ductile skirt of an instrument panel.

(4) The solid tubing used as a backrest of the front seats of tandem aircraft is a set-up for excessive head injury. The suggestion has been made that backs of forward seats be allowed to pivot forward so that the head of the occupant of the rearward seat would not

contact the solid members when the body pivots about the belt.

(5) Panels should be smooth, with top edge curved in a substantial radius.

(6) Apertures for instruments should preferably have bevelled instead of sharp edges.

(7) In personal aircraft, every consideration should be given to holding the body by adequate safety belt installations, and by the support which can be provided in control wheels and instrument panels. The present "1000 pound" safety belts have failed in a high percentage of accidents without causing internal injuries or bruising of the hips. In failing, they have exposed the pilot to excessive injuries.

(8) Control wheels should be designed to provide broad areas of support for the chest. Wheels which break under heavy loads from the hands or deform to permit contact between the chest and a small hub, localize force and set up chances of unnecessary chest injury.

(Supp. 10, 16 F. R. 3290, Apr. 14, 1951.)

3.387 *Exits.*

(a) Closed cabins on airplanes carrying more than 5 persons shall be provided with emergency exits consisting of movable windows or panels or of additional external doors which provide a clear and unobstructed opening, the minimum dimensions of which shall be such that a 19-by-26-inch ellipse may be completely inscribed therein. The exits shall be readily accessible, shall not require exceptional agility of a person using them, and shall be distributed

so as to facilitate egress without crowding in all probable attitudes resulting from a crash. The method of opening shall be simple and obvious, and the exits shall be so arranged and marked as to be readily located and operated even in darkness. Reasonable provisions shall be made against the jamming of exits as a result of fuselage deformation. The proper functioning of exits shall be demonstrated by tests.

(b) The number of emergency exits required is as follows:

(1) Airplanes with a total seating capacity of more than 5 persons, but not in excess of 15, shall be provided with at least one emergency exit or one suitable door in addition to the main door specified in section 3.389. This emergency exit, or second door, shall be on the opposite side of the cabin from the main door.

(2) Airplanes with a seating capacity of more than 15 persons shall be provided with emergency exits or doors in addition to those required in paragraph (b) (1) of this section. There shall be one such additional exit or door located either in the top or side of the cabin for every additional 7 persons or fraction thereof above 15, except that not more than four exits, including doors, will be required if the arrangement and dimensions are suitable for quick evacuation of all occupants.

(c) If the pilot compartment is separated from the cabin by a door which is likely to block escape in the event of a minor crash, it shall have its own exit, but such exit shall not be considered as an emergency exit for the passengers.

[(d) Category A airplanes shall be provided with exits which will permit all occupants to bail out quickly with parachutes at any speed between V_{s_0} and V_d.]

3.388 ***Fire precautions.***

(a) *Cabin interiors.* Only materials which are flash-resistant shall be used. In compartments where smoking is to be permitted, the wall and ceiling linings, the covering of all upholstering, floors, and furnishings shall be flame-resistant. Such compartments shall be equipped with an adequate number of self-contained ash trays. All other compartments shall be placarded against smoking.

(b) *Combustion heaters.* If combustion heaters are installed, they shall be of an approved type. The installation shall comply with applicable parts of the powerplant installation requirements covering fire hazards and precautions. All applicable requirements concerning fuel tanks, lines, and exhaust systems shall be considered. In addition to the components provided for normal continuous control of air temperature, air flow, and fuel flow, means independent of such components shall be provided for each heater to automatically shut off and hold off the ignition and fuel supply to the heater at a point remote from the heater when the heat exchanger temperature or ventilating air temperature exceeds safe limits or when either the combustion air flow or the ventilating air flow becomes inadequate for safe operation.

3.388-1 *Heater isolation (CAA policies which apply to sec. 3.388 (b)).*

(a) Under sections 3.388 (b) and 3.623, heaters should be isolated from the remainder of the airplane by means of a fireproof shield. However, this need not necessarily mean a complete shield around the entire heater unit (although this would be satisfactory) since in many heater designs, a fireproof air jacket largely surrounds the flame chamber. Thus, the heater design itself practically provides a steel shield between the combustion unit and the remainder of the airplane. In such cases, it should suffice to provide isolation for the fuel system components mounted on the heater and for the heater exhaust and combustion chamber drains.

(b) The following schematic sketch shows an example of an installation which should be satisfactory.

The shut-off valve shown in the sketch should be provided if there are fuel system components within the ventilating air shroud which may be subject to leakage or failure. In such cases, that portion of the ventilating air duct up to the valve, as well as the valve itself, should be of fire-resistant construction and the valve should provide as flame-tight a seal as possible. If the fuel system is so arranged that there are no fittings or connections within the ventilating air shroud, the downstream air shut-off valve and fire-resistant duct between the heater and the valve may be dispensed with.

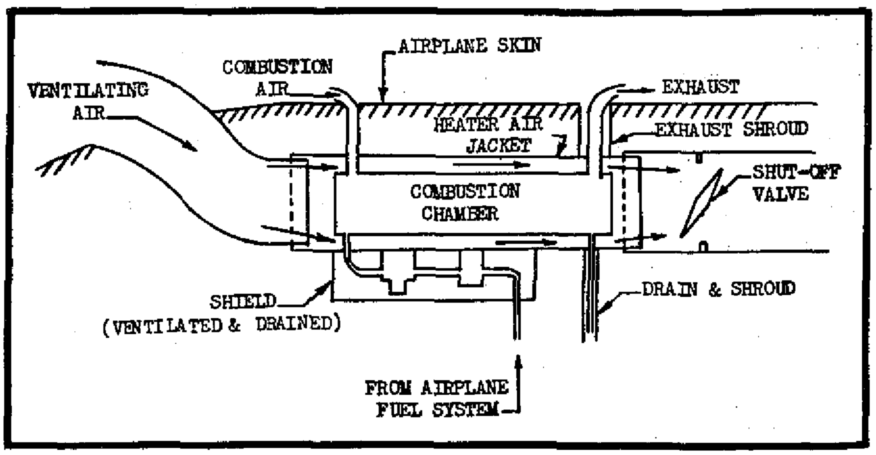


Figure 2.—Heater isolation.

(c) As regards shrouds for the combustion chamber drain lines, the necessity for these will generally depend upon the location of the drain in the heater. If the drain outlet from the combustion chamber is so located that products of combustion can issue through the drain line, it will no doubt become hot and require isolation. However, drains are sometimes connected in such a manner that they do not carry exhaust gases and remain relatively cool. In such cases, shrouds are not necessary.

(Supp. 10, 16 F. R. 3290, Apr. 14, 1951.)

3.388-2 *Fire-detector and extinguisher equipment (CAA policies which apply to sec. 3.388 (b)).*

(a) For nontransport category airplanes equipment of this type is not required. If such equipment is installed and it is shown to provide equivalent safety to the use of fireproof isolation, it may be considered a suitable alternative for fireproof isolation provisions discussed in section 3.388-1 (b). In such cases, the detection and extinguishing provisions should comply with the requirements for transport category airplanes; that is, detectors and extinguishers should be provided wherever potential sources of fuel leakage and sources of ignition are in close proximity.

(b) In the sketch in section 3.388-1 (b) the space within the shield would require such protection. In addition, detectors and extinguisher nozzles should be installed in the ventilating air passages of the heater if this chamber contains fuel system fittings or connections that may be subject to leakage.

(c) Hand fire extinguishers should be considered equivalent to a fixed fire extinguisher installation only when the heater is located in such a manner that it is readily accessible to the crew and when all fire zones in the installation can easily be reached with a hand extinguisher.

(d) All extinguishers may also be dispensed with when the heater is so shielded and located that a fire could be permitted to burn itself out without danger of damage to any important structural members or otherwise endangering the safety of the airplane.

(e) Detectors may be dispensed with as an alternative to fireproof isolation, only when the heater is so located that the occurrence of fire would immediately be noted by the crew.

3.388-3 *Heater fuel system (CAA policies which apply to sec. 3.388 (b)).*

(a) The heater fuel system should comply with airworthiness standards for the engine fuel system as regards fuel lines, fittings and accessories.

(b) Valves should be provided for shutting off in flight the flow of fuel at its source, unless equivalent provisions in the form of a separate heater fuel pump are available.

(c) All pressure lines should comply with the provisions of section 3.432 regarding pressure cross feed arrangements.

(Supp. 10, 16 F. R. 3291, Apr. 14, 1951.)

3.388-4 *Combustion heaters (CAA rules which apply to sec. 3.388 (b)).* The minimum safety requirements for combustion heaters which are intended for use in civil aircraft have been established by the Administrator in Technical Standard Order No. TSO-C20, effective June 15, 1949, "Combustion Heaters" (Sec. 514.30 of this title).

(Supp. 10, 16 F. R. 3291, Apr. 14, 1951.)

Personnel and Cargo Accommodations

3.389 *Doors.* Closed cabins on all airplanes carrying passengers shall be provided with at least one adequate and easily accessible external door. No passenger door shall be so located with respect to the propeller discs as to endanger persons using the door.

3.390 *Seats and berths.* All seats and berths shall be of an approved type. They and their supporting structures shall be designed for an occupant weighing at least 170 pounds (190 pounds with parachute for seats intended for the acrobatic and utility categories) and for the maximum load factors corresponding with all specified flight and ground load conditions including the emergency landing conditions prescribed in section 3.386. The provisions of paragraphs (a) through (d) of this section shall also apply:

(a) Pilot seats shall be designed for the reactions resulting from the application of pilot forces to the primary flight controls as prescribed in section 3.231.

(b) All seats in the U and A categories shall be designed to accommodate passengers wearing parachutes, unless placarded in accordance with section 3.74 (b).

(c) Berths shall be so designed that the forward portion is provided with a padded end board, a canvas diaphragm, or other equivalent means, capable of withstanding the static load reaction of the occupant when subjected to the forward accelerations prescribed in section 3.386. Berths shall be provided with an approved safety belt and shall be free from corners or protuberances likely to cause serious injury to a person occupying the berth during emergency conditions. Berth safety belt attachments shall withstand the critical loads resulting from all relevant flight and ground load conditions and from the emergency landing conditions of section 3.386 with the exception of the forward load.

(d) In determining the strength of the attachment of the seat and berth to the structure, the accelerations prescribed in section 3.386 shall be multiplied by a factor of 1.33.

3.390-1 *Approved seats and berths (CAA interpretations which apply to sec. 3.390).* An approved seat or berth is one which complies with the pertinent requirements in the regulations in this subchapter as implemented by TSO-C25 "Aircraft Seats and Berths" (sec. 514.35 of this title).

3.390-2 *Proof of strength for seats and berths and their installations (CAA policies which apply to sec. 3.390).*

(a) Proof of compliance with strength and deformation requirements for seats and berths, approved as a part of the type design, and for all seat and berth installations, may be shown by one of the following methods:

(1) Structural analysis alone when the structure conforms with conventional types for which existing methods of analysis are known to be reliable.

(2) A combination of structural analysis and static load tests to limit loads.

(3) Static load tests alone when such tests are carried to ultimate loads.

(Supp. 14, 17 F. R. 9066, Oct. 11, 1952.)

3.390-3 *Application of loads (CAA policies which apply to sec. 3.390).* The actual forces acting on seats, berths, and supporting structure in the various flight, ground and emergency landing conditions will consist of many possible combinations of forward, sideward, downward, upward, and aft loads. However, in order to simplify the structural analysis and testing of these structures, it will be permissible to assume that the critical load in each of these directions, as determined from the prescribed flight, ground, and emergency landing conditions, acts separately. If the applicant desires, selected combinations of loads may be used, provided the required strength in all specified directions is substantiated (TSO C-25, Aircraft Seats and Berths, section 514.35 of this title, outlines acceptable methods for testing seats and berths).

(Supp. 17, 18 F. R. 5563, Sept. 17, 1953.)

3.392 *Cargo compartments.* Each cargo compartment shall be designed for the placarded maximum weight of contents and critical load distributions at the appropriate maximum

load factors corresponding to all specified flight and ground load conditions. Suitable provisions shall be made to prevent the contents of cargo compartments from becoming a hazard by shifting. Such provisions shall be adequate to protect the passengers from injury by the contents of any cargo compartment when the ultimate forward acting accelerating force is 4.5*g*.

3.392–1 *Load factors for design of cargo compartments located in the fuselage* (*CAA interpretations which apply to sec. 3.392*).

(a) It would seem on examination of sections 3.392 and 3.386 that there is a conflict between the load factors required for the design of cargo compartments which are located in the fuselage. The following explanation should clarify this possible misconception:

(1) Section 3.392 was specially promulgated to overcome objections to the excessively heavy cargo compartment structure that would be required to meet the c ash conditions of section 3.386. In past cases of crashes, injuries to passengers caused by shifting cargo or baggage have not been prevalent despite the fact that in many cases the lower design factors of Bulletin 7a and Part 4a of this subchapter were in effect. Because of this, section 3.392 was incorporated in the requirements, to apply specifically to cargo compartments. It should therefore not be necessary to consider the strength requirements of section 3.386 in their design.

(Supp. 10, 16 F. R. 3291, Apr. 14, 1951.)

3.393 *Ventilation.* All passenger and crew compartments shall be suitably ventilated. Carbon monoxide concentration shall not exceed 1 part in 20,000 parts of air.

[3.394 *Pressurized cabins; general.* The design of pressurized cabins shall comply with the requirements of sections 3.395 and 3.396. (See also secs. 3.197, 3.270, and 3.383.)

[3.395 *Pressure control.* Pressurized cabins shall be provided with at least the following valves, controls, and indicators for controlling cabin pressure:

[(a) Two pressure relief valves, at least one of which is the normal regulating valve, shall be installed to limit automatically the positive pressure differential to a predetermined value at the maximum rate of flow delivered by the pressure source. The combined capacity of the relief valves shall be such that the failure of any one valve would not cause an appreciable rise in the pressure differential. The pressure differential shall be considered positive when the internal pressure is greater than the external.

[(b) Two reverse pressure differential relief valves (or equivalent) shall be installed to prevent automatically a negative pressure differential which would damage the structure, except that one such valve shall be considered sufficient if it is of a design which reasonably precludes its malfunctioning.

[(c) Means shall be provided by which the pressure differential can be rapidly equalized.

[(d) An automatic or manual regulator for controlling the intake and/or exhaust air flow shall be installed so that the required internal pressures and air flow rates can be maintained.

[(e) Instruments shall be provided for the pilot to show the pressure differential, the absolute pressure in the cabin, and the rate of change of the absolute pressure.

[(f) Warning indication shall be provided for the pilot to indicate when the safe or preset limits on pressure differential and on absolute cabin pressure are exceeded.

[(g) If the structure is not designed for pressure differentials up to the maximum relief valve setting in combination with landing loads (see sec. 3.197 (b)), a warning placard shall be provided for the pilot.

[(h) If continued rotation of an engine-driven cabin compressor or if continued flow of any compressor bleed air will constitute a hazard in case malfunction occurs, means shall be provided to stop rotation of the compressor or to divert air flow from the cabin.

[3.396 *Tests.*

[(a) *Strength test.* The complete pressurized cabin, including doors, windows, canopy, and all valves shall be tested as a pressure vessel for the pressure differential specified in section 3.197 (c).

[(b) *Functional tests.* The following functional tests shall be performed:

[(1) To simulate the condition of regulator valves closed, the functioning and the capacity shall be tested of the positive and negative pres-

sure differential valves and of the emergency release valve.

(2) All parts of the pressurization system shall be tested to show proper functioning under all possible conditions of pressure, temperature, and moisture up to the maximum altitude selected for certification.

(3) Flight tests shall be conducted to demonstrate the performance of the pressure supply, pressure and flow regulators, indicators, and warning signals in steady and stepped climbs and descents at rates corresponding with the maximum attainable without exceeding the operating limitations of the airplane up to the maximum altitude selected for certification.

(4) All doors and emergency exits shall be tested to ascertain that they operate properly after being subjected to the flight tests prescribed in subparagraph (3) of this paragraph.

Miscellaneous

3.401 *Leveling marks.* Leveling marks shall be provided for leveling the airplane on the ground.

Subpart E—Power-Plant Installations; Reciprocating Engines

General

3.411 *Components.*

(a) The power-plant installation shall be considered to include all components of the airplane which are necessary for its propulsion. It shall also be considered to include all components which affect the control of the major propulsive units of which affect their continued safety of operation.

(b) All components of the power-plant installation shall be constructed, arranged, and installed in a manner which will assure the continued safe operation of the airplane and power plant. Accessibility shall be provided to permit such inspection and maintenance as is necessary to assure continued airworthiness.

[3.411-1 *Powerplant installation components (CAA interpretations which apply to sec. 3.411).* The term "all components" includes engines and propellers and their parts, appurtenances, and accessories which are furnished by the engine or propeller manufacturer and all other components of the powerplant installation which are furnished by the airplane manufacturer. For example: fuel pumps, lines, valves, and other components of the fuel system which are integral parts of the type certificated engine are also components of the airplane powerplant installation.]

(23 F. R. 9018, Nov. 20, 1958, effective Dec. 22, 1958.)

Engines and Propellers

3.415 *Engines.* Engines installed in certificated airplanes shall be of a type which has been certificated in accordance with the provisions of Part 13 of this subchapter.

3.416 *Propellers.*

(a) Propellers installed in certificated airplanes shall be of a type which has been certificated in accordance with the provisions of Part 14 of this subchapter.

(b) The maximum engine power and propeller shaft rotational speed permissible for use in the particular airplane involved shall not exceed the corresponding limits for which the propeller has been certificated.

3.417 *Propeller vibration.* In the case of propellers with metal blades or other highly stressed metal components, the magnitude of the critical vibration stresses under all normal conditions of operation shall be determined by actual measurements or by comparison with similar installations for which such measurements have been made. The vibration stresses thus determined shall not exceed values which have been demonstrated to be safe for continuous operation. Vibration tests may be waived and the propeller installation accepted on the basis of service experience, engine or ground tests which show adequate margins of safety, or other considerations which satisfactorily substantiate its safety in this respect. In addition to metal propellers, the Administrator may require that similar substantiation of the vibration characteristics be accomplished for other types of propellers, with the exception of conventional fixed-pitch wood propellers.

3.418 *Propeller pitch and speed limitations.* The propeller pitch and speed shall be limited to values which will assure safe operation under all normal conditions of operation and will assure compliance with the per-

formance requirements specified in sections 3.81–3.86.

3.419 ***Speed limitations for fixed-pitch propellers, ground adjustable pitch propellers, and automatically varying pitch propellers which cannot be controlled in flight.***

(a) During take-off and initial climb at best rate-of-climb speed, the propeller, in the case of fixed-pitch or ground adjustable types, shall restrain the engine to a speed not exceeding its maximum permissible take-off speed and, in the case of automatic variable-pitch types, shall limit the maximum governed engine revolutions per minute to a speed not exceeding the maximum permissible take-off speed. In demonstrating compliance with this provision the engine shall be operated at full throttle or the throttle setting corresponding to the maximum permissible take-off manifold pressure.

(b) During a closed throttle glide at the placard, "never-exceed speed" (see sec. 3.739), the propeller shall not cause the engine to rotate at a speed in excess of 110 percent of its maximum allowable continuous speed.

3.419–1 *Propeller pitch and speed limitations (CAA interpretations which apply to sec. 3.419).*

(a) The low pitch setting should comply with section 3.419 (a) which states that the propeller shall not exceed the rated engine takeoff r. p. m. with takeoff power (full throttle unless limited by manifold pressure) during takeoff and initial climb at best rate of climb speed. It is not permissible to use a lower pitch setting than that specified above in order to obtain takeoff r. p. m. at the best angle of climb speed for the purpose of showing compliance with section 3.85 (c), Balked Landing Conditions. An exception to the above may be granted in the specific case covered by section 3.85–5 when satisfactory engine cooling can be demonstrated at the best angle of climb speed in the balked landing configuration (sec. 3.85 (c)). However, in cases where the interpretation of section 3.85 does not govern, it will be necessary to conduct the balked landing climb with whatever r. p. m. is possible without exceeding the engine takeoff limitations with the low pitch setting determined in accordance with section 3.419 (a).

(b) In cases where the airplane is to be operated using either the water injection or dry takeoff power ratings of the engines, the low pitch stop setting shall be determined on the basis of whichever rating will result in the lower pitch. This will generally be the "dry" rating. In instances where the airplane is intended to be operated only at the water injection takeoff power ratings of the engines, the low pitch stop for the propellers should be determined on that basis. These settings are to be determined in the usual manner with the airplane static unless there are unconventional features in the propeller installation requiring this determination by some other means.

(c) In cases where dual engines drive a single propeller through free wheeling clutches, the setting of the low pitch stop should be such that the propeller will not overspeed when takeoff power is applied to one engine at an airplane speed of V_2.

(Supp. 10, 16 F. R. 3291, Apr. 14, 1951.)

3.420 ***Speed and pitch limitations for controllable pitch propellers without constant speed controls.*** **The stops or other means incorporated in the propeller mechanism to restrict the pitch range shall limit (a) the lowest possible blade pitch to a value which will assure compliance with the provisions of section 3.419 (a), and (b) the highest possible blade pitch to a value not lower than the flattest blade pitch with which compliance with the provisions of section 3.419 (b) can be demonstrated.**

3.421 ***Variable pitch propellers with constant speed controls.***

(a) Suitable means shall be provided at the governor to limit the speed of the propeller. Such means shall limit the maximum governed engine speed to a value not exceeding its maximum permissible takeoff revolutions per minute.

(b) The low pitch blade stop, or other means incorporated in the propeller mechanism to restrict the pitch range, shall limit the speed of the engine to a value not exceeding 103 percent of the maximum permissible takeoff revolutions per minute under the following conditions:

(1) Propeller blades set in the lowest possible pitch and the governor inoperative.

(2) Engine operating takeoff manifold pressure with the airplane stationary and with no wind.

3.422 *Propeller clearance.* With the airplane loaded to the maximum weight and most adverse center of gravity position and the propeller in the most adverse pitch position, propeller clearances shall not be less than the following, unless smaller clearances are properly substantiated for the particular design involved:

(a) *Ground clearance.*

(1) Seven inches (for airplanes equipped with nose wheel type landing gears) or 9 inches (for airplanes equipped with tail wheel type landing gears) with the landing gear statically deflected and the airplane in the level, normal takeoff, or taxiing attitude, whichever is most critical.

(2) In addition to subparagraph (1) of this paragraph, there shall be positive clearance between the propeller and the ground when, with the airplane in the level takeoff attitude, the critical tire is completely deflated and the corresponding landing gear strut is completely bottomed.

(b) *Water clearance.* A minimum clearance of 18 inches shall be provided unless compliance with section 3.147 can be demonstrated with lesser clearance.

(c) *Structural clearance.*

(1) One inch radial clearance between the bladetips and the airplane structure, or whatever additional radial clearance is necessary to preclude harmful vibration of the propeller or airplane.

(2) One-half inch longitudinal clearance between the propeller blades or cuffs and stationary portions of the airplane. Adequate positive clearance shall be provided between other rotating portions of the propeller or spinner and stationary portions of the airplane.

3.422-1 *Propeller clearance on tricycle gear airplanes (CAA interpretations which apply to sec. 3.422 (a) (1)).* In determining minimum propeller clearance for aircraft equipped with tricycle gear, dynamic effects need not be considered.

(Supp. 10, 16 F. R. 3291, Apr. 14, 1951.)

3.422-2 *Propeller clearance on aircraft with leaf spring type shock struts (CAA interpretations which apply to sec. 3.422 (a) (2)).* Section 3.422 (a) (2) applies only to conventional landing gear struts employing fluid and for mechanical means for absorbing landing shocks. For aircraft employing struts of the leaf spring type, a deflection corresponding to 1.5g should be used to determine whether positive clearance exists.

(Supp. 10, 16 F. R. 3291, Apr. 14, 1951.)

Fuel System

3.429 *General.* The fuel system shall be constructed and arranged in a manner to assure the provision of fuel to each engine at a flow rate and pressure adequate for proper engine functioning under all normal conditions of operation, including all maneuvers and acrobatics for which the airplane is intended.

Arrangement

3.430 *Fuel system arrangement.* Fuel systems shall be so arranged as to permit any one fuel pump to draw fuel from only one tank at a time. Gravity feed systems shall not supply fuel to any one engine from more than one tank at a time unless the tank air spaces are interconnected in such a manner as to assure that all interconnected tanks will feed equally. (See also sec. 3.439.)

3.431 *Multiengine fuel system arrangement.* The fuel systems of multiengine airplanes shall be arranged to permit operation in at least one configuration in such a manner that the failure of any one component will not result in the loss of power of more than one engine and will not require immediate action by the pilot to prevent the loss of power of more than one engine. Unless other provisions are made to comply with this requirement, the fuel system shall be arranged to permit supplying fuel to each engine through a system entirely independent of any portion of the system supplying fuel to the other engines.

NOTE: It is not necessarily intended that fuel tanks proper be separate for each engine if a common tank is provided with separate outlets and the remainder of the fuel system is independent.

3.431-1 *Multiengine single tank fuel system (CAA policies which apply to sec. 3.431).*

When a common fuel tank is provided in multiengine aircraft, unless other acceptable provisions are made, a shutoff valve should be installed at the tank outlet of each individual fuel system. This valve may also serve as the fire wall shutoff valve required by section 3.551, provided the line between the valve and the engine compartment does not contain a hazardous amount of fuel (more than 1 quart) which can drain into the engine compartment.

(22 F. R. 4877, July 11, 1957, effective Aug. 1, 1957.)

3.432 *Pressure cross feed arrangements.* Pressure cross feed lines shall not pass through portions of the airplane devoted to carrying personnel or cargo, unless means are provided to permit the flight personnel to shut off the supply of fuel to these lines, or unless any joints, fittings, or other possible sources of leakage installed in such lines are enclosed in a fuel- and fume-proof enclosure which is ventilated and drained to the exterior of the airplane. Bare tubing need not be enclosed but shall be protected where necessary against possible inadvertent damage.

Operation

3.433 *Fuel flow rate.* The ability of the fuel system to provide the required fuel flow rate and pressure shall be demonstrated when the airplane is in the attitude which represents the most adverse condition from the standpoint of fuel feed and quantity of unusable fuel in the tank. During this test fuel shall be delivered *to the engine at the applicable flow rate* (see secs. 3.434–3.436) and at a pressure not less than the minimum required for proper carburetor operation. A suitable mock-up of the

system, in which the most adverse conditions are simulated, may be used for this purpose. The quantity of fuel in the tank being tested shall not exceed the amount established as the unusable fuel supply for that tank as determined by demonstration of compliance with the provisions of section 3.437 (see also secs. 3.440 and 3.672), plus whatever minimum quantity of fuel it may be necessary to add for the purpose of conducting the flow test. If a fuel flowmeter is provided, the meter shall be blocked during the flow test and the fuel shall flow through the meter bypass.

3.434 *Fuel flow rate for gravity systems.* The fuel flow rate for gravity systems (main and reserve supply) shall be 150 percent of the actual take-off fuel consumption of the engine.

3.435 *Fuel flow rate for pump systems.* The fuel flow rate for pump systems (main and reserve supply) shall be 0.9 pound per hour for each take-off horsepower or 125 percent of the actual take-off fuel consumption of the engine, whichever is greater. This flow rate shall be applicable to both the primary engine-driven pump and the emergency pumps and shall be available when the pump is running at the speed at which it would normally be operating during take-off. In the case of hand-operated pumps, this speed shall be considered to be not more than 60 complete cycles (120 single strokes) per minute.

3.436 *Fuel flow rate for auxiliary fuel systems and fuel transfer systems.* The provisions of section 3.434 or section 3.435, whichever is applicable, shall also apply to auxiliary and transfer systems with the exception that the required fuel flow rate shall be established upon the basis of maximum continuous power and speed instead of take-off power and speed. A lesser flow rate shall be acceptable, however, in the case of a small auxiliary tank feeding into a large main tank, provided a suitable placard is installed to require that the auxiliary tank must only be opened to the main tank when a predetermined satisfactory amount of fuel still remains in the main tank.

3.437 *Determination of unusable fuel supply and fuel system operation on low fuel.*

(a) The unusable fuel supply for each tank shall be established as not less than the quantity at which the first evidence of malfunctioning occurs under the conditions specified in this section. (See also sec. 3.440.) In the case of airplanes equipped with more than one fuel tank, any tank which is not required to feed the engine in all of the conditions specified in this section need be investigated only for those flight conditions in which it shall be used and the unusable fuel supply for the particular tank in question shall then be based on the most critical of those conditions which are found to be applicable. In all such cases, information regarding the conditions under which the full amount of usable fuel in the tank can safely be used shall be made available to the operating personnel by means of a suitable placard or instructions in the Airplane Flight Manual.

(b) Upon presentation of the airplane for test, the applicant shall stipulate the quantity of fuel with which he chooses to demonstrate compliance with this provision and shall also indicate which of the following conditions is most critical from the standpoint of establishing the unusable fuel supply. He shall also indicate the order in which the other conditions are critical from this standpoint:

(1) Level flight at maximum continuous power or the power required for level flight at V_c, whichever is less.

(2) Climb at maximum continuous power at the calculated best angle of climb at minimum weight.

(3) Rapid application of power and subsequent transition to best rate of climb following a power-off glide at 1.3 V_{s_0}.

(4) Sideslips and skids in level flight, climb, and glide under the conditions specified in subparagraphs (1), (2), and (3) of this paragraph, of the greatest severity likely to be encountered in normal service or in turbulent air.

(c) In the case of utility category airplanes, there shall be no evidence of malfunctioning during the execution of all approved maneuvers included in the Airplane Flight Manual. During this test the quantity of fuel in each tank shall not exceed the quantity established as the unusable fuel supply, in accordance with paragraph (b) of this section, plus 0.03 gallon for each maximum continuous horsepower for which the airplane is certificated.

(d) In the case of acrobatic category airplanes, there shall be no evidence of malfunctioning during the execution of all approved maneuvers included in the Airplane Flight Manual. During this test the quantity of fuel in each tank shall not exceed that specified in paragraph (c) of this section.

(e) If an engine can be supplied with fuel from more than one tank, it shall be possible to regain the full power and fuel pressure of that engine in not more than 10 seconds (for single-engine airplanes) or 20 seconds (for multiengine airplanes) after switching to any full tank after engine malfunctioning becomes apparent due to the depletion of the fuel supply in any tank from which the engine can be fed. Compliance with this provision shall be demonstrated in level flight.

(f) There shall be no evidence of malfunctioning during take-off and climb for 1 minute at the calculated attitude of best angle of climb at take-off power and minimum weight. At the beginning of this test the quantity of fuel in each tank shall not exceed that specified in paragraph (c) of this section.

3.438 *Fuel system hot weather operation.* Airplanes with suction lift fuel systems or other fuel system features conducive to vapor formation shall be demonstrated to be free from vapor lock when using fuel at a temperature of 110° F. under critical operating conditions.

3.439 *Flow between interconnected tanks.* In the case of gravity feed systems with tanks whose outlets are interconnected, it shall not be possible for fuel to flow between tanks in quantities sufficient to cause an overflow of fuel from the tank vent when the airplane is operated as specified in section 3.437 (a) and the tanks are full.

3.440 *General.* Fuel tanks shall be capable of withstanding without failure any vibration, inertia, and fluid and structural loads to which they may be subjected in operation. Flexible fuel tank liners shall be of an acceptable type. Integral type fuel tanks shall be provided with adequate facilities for the inspection and repair of the tank interior. The total usable capacity of the fuel tanks shall be sufficient for not less than one-half hour operation at rated maximum continuous power (see sec. 3.74 (d)). The unusable capacity shall be considered to be the minimum quantity of fuel which will permit compliance with the provisions of section 3.437. The fuel quantity indicator shall be adjusted to account for the unusable fuel supply as specified in section 3.672. If the unusable fuel supply in any tank exceeds 5 percent of the tank capacity or 1 gallon, whichever is greater, a placard and a suitable notation in the Airplane Flight Manual shall be provided to indicate to the flight personnel that the fuel remaining in the tank when the quantity indicator reads zero cannot be used safely in flight. The weight of the unusable fuel supply shall be included in the empty weight of the airplane.

3.441 *Fuel tank tests.*

(a) Fuel tanks shall be capable of withstanding the following pressure tests without failure or leakage. These pressures may be applied in a manner simulating the actual pressure distribution in service:

(1) Conventional metal tanks and nonmetallic tanks whose walls are not supported by the airplane structure: A pressure of 3.5 p. s. i. or the pressure developed during the maximum ultimate acceleration of the airplane with a full tank, whichever is greater.

(2) Integral tanks: The pressure developed during the maximum limit acceleration of the airplane with a full tank, simultaneously with the application of the critical limit structural loads.

(3) Nonmetallic tanks the walls of which are supported by the airplane structure. Tanks constructed of an acceptable basic tank material and type of construction and with actual or simulated support conditions shall be subjected to a pressure of 2 p. s. i. for the first tank of a specific design. The supporting structure shall be designed for the critical loads occurring in the flight or landing strength conditions combined with the fuel pressure loads resulting from the corresponding accelerations.

(b) (1) Tanks with large unsupported or unstiffened flat areas shall be capable of withstanding the following tests without leakage or failure. The complete tank assembly, together with its supports, shall be subjected to a vibration test when mounted in a manner simulating the actual installation. The tank assembly shall be vibrated for 25 hours at a total amplitude of not less than 1/32 of an inch while filled 2/3 full of water. The frequency of vibration shall be 90 percent of the maximum continuous rated speed of the engine unless some other frequency within the normal operating range of speeds of the engine is more critical, in which case the latter speed shall be employed and the time of test shall be adjusted to accomplish the same number of vibration cycles.

(2) In conjunction with the vibration test, the tank assembly shall be rocked through an angle of 15° on either side of the horizontal (30° total) about an axis parallel to the axis of the fuselage. The assembly shall be rocked at the rate of 16 to 20 complete cycles per minute.

(c) Integral tanks which incorporate methods of construction and sealing not previously substantiated by satisfactory test data or service experience shall be capable of withstanding the vibration test specified in paragraph (b) of this section.

(d) (1) Tanks with nonmetallic liners shall be subjected to the sloshing portion of the test outlined under paragraph (b) of this section with fuel at room temperature.

(2) In addition, a specimen liner of the same basic construction as that to be used in the airplane shall, when installed in a suitable test tank, satisfactorily withstand the slosh test with fuel at a temperature of 110° F.

3.442 *Fuel tank installation.*

(a) The method of supporting tanks shall not be such as to concentrate the loads resulting from the weight of the fuel in the tanks. Pads shall be provided to prevent chafing between the tank and its supports. Materials employed for padding shall be nonabsorbent or shall be treated to prevent the absorption of fuels. If flexible tank liners are employed, they shall be of an approved type, and they shall be so supported that the liner is not required to withstand fluid loads. Interior surfaces of compartments for such liners shall be smooth and free of projections which are apt to cause wear of the liner, unless provisions are made for the protection of the liner at such points or unless the construction of the liner itself provides such protection. A positive pressure shall be maintained within the vapor space of all bladder cells under all conditions of operation including the critical condition of low air speed and rate of descent likely to be encountered in normal operation.

(b) Tank compartments shall be ventilated and drained to prevent the accumulation of inflammable fluids or vapors. Compartments adjacent to tanks which are an integral part of the airplane structure shall also be ventilated and drained.

(c) Fuel tanks shall not be located on the engine side of the fire wall. Not less than one-half inch of clear air space shall be provided between the fuel tank and the fire wall. No portion of engine nacelle skin which lies immediately behind a major air egress opening from the engine compartment shall act as the wall of an integral tank. Fuel tanks shall not be located in personnel compartments, except in the case of single-engine airplanes. In such cases fuel tanks the capacity of which does not exceed 25 gallons may be located in personnel compartments, if adequate ventilation and drainage are provided. In all other cases, fuel tanks shall be isolated from personnel compartments by means of fume and fuel proof enclosures.

3.442–1 *Bladder type fuel cells located in a personnel compartment (CAA interpretations which apply to sec. 3.442).* In the case where a bladder type fuel cell having a fuel capacity in excess of 25 gallons is located in a personnel compartment, a separate fume and fuel proof enclosure for the fuel cell and its retaining shell is not deemed necessary provided the retaining shell is at least equivalent to a conventional metal fuel tank in structural integrity and fume and fuel tightness. The shell surrounding the tank should be adequately drained to the exterior of the airplane.

(Supp. 10, 16 F. R. 3291, Apr. 14, 1951.)

3.443 *Fuel tank expansion space.* Fuel tanks shall be provided with an expansion space of not less than 2 percent of the tank capacity, unless the tank vent discharges clear of the aircraft in which case no expansion space will be required. It shall not be possible inadvertently to fill the fuel tank expansion space when the airplane is in the normal ground attitude.

3.444 *Fuel tank sump.*

(a) Each tank shall be provided with a drainable sump having a capacity of not less than 0.25 percent of the tank capacity or 1/16 gallon, whichever is the greater. It shall be acceptable to dispense with the sump if the fuel system is provided with a sediment bowl permitting ground inspection. The sediment bowl shall also be accessible for drainage. The capacity of the sediment chamber shall not be less than 1 ounce per each 20 gallons of the fuel tank capacity.

(b) If a fuel tank sump is provided, the capacity specified in paragraph (a) of this section shall be effective with the airplane in the normal ground attitude and in all normal flight attitudes.

(c) If a separate sediment bowl is provided in lieu of a tank sump, the fuel tank outlet shall be so located that, when the airplane is in the normal ground attitude, water will drain from all portions of the tank to the sediment bowl.

3.445 *Fuel tank filler connection.*

(a) Fuel tank filler connections shall be marked as specified in section 3.767.

(b) Provision shall be made to prevent the entrance of spilled fuel into the fuel tank compartment or any portions of the airplane other than the tank itself. The filler cap shall provide a fuel-tight seal for the main filler opening. However, small openings in the fuel tank cap for venting purposes or to permit passage of a fuel gauge through the cap shall be permissible.

3.446 *Fuel tank vents and carburetor vapor vents.*

(a) Fuel tanks shall be vented from the top portion of the expansion space. Vent outlets shall be so located and constructed as to minimize the possibility of their being obstructed by ice or other foreign matter. The vent shall be so constructed as to preclude the possibility of siphoning fuel during normal operation. The vent shall be of sufficient size to permit the rapid relief of excessive differences of pressure between the interior and exterior of the tank. Air spaces of tanks the outlets of which are interconnected shall also be interconnected. There shall be no undrainable points in the vent line where moisture is apt to accumulate with the airplane in either the ground or level flight attitude. Vents shall not terminate at points where the discharge of fuel from the vent outlet will constitute a fire hazard or from which fumes may enter personnel compartments.

(b) Carburetors which are provided with vapor elimination connections shall be provided with a vent line which will lead vapors back to one of the airplane fuel tanks. If more than one fuel tank is provided and it is necessary to use these tanks in a definite sequence for any reason, the vapor vent return line shall lead back to the fuel tank which must be used first unless the relative capacities of the tanks are such that return to another tank is preferable.

3.447–A *Fuel tank vents.* Provision shall be made to prevent excessive loss of fuel during acrobatic maneuvers including short periods of inverted flight. It shall not be possible for fuel to siphon from the vent when normal flight has been resumed after having executed any acrobatic maneuver for which the airplane is intended.

3.448 *Fuel tank outlet.* The fuel tank outlet shall be provided with a screen of from 8 to 16 meshes per inch. If a finger strainer is used, the length of the strainer shall not be less than 4 times the outlet diameter. The diameter of the strainer shall not be less than the diameter of the fuel tank outlet. Finger strainers shall be accessible for inspection and cleaning.

Fuel Pumps

3.449 *Fuel pump and pump installation.*

(a) If fuel pumps are provided to maintain a supply of fuel to the engine, at least one pump for each engine shall be directly driven by the engine. Fuel pumps shall be adequate to meet the flow requirements of the applicable portions of sections 3.433–3.436.

(b) Emergency fuel pumps shall be provided to permit supplying all engines with fuel in case of the failure of any one engine-driven pump, except that if an engine fuel injection pump which has been certificated as an integral part of the engine is used, an emergency pump is

not required. Emergency pumps shall be available for immediate use in case of the failure of any other pump. If both the normal pump and emergency pump operate continuously, means shall be provided to indicate to the crew when either pump is malfunctioning.

[3.449–1 *Fuel injection pump (CAA interpretations which apply to sec. 3.449 (b)).* The phrase "fuel injection pump" means a pump that supplies proper flow and pressure conditions for fuel injection[7a] when such injection is not accomplished in a carburetor.]

(23 F. R. 7481, Sept. 26, 1958, effective Oct. 20, 1958.)

Lines, Fittings, and Accessories

3.550 *Fuel system lines and fittings.* (See Sec. 3.638.)

(a) Fuel lines shall be installed and supported to prevent excessive vibration and to withstand loads due to fuel pressure and due to accelerated flight conditions.

(b) Fuel lines which are connected to components of the airplane between which relative motion could exist shall incorporate provisions for flexibility.

(c) Provisions for flexibility in fuel lines which may be under pressure and subjected to axial loading shall employ flexible hose assemblies rather than hose-clamp connections.

(d) Flexible hose shall be of an approved type or shall be shown to be suitable for the particular application.

(e) Flexible hoses which might be adversely affected by exposure to high temperatures shall not be employed in locations where excessive temperatures will exist during operation or after engine shutdown.

3.551 *Fuel valves.*

(a) Means shall be provided to permit the flight personnel to shut off rapidly the flow of fuel to any engine individually in flight. Valves provided for this purpose shall be located on the side of the fire wall most remote from the engine.

[7a Fuel injection is a special form of carburetion: the charging of air or gas with volatile carbon compounds. It is either an intermittent charging of air by discrete, metered quantities of fuel such as occurs in a Diesel cylinder or it is a continuous charging of air by fuel, the fuel flow being proportioned to the airflow through the engine. Examples of continuous injection are injections into the supercharger section of a reciprocating engine or into the combustion chambers of a turbine engine.]

(b) Shut-off valves shall be so constructed as to make it possible for the flight personnel to reopen the valves rapidly after they have once been closed.

(c) Valves shall be provided with either positive stops or "feel" in the on and off positions and shall be supported in such a manner that loads resulting from their operation or from accelerated flight conditions are not transmitted to the lines connected to the valve. Valves shall be so installed that the effect of gravity and vibration will tend to turn their handles to the open rather than the closed position.

(d) Fuel valve handles and their connections to the valve mechanism shall incorporate design features to minimize the possibility of incorrect installation.

3.552 *Fuel strainer.* A fuel strainer shall be provided between the fuel tank outlet and the carburetor inlet. If an engine-driven fuel pump is provided, the strainer shall be located between the tank outlet and the engine-driven pump inlet. The strainer shall be accessible for drainage and cleaning, and the strainer screen shall be removable.

Drains and Instruments

3.553 *Fuel system drains.* Drains shall be provided to permit safe drainage of the entire fuel system and shall incorporate means for locking in the closed position. The provisions for drainage shall be effective in the normal ground attitude.

3.554 *Fuel system instruments.* (See sec. 3.655 and secs. 3.670 through 3.673.)

Oil System

3.561 *Oil system.* Each engine shall be provided with an independent oil system capable of supplying the engine with an ample quantity of oil at a temperature not exceeding the maximum which has been established as safe for continuous operation. The usable oil tank capacity shall not be less than the product of the endurance of the airplane under critical operating conditions and the maximum oil consumption of the engine under the same conditions, plus a suitable margin to assure adequate system circulation and cooling.

3.561–1 *"Capacity" (CAA interpretations which apply to sec. 3.561).* The word "capacity" as used in section 3.561 is interpreted by the Administrator as follows:

(a) Only the usable fuel system capacity need be considered.

(b) In a conventional oil system (no transfer system provided) only the usable oil tank capacity shall be considered. The quantity of oil in the engine oil lines, the oil radiator, or in the feathering reserve shall not be included. When an oil transfer system is installed, and the transfer pump is so located that it can pump some of the oil in the transfer lines into the main engine oil tanks, the quantity of oil in these lines which can be pumped by the transfer pump may be added to the oil capacity.

(Supp. 1, 12 F. R. 3438, May 28, 1947, as amended by Amdt. 1, 14 F. R. 36, Jan. 5, 1949.)

3.562 *Oil cooling.* See section 3.581 and pertinent sections.

Oil Tanks

3.563 *Oil tanks.* Oil tanks shall be capable of withstanding without failure all vibration, inertia, and fluid loads to which they might be subjected in operation. Flexible oil tank liners shall be of an acceptable type.

3.564 *Oil tank tests.* Oil tank tests shall be the same as fuel tank tests (see sec. 3.441), except as follows:

(a) The applied pressure shall be 5 p. s. i. for all tank constructions instead of those specified in section 3.441 (a).

(b) In the case of tanks with nonmetallic liners, the test fluid shall be oil rather than fuel as specified in section 3.441 (d) and the slosh test on a specimen liner shall be conducted with oil at a temperature of 250° F.

3.565 *Oil tank installation.* Oil tank installations shall comply with the requirements of section 3.442 (a) and (b).

3.566 *Oil tank expansion space.* Oil tanks shall be provided with an expansion space of not less than 10 percent of the tank capacity or 1/2 gallon, whichever is greater. It shall not be possible inadvertently to fill the oil tank expansion space when the airplane is in the normal ground attitude.

3.567 *Oil tank filler connection.* Oil tank filler connections shall be marked as specified in section 3.767.

3.568 *Oil tank vent.*

(a) Oil tanks shall be vented to the engine crankcase from the top of the expansion space in such a manner that the vent connection is not covered by oil under any normal flight conditions. Oil tank vents shall be so arranged that condensed water vapor which might freeze and obstruct the line cannot accumulate at any point.

(b) *Category A.* Provision shall be made to prevent hazardous loss of oil during acrobatic maneuvers including short periods of inverted flight.

3.569 *Oil tank outlet.* The oil tank outlet shall not be enclosed or covered by any screen or other guard which might impede the flow of oil. The diameter of the oil tank outlet shall not be less than the diameter of the engine oil pump inlet. (See also sec. 3.577.)

Lines, Fittings, and Accessories

3.570 *Oil system lines, fittings, and accessories.* Oil lines shall comply with the provisions of section 3.550, except that the inside diameter of the engine oil inlet and outlet lines shall not be less than the diameter of the corresponding engine oil pump inlet and outlet.

3.571 *Oil valves.* See section 3.637.

3.572 *Oil radiators.* Oil radiators and their support shall be capable of withstanding without failure any vibration, inertia, and oil pressure loads to which they might normally be subjected.

3.573 *Oil filters.* If the engine is equipped with an oil filter, the filter shall be constructed and installed in such a manner that complete blocking of the flow through the filter element will not jeopardize the continued operation of the engine oil supply system.

3.574 *Oil system drains.* Drains shall be provided to permit safe drainage of the entire oil system and shall incorporate means for positive locking in the closed position.

3.575 *Engine breather lines.*

(a) Engine breather lines shall be so arranged that condensed water vapor which

might freeze and obstruct the line cannot accumulate at any point. Breathers shall discharge in a location which will not constitute a fire hazard in case foaming occurs and so that oil emitted from the line will not impinge upon the pilot's windshield. The breather shall not discharge into the engine air induction system.

(b) *Category A.* In the case of acrobatic type airplanes, provision shall be made to prevent excessive loss of oil from the breather during acrobatic maneuvers including short periods of inverted flight.

3.576 *Oil system instruments.* **See sections 3.655, 3.670, 3.671, and 3.674.**

3.577 *Propeller feathering system.* **If the propeller feathering system is dependent upon the use of the engine oil supply, provision, shall be made to trap a quantity of oil in the tank in case the supply becomes depleted due to failure of any portion of the lubricating system other than the tank itself. The quantity of oil so trapped shall be sufficient to accomplish the feathering operation and shall be available only to the feathering pump. The ability of the system to accomplish feathering when the supply of oil has fallen to the above level shall be demonstrated.**

Cooling

3.581 *General.* **The power-plant cooling provisions shall be capable of maintaining the temperatures of all power-plant components, engine parts, and engine fluids (oil and coolant), at or below the maximum established safe values under critical conditions of ground and flight operation.**

Tests

3.582 *Cooling tests.* **Compliance with the provisions of section 3.581 shall be demonstrated under critical ground, water, and flight operating conditions. If the tests are conducted under conditions which deviate from the highest anticipated summer air temperature (see sec. 3.583), the recorded powerplant temperatures shall be corrected in accordance with the provisions of sections 3.584 and 3.585. The corrected temperatures determined in this manner shall not exceed the maximum established safe values. The fuel used during the cooling tests shall be of the minimum octane number approved for the engines involved, and the mixture settings shall be those appropriate to the operating conditions. The test procedures shall be as outlined in sections 3.586 and 3.587.**

3.582–1 *Water taxiing tests (CAA interpretations which apply to sec. 3.582).* No water taxiing tests need be conducted on aircraft certificated under this part, except in the case of flying boats which may reasonably be expected to be taxied for extended periods.

(Supp. 10, 16 F. R. 3291, Apr. 14, 1951.)

3.583 *Maximum anticipated summer air temperatures.* **The maximum anticipated summer air temperature shall be considered to be 100° F. at sea level and to decrease from this value at the rate of 3.6° F. per thousand feet of altitude above seal level.**

3.583–1 *Powerplant winterization equipment (CAA interpretations which apply to sec. 3.583).*

(a) Cooling test results for winterization installations may be corrected to any temperature desired by the manufacturer rather than the conventional 100° F. hot day. For example, if a manufacturer chooses to demonstrate cooling to comply with requirements for a 50° or 60° F. day with winterization equipment installed, he may do so. In such a case the sea level temperature for correction purposes should be considered to be the value elected by the manufacturer with a rate of temperature drop of 3.6° F. per thousand feet above sea level.

(b) Cooling tests and temperature correction methods should be the same as for conventional cooling tests.

(c) The airplane flight manual should clearly indicate that winterization equipment must be removed whenever the temperature reaches the limit for which adequate cooling has been demonstrated. The cockpit should also be placarded accordingly. In addition, the airplane should be equipped with an ambient air temperature gauge or, alternatively, a cylinder head, barrel, or oil inlet temperature gauge (depending upon which is critical).

(d) If practical, winterization equipment such as baffles for oil radiators or for engine cooling air openings should be marked clearly to indicate the limiting temperature at which this equipment should be removed.

(e) Since winterization equipment is often supplied in kit form, accompanied by instructions for its installation, suitable information regarding temperature limitations should be included in the installation instructions for such kits.

(Supp. 10, 16 F. R. 3291, Apr. 14, 1951.)

3.584 *Correction factor for cylinder head, oil inlet, carburetor air, and engine coolant inlet temperatures.* **These temperatures shall be corrected by adding the difference between the maximum anticipated summer air temperature and the temperature of**

the ambient air at the time of the first occurrence of maximum head, air, oil, or coolant temperature recorded during the cooling test.

3.585 *Correction factor for cylinder barrel temperatures.* **Cylinder barrel temperatures shall be corrected by adding 0.7 of the difference between the maximum anticipated summer air temperature and the temperature of the ambient air at the time of the first occurrence of the maximum cylinder barrel temperature recorded during the cooling test.**

3.586 *Cooling test procedure for single-engine airplanes.* **[The engine cooling tests shall be conducted by stabilizing the engine temperatures in flight with the engines operating at not less than 75 percent of the maximum continuous power rating. After engine temperatures have stabilized, the climb shall be started at the lowest practicable altitude and continued for 1 minute with the engines operating at the take-off rating.] At the end of 1 minute, the climb shall be continued at maximum continuous power until at least 5 minutes after the occurrence of the highest temperature recorded. The climb shall not be conducted at a speed greater than the best rate-of-climb speed with maximum continuous power unless:**

(a) The slope of the flight path at the speed chosen for the cooling test is equal to or greater than the minimum required angle of climb (see sec. 3.85 (a)), and

(b) A cylinder head temperature indicator is provided as specified in section 3.675.

3.587 *Cooling test procedure for multi-engine airplanes.*

(a) ***Airplanes which meet the minimum one-engine-inoperative climb performance specified in section 3.85 (b).*** **The engine cooling test for these airplanes shall be conducted with the airplane in the configuration specified in section 3.85 (b), except that the operating engine(s) shall be operated at maximum continuous power or at full throttle when above the critical altitude. [Temperatures of the operating engines shall be stabilized in flight with the engines operating at not less than 75 percent of the maximum continuous power rating.] After stabilizing temperatures in flight, the climb shall be started at the lower of the two following altitudes and shall be continued until at least 5 minutes after the highest temperature has been recorded:**

(1) 1,000 feet below the engine critical altitude or at the lowest practicable altitude (when applicable).

(2) 1,000 feet below the altitude at which the single-engine-inoperative rate of climb is 0.02 $V_{s_0}{}^2$.

The climb shall be conducted at a speed not in excess of the highest speed at which compliance with the climb requirement of section 3.85 (b) can be shown. However, if the speed used exceeds the speed for best rate of climb with one engine inoperative, a cylinder head temperature indicator shall be provided as specified in section 3.675.

(b) ***Airplanes which cannot meet the minimum one-engine-inoperative climb performance specified in section 3.85 (b).*** **The engine cooling test for these airplanes shall be the same as in paragraph (a) of this section, except that after stabilizing temperatures in flight, the climb (or descent, in the case of airplanes with zero or negative one-engine-inoperative rate of climb) shall be commenced at as near sea level as practicable and shall be conducted at the best rate-of-climb speed (or the speed of minimum rate of descent, in the case of airplanes with zero or negative one-engine-inoperative rate of climb).**

3.587–1 *Cooling test procedure for twin-engine aircraft which do not meet the minimum one-engine-inoperative climb performance (CAA interpretations which apply to sec. 3.587 (b)).* In order to provide a practicable test procedure for compliance with this requirement, the engine temperatures should be stabilized in flight at the lowest practicable altitude above the ground, with maximum continuous power from the engine on which cooling is being investigated, and with just sufficient power on the other engine to maintain level flight at the speed for minimum rate of descent.

(Supp. 10, 16 F. R. 3292, Apr. 14, 1951.)

Liquid Cooling Systems

3.588 *Independent systems.* **Each liquid cooled engine shall be provided with an independent cooling system. The cooling system shall be so arranged that no air or vapor can be trapped in any portion of the system, except the** expansion tank, either during filling or during operation.

3.589 *Coolant tank.* A coolant tank shall be provided. The tank capacity shall not be less than 1 gallon plus 10 percent of the cooling system capacity. Coolant tanks shall be capable of withstanding without failure all vibration, inertia, and fluid loads to which they may be subjected in operation. Coolant tanks shall be provided with an expansion space of not less than 10 percent of the total cooling system capacity. It shall not be possible inadvertently to fill the expansion space with the airplane in the normal ground attitude.

3.590 *Coolant tank tests.* Coolant tank tests shall be the same as fuel tank tests (see sec. 3.441), except as follows:

(a) The 3.5 pounds per square inch pressure test of section 3.441 (a) shall be replaced by the sum of the pressure developed during the maximum ultimate acceleration with a full tank or a pressure of 3.5 pounds per square inch, whichever is greater, plus the maximum working pressure of the system.

(b) In the case of tanks with nonmetallic liners, the test fluid shall be coolant rather than fuel as specified in section 3.441 (d), and the slosh test on a specimen liner shall be conducted with coolant at operating temperature.

3.591 *Coolant tank installation.* Coolant tanks shall be supported in a manner so as to distribute the tank loads over a large portion of the tank surface. Pads shall be provided to prevent chafing between the tank and the support. Material used for padding shall be nonabsorbent or shall be treated to prevent the absorption of inflamamble fluids.

3.592 *Coolant tank filler connection.* Coolant tank filler connections shall be marked as specified in section 3.767. Provisions shall be made to prevent the entrance of spilled coolant into the coolant tank compartment or any portions of the airplane other than the tank itself. Recessed coolant filler connections shall be drained and the drain shall discharge clear of all portions of the airplane.

3.593 *Coolant lines, fittings, and accessories.* Coolant lines shall comply with the provisions of section 3.550, except that the inside diameter of the engine coolant inlet and outlet lines shall not be less than the diameter of the corresponding engine inlet and outlet connections.

3.594 *Coolant radiators.* Coolant radiators shall be capable of withstanding without failure any vibration, inertia, and coolant pressure loads to which they may normally be subjected. Radiators shall be supported in a manner which will permit expansion due to operating temperatures and prevent the transmittal of harmful vibration to the radiator. If the coolant employed is inflammable, the air intake duct to the coolant radiator shall be so located that flames issuing from the nacelle in case of fire cannot impinge upon the radiator.

3.595 *Cooling system drains.* One or more drains shall be provided to permit drainage of the entire cooling system, including the coolant tank, radiator, and the engine, when the airplane is in the normal ground attitude. Drains shall discharge clear of all portions of the airplane and shall be provided with means for positively locking the drain in the closed position. Cooling system drains shall be accessible.

3.596 *Cooling system instruments.* (See secs. 3.655, 3.670, and 3.671.)

Induction System

3.605 *General.*

(a) The engine air induction system shall permit supplying an adequate quantity of air to the engine under all conditions of operation.

(b) Each engine shall be provided with at least two separate air intake sources, except that in the case of an engine equipped with a fuel injector only one air intake source need be provided, if the air intake, opening, or passage is unobstructed by a screen, filter, or other part on which ice might form and so restrict the air flow as to affect adversely engine operation. It shall be permissible for primary air intakes to open within the cowling only if that portion of the cowling is isolated from the engine accessory section by means of a fire-resistant diaphragm or if provision is made to prevent the emergence of backfire flames. Alternate air intakes shall be located in a sheltered position and shall not open within the cowling unless they are so located that the emergence of backfire flames will not result in a hazard. Supplying air to the engine through the alternate air intake system of the carburetor air preheater shall not result in the loss of excessive power in addition to the power lost due to the rise in the temperature of the air.

3.606 *Induction system de-icing and anti-icing provisions.* The engine air induction system shall incorporate means for the prevention and elimination of ice accumulations in accordance with the provisions in this section. It shall be demonstrated that compliance with the provisions outlined in the following paragraphs can be accomplished when the airplane is operating in air at a temperature of 30° F. when the air is free of visible moisture.

(a) Airplanes equipped with sea level engines employing conventional venturi carburetors

shall be provided with a preheater capable of providing a heat rise of 90° F. when the engine is operating at 75 percent of its maximum continuous power.

(b) Airplanes equipped with altitude engines employing conventional venturi carburetors shall be provided with a preheater capable of providing a heat rise of 120° F. when the engine is operating at 75 percent of its maximum continuous power.

(c) Airplanes equipped with altitude engines employing carburetors which embody features tending to reduce the possibility of ice formation shall be provided with a preheater capable of providing a heat rise of 100° F. when the engine is operating at 60 percent of its maximum continuous power. However, the preheater need not provide a heat rise in excess of 40° F. if a fluid de-icing system complying with the provisions of sections 3.607–3.609 is also installed.

(d) Airplanes equipped with sea level engines employing carburetors which embody features tending to reduce the possibility of ice formation shall be provided with a sheltered alternate source of air. The preheat supplied to this alternate air intake shall be not less than that provided by the engine cooling air downstream of the cylinders.

3.606–1 *Induction system de-icing provisions (CAA policies which apply to sec. 3.606).*

(a) A series of pressure type carburetors for small engines has been developed which incorporate the feature of injecting fuel into the intake air at a point downstream from the throttle and carburetor venturi. This feature tends to greatly reduce the possibility of ice formation in the engine induction system and the results of extensive tests have demonstrated the carburetors to be relatively free of icing hazards.

(b) In order to outline the limitations of our approval for the elimination of preheat on the carburetors, and to provide what are considered equivalent safety margins the following is stipulated:

(1) This approval applies only to sea level engines of the general power class with which the largest of these pressure carburetors has been tested. No tests have as yet been conducted on any altitude engines. The largest of these carburetors which has been tested at present is a model which is intended for use on engines in the general range of approximately 220 horsepower.

(2) Unless the main carburetor air intake is located in a sheltered position where it is free from impact icing possibilities, a sheltered alternate air intake should be provided even though there is no preheater.

(c) During tests of the non-icing qualities of these carburetors, it was found that in some cases poor idling of the engine was encountered and this was attributed to a possible ice formation in the internal carburetor passage which acts as the air bleed for the main discharge nozzle. As a result, it is necessary to provide a small intensifier tube to supply hot air to the air bleed side of the main discharge nozzle on installation in which the carburetor and a portion of the induction system are exposed to the exterior of the airplane. When the installation is completely cowled, the hot air bleed will not be necessary.

(Supp. 10, 16 F. R. 3292, Apr. 14, 1951.)

3.607 ***Carburetor de-icing fluid flow rate.*** **The system shall be capable of providing each engine with a rate of fluid flow, expressed in pounds per hour, of not less than 2.5 multiplied by the square root of the maximum continuous power of the engine. This flow shall be available to all engines simultaneously. The fluid shall be introduced into the air induction system at a point close to, and upstream from, the carburetor. The fluid shall be introduced in a manner to assure its equal distribution over the entire cross section of the induction system air passages.**

3.608 ***Carburetor fluid de-icing system capacity.*** **The fluid de-icing system capacity shall not be less than that required to provide fluid at the rate specified in section 3.607 for a time equal to 3 percent of the maximum endurance of the airplane. However, the capacity need not in any case exceed that required for 2 hours of operation nor shall it be less than that required for 20 minutes of operation at the above flow rate. If the available preheat exceeds 50° F. but is less than 100° F., it shall be permissible to decrease the capacity of the system in proportion to the heat rise available in excess of 50° F.**

3.609 *Carburetor fluid de-icing system detail design.* Carburetor fluid de-icing systems shall comply with provisions for the design of fuel systems, except as specified in sections 3.607 and 3.608, unless such provisions are manifestly inapplicable.

3.610 *Carburetor air preheater design.* Means shall be provided to assure adequate ventilation of the carburetor air preheater when the engine is being operated in cold air. The preheater shall be constructed in such a manner as to permit inspection of exhaust manifold parts which it surrounds and also to permit inspection of critical portions of the preheater itself.

3.611 *Induction system ducts.* Induction system ducts shall be provided with drains which will prevent the accumulation of fuel or moisture in all normal ground and flight attitudes. No open drains shall be located on the pressure side of turbo-supercharger installations. Drains shall not discharge in a location which will constitute a fire hazard. Ducts which are connected to components of the airplane between which relative motion may exist shall incorporate provisions for flexibility.

3.612 *Induction system screens.* If induction system screens are employed, they shall be located upstream from the carburetor. It shall not be possible for fuel to impinge upon the screen. Screens shall not be located in portions of the induction system which constitute the only passage through which air can reach the engine, unless the available preheat is 100° F. or over and the screen is so located that it can be de-iced by the application of heated air. De-icing of screens by means of alcohol in lieu of heated air shall not be acceptable.

Exhaust System

3.615 *General.*

(a) The exhaust system shall be constructed and arranged in such a manner as to assure the safe disposal of exhaust gases without the existence of a hazard of fire or carbon monoxide contamination of air in personnel compartments.

(b) Unless suitable precautions are taken, exhaust system parts shall not be located in close proximity to portions of any systems carrying inflammable fluids or vapors nor shall they be located under portions of such systems which may be subject to leakage. All exhaust system components shall be separated from adjacent inflammable portions of the airplane which are outside the engine compartment by means of fireproof shields. Exhaust gases shall not be discharged at a location which will cause a glare seriously affecting pilot visibility at night, nor shall they discharge within dangerous proximity of any fuel or oil system drains. All exhaust system components shall be ventilated to prevent the existence of points of excessively high temperature.

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

3.617 *Exhaust heat exchangers.*

(a) Exhaust heat exchangers shall be constructed and installed in such a manner as to assure their ability to withstand without failure all vibration, inertia, and other loads to which they might normally be subjected. Heat exchangers shall be constructed of materials which are suitable for continued operation at high temperatures and which are adequately resistant to corrosion due to products contained in exhaust gases.

(b) Provisions shall be made for the inspection of all critical portions of exhaust heat exchangers, particularly if a welded construction is employed. Heat exchangers shall be ventilated under all conditions in which they are subject to contact with exhaust gases.

3.618 *Exhaust heat exchangers used in ventilating air heating systems.* Heat exchangers of this type shall be so constructed as to preclude the possibility of exhaust gases entering the ventilating air.

Fire Wall and Cowling

3.623 *Fire walls.* All engines, auxiliary power units, fuel burning heaters, and other

combustion equipment which are intended for operation in flight shall be isolated from the remainder of the airplane by means of fire walls, or shrouds, or other equivalent means.

3.623–1 *Fireproof materials for fire walls* (*CAA rules which apply to sec. 3.623*).

(a) The test for demonstrating compliance with criteria for fireproof material or components shall subject the material or unit to a 2,000±50° F. flame. Sheet materials shall be tested by subjecting a sample approximately 10 inches square to a flame from a suitable burner. The flame shall be large enough to maintain the required test temperature over an area approximately five inches square.

(b) Fire-wall materials and fittings shall resist flame penetration for 15 minutes.

(c) The following materials are considered satisfactory for use in fire walls or shrouds without being tested as outlined in paragraphs (a) and (b) of this section:

(1) Stainless steel sheet, 0.015 inch thick.

(2) Mild steel sheet coated with aluminum or otherwise protected against corrosion, 0.018 inch thick.

(3) Terne plate, 0.018 inch thick.

(4) Monel metal, 0.018 inch thick.

(5) Steel or copper base alloy fire wall fittings.

(Supp. 10, 16 F. R. 3292, Apr. 14, 1951.)

3.624 ***Fire wall construction.***

(a) Fire walls and shrouds shall be constructed in such a manner that no hazardous quantity of liquids, gases, or flame could pass from the engine compartment to other portions of the airplane. All openings in the fire wall or shroud shall be sealed tight with fireproof grommets, bushings, or fire-wall fittings, except that, such seals of fire-resistant materials shall be acceptable for use on single-engine airplanes and multiengine airplanes not required to comply with section 3.85 (b) or section 3.85a (b), if such airplanes are equipped with engine(s) having a volumetric displacement of 1,000 cubic inches or less; and if the openings in the fire walls or shrouds are such that, without seals, the passage of a hazardous quantity of flame could not result.

(b) Fire walls and shrouds shall be constructed of fireproof material and shall be protected against corrosion. The following materials have been found to comply with this requirement.

(1) Heat- and corrosion-resistant steel 0.015 inch thick.

(2) Low carbon steel, suitably protected against corrosion, 0.018 inch thick.

3.625 ***Cowling.***

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

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

Power-Plant Controls and Accessories

Controls

3.627 ***Power-plant controls.*** **Power-plant controls shall comply with the provisions of sections 3.384 and 3.762. Controls shall maintain any necessary position without constant attention by the flight personnel and shall not tend to creep due to control loads or vibration. Flexible controls shall be of an acceptable type. Controls shall have adequate strength and rigidity to withstand loads without failure or excessive deflection.**

3.628 ***Throttle controls.*** **A throttle control shall be provided to give independent control for each engine. Throttle controls shall afford a positive and immediately responsive means of controlling the engine(s). Throttle controls shall be grouped and arranged in such a manner as to permit separate control of each engine and also simultaneous control of all engines.**

3.629 ***Ignition switches.*** **Ignition switches shall provide control for each ignition circuit on each engine. It shall be possible**

to shut off quickly all ignition on multiengine airplanes, either by grouping of the individual switches or by providing a master ignition control. If a master control is provided, suitable means shall be incorporated to prevent its inadvertent operation.

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

3.631 *Propeller speed and pitch controls.* (See also sec. 3.421 (a).) If propeller speed or pitch controls are provided, the controls shall be grouped and arranged in such a manner as to permit control of all propellers, both separately and together. The controls shall permit ready synchronization of all propellers on multiengine airplanes.

3.632 *Propeller feathering controls.* If propeller feathering controls are provided, a separate control shall be provided for each propeller. Propeller feathering controls shall be provided with means to prevent inadvertent operation.

3.633 *Fuel system controls.* Fuel system controls shall comply with requirements of section 3.551 (c).

3.634 *Carburetor air preheat controls.* Separate controls shall be provided to regulate the temperature of the carburetor air for each engine.

Accessories

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

3.636 *Engine battery ignition systems.*

(a) Battery ignition systems shall be supplemented with a generator which is automatically made available as an alternate source of electrical energy to permit continued engine operation in the event of the depletion of any battery.

(b) The capacity of batteries and generators shall be sufficient to meet the simultaneous demands of the engine ignition system and the greatest demands of any of the airplane's electrical system components which may draw electrical energy from the same source. Consideration shall be given to the condition of an inoperative generator, and to the condition of a completely depleted battery when the generator is running at its normal operating speed. If only one battery is provided, consideration shall also be given to the condition in which the battery is completely depleted and the generator is operating at idling speed.

(c) Means shall be provided to warn the appropriate flight personnel if malfunctioning of any part of the electrical system is causing the continuous discharging of a battery used for engine ignition. (See sec. 3.629 for ignition switches.)

Power-Plant Fire Protection

[3.637 *Flammable fluids; shutoff means.* The provisions of paragraphs (a) through (d) of this section shall be applicable to multiengine aircraft which are required to comply with the provisions of section 3.85 (b).

[(a) Means for each individual engine shall be provided for shutting off or otherwise preventing hazardous quantities of fuel, oil, deicer, and other flammable fluids from flowing into, within, or through the engine compartment except that means need not be provided to shut off flow in lines forming an integral part of an engine. Closing the fuel shutoff valve for any engine shall not make any of the fuel supply unavailable to the remaining engines

[(b) Operation of the shutoff means shall not interfere with the subsequent emergency operation of other equipment, such as feathering the propeller.

[(c) The shutoff means shall be located outside of the engine compartment unless an equally high degree of safety is otherwise provided. It shall be shown that no hazardous quantity of flammable fluid could drain into the engine compartment after shutting off has been accomplished.

[(d) Provisions shall be made to guard against inadvertent operation of the shutoff means and to make it possible for the crew to reopen the shutoff means in flight after it has once been closed.

[3.638 *Lines and fittings.* All lines and fittings carrying flammable fluids or gases in the engine compartment shall comply with the rain, oil, and other detrimental elements. [Means shall be provided for indicating the adequacy of the power being supplied to the instruments.] In addition, the following provisions shall be applicable to multiengine airplanes:

(a) There shall be provided at least two independent sources of power, a manual or an automatic means for selecting the power source, and a means for indicating the adequacy of the power being supplied by each source.

(b) The installation and power supply systems shall be such that failure of one instrument or of the energy supply from one source will not interfere with the proper supply of energy to the remaining instruments or from the other source.

3.669 ***Flight director instrument.*** If a flight director instrument is installed, its installation shall not affect the performance and accuracy of the required instruments. A means for disconnecting the flight director instrument from the required instruments or their installations shall be provided.

Power-Plant Instruments

3.670 ***Operational markings.*** Instruments shall be marked as specified in section 3.759.

3.671 ***Instrument lines.*** Power-plant instrument lines shall comply with the provisions of section 3.550. In addition, instrument lines carrying inflammable fluids or gases under pressure shall be provided with restricted orifices or other safety devices at the source of the pressure to prevent escape of excessive fluid or gas in case of line failure.

3.672 ***Fuel quantity indicator.*** Means shall be provided to indicate to the flight personnel the quantity of fuel in each tank during flight. Tanks, the outlets and air spaces of which are interconnected, may be considered as one tank and need not be provided with separate indicators. Exposed sight gauges shall be so installed and guarded as to preclude the possibility of breakage or damage. Sight gauges which form a trap in which water can collect and freeze shall be provided with means to permit drainage on the ground. Fuel quantity gauges shall be calibrated to read zero during level flight when the quantity of fuel remaining in the tank is equal to the unusable fuel supply as defined by section 3.437. Fuel gauges need not be provided for small auxiliary tanks which are used only to transfer fuel to other tanks, provided that the relative size of the tanks, the rate of fuel transfer, and the instructions pertaining to the use of the tanks are adequate to guard against overflow and to assume that the crew will receive prompt warning in case transfer is not being achieved as intended.

3.672–1 *Means to indicate fuel quantity (CAA policies which apply to sec. 3.672).* The Administrator will accept, as a "means to indicate to the flight personnel the quantity of fuel in each tank during flight," a fuel tank calibrated to read in either gallons or pounds, providing the gauge is clearly marked to indicate which scale is being used.

(Supp. 1, 12 F. R. 3438, May 28, 1947, as amended by Amdt. 1, 14 F. R. 36, Jan. 5, 1949.)

3.673 ***Fuel flowmeter system.*** When a fuel flowmeter system is installed in the fuel line(s), the metering component shall be of such design as to include a suitable means for bypassing the fuel supply in the event that malfunctioning of the metering component offers a severe restriction to fuel flow.

3.674 ***Oil quantity indicator.*** Ground means, such as a stick gauge, shall be provided to indicate the quantity of oil in each tank. If an oil transfer system or a reserve oil supply system is installed, means shall be provided to indicate to the flight personnel during flight the quantity of oil in each tank.

3.675 ***Cylinder head temperature indicating system for air-cooled engines.*** A cylinder head temperature indicator shall be provided for each engine on airplanes equipped with cowl flaps. In the case of airplanes which do not have cowl flaps, an indicator shall be provided if compliance with the provisions of section 3.581 is demonstrated at a speed in excess of the speed of best rate of climb.

Electrical Systems and Equipment

3.681 ***Installation.***

(a) Electrical systems in airplanes shall be free from hazards in themselves, in their method of operation, and in their effects on other parts of the airplane. Electrical equip-

ment shall be of a type and design adequate for the use intended. Electrical systems shall be installed in such a manner that they are suitably protected from fuel, oil, water, other detrimental substances, and mechanical damage.

(b) Items of electrical equipment required for a specific type of operation are listed in other pertinent parts of this subchapter.

3.681-1 *Shielding of flare circuits (CAA policies which apply to sec. 3.681).*

Flare circuits should be shielded or separated from other circuits far enough to preclude induction of other current into flare circuit.

(Supp. 10, 16 F. R. 3292, Apr. 14, 1951.)

3.681-2 *Generator capacity (CAA policies which apply to sec. 3.681).* When a generator is required,[7b] its capacity should be sufficient to supply during flight all probable combinations of continuous loads,[7c] with adequate reserve for storage battery charging. In no case should the maximum probable continuous load exceed 80 percent of total generator rating.

(22 F. R. 1025, Feb. 20, 1957, effective Mar. 15, 1957.)

Batteries

3.682 *Batteries.* When an item of electrical equipment which is essential to the safe operation of the airplane is installed, the battery required shall have sufficient capacity to supply the electrical power necessary for dependable operation of the connected electrical equipment.

3.682-1 *Dry-cell batteries (CAA policies which apply to sec. 3.682).* When a battery is installed to provide power for electrical equipment which is essential to the safe operation of the airplane, it should be of a type whose preflight state of charge can readily be determined by simple and reliable means. Dry-cell batteries are not considered to be of this type, and should not be used to supply essential electrical equipment.

[7b] A generator of adequate capacity is required by section 43.30 (c) (7) of this subchapter for operation under instrument flight rules.

[7c] Continuous loads are those which draw current continuously during flight, such as radio equipment and position lights. Occasional intermittent loads (such as landing gear, flaps, or landing lights) are not considered.

3.683 *Protection against acid.* If batteries are of such a type that corrosive substance may escape during servicing or flight, means such as a completely enclosed compartment shall be provided to prevent such substances from coming in contact with other parts of the airplane which are essential to safe operation. Batteries shall be accessible for servicing and inspection on the ground.

3.684 *Battery vents.* The battery container or compartment shall be vented in such manner that gases released by the battery are carried outside the airplane.

Generators

3.685 *Generator.* Generators shall be capable of delivering their continuous rated power.

3.686 *Generator controls.*

[(a)] Generator voltage control equipment shall be capable of dependably regulating the generator output within rated limits.

[(b) A generator reverse current cut-out shall be incorporated and designed to disconnect the generator from the battery and other generators when the generator is developing a voltage of such value that current sufficient to cause malfunctioning can flow into the generator.

[Instruments

[3.687 *Electric power system instruments.* Means shall be provided to indicate to appropriate crew members those electric power system quantities which are essential for the safe operation of the system.

[NOTE: For direct current systems an ammeter which can be switched into each generator feeder would be acceptable. When only one generator is installed, the ammeter may be in the battery feeder.]

Master Switch

3.688 *Arrangement.* If electrical equipment is installed, a master switch arrangement shall be provided which will disconnect all sources of electrical power from the main distribution system at a point adjacent to the power sources.

3.688-1 *Load circuit connections with respect to master switch (CAA policies which apply to sec. 3.688).* All load circuits should be connected to electric power sources in such manner that the master switch can interrupt service, failure of the conductor. Cable insulation should be flame-resistant and should not emit toxic fumes when overheated. Cable conforming to Military Specification MIL-W-5086 or the equivalent is acceptable for this application.

(22 F. R. 6883, Aug. 27, 1957, effective Sept. 15, 1957.)

Switches

3.694 *Switches.* Switches shall be capable of carrying their rated current and shall be of such construction that there is sufficient distance or insulating material between current carrying parts and the housing so that vibration in flight will not cause shorting.

3.695 *Switch installation.* Switches shall be so installed as to be readily accessible to the appropriate crew member and shall be suitably labeled as to operation and the circuit controlled.

Instrument Lights

3.696 *Instrument lights.* If instrument lights are required, they shall be of such construction that there is sufficient distance or insulating material between current carrying parts and the housing so that vibration in flight will not cause shorting. They shall provide sufficient illumination to make all instruments and controls easily readable and discernible, respectively.

3.696-1 *Instrument lights (CAA interpretations which apply to sec. 3.696).* The use of the cabin dome light is not considered adequate to comply with the provision of section 3.696.

(Supp. 10, 16 F. R. 3992, Apr. 14, 1951.)

3.697 *Instrument light installation.* Instrument lights shall be installed in such a manner that their direct rays are shielded from the pilot's eyes. Direct rays shall not be reflected from the windshield or other surfaces into the pilot's eyes.

Landing Lights

3.698 *Landing lights.* If landing lights are installed, they shall be of an acceptable type.

3.699 *Landing light installation.* Landing lights shall be so installed that there is no dangerous glare visible to the pilot and also so that the pilot is not seriously affected by halation. They shall be installed at such a location that they provide adequate illumination for night landing.

Position Lights

3.700 *Position light system installation.*

(a) *General.* The provisions of sections 3.700 through 3.703 shall be applicable to the position light system as a whole. The position light system shall include the items specified in paragraphs (b) through (e) of this section.

(b) *Forward position lights.* Forward position lights shall consist of a red and a green light spaced laterally as far apart as practicable

and installed forward on the airplane in such a location that, with the airplane in normal flying position, the red light is displayed on the left side and the green light is displayed on the right side. The individual lights shall be of an approved type.

(c) *Rear position light.* The rear position light shall be a white light mounted as far aft as practicable. The light shall be of an approved type.

(d) *Circuit.* The two forward position lights and the rear position light shall constitute a single circuit.

(e) *Light covers and color filters.* Light covers or color filters used shall be of noncombustible material and shall be constructed so that they will not change color or shape or suffer any appreciable loss or light transmission during normal use.

3.700-1 *Red passing lights (CAA policies which apply to sec. 3.700 (a)).* When it is desired to improve the conspicuity of the aircraft, a steady red light, commonly known as a passing light, may be installed. This light is not considered to be a position light and therefore need not be type certificated. When installed, its location should be one of the following:

(a) Within the left landing light unit.

(b) On the centerline of the aircraft nose.

(c) In the leading edge of the left wing, outboard of the propeller disc.

(Supp. 11, 16 F. R. 3211, Apr. 12, 1951.)

3.701 *Position light system dihedral angles.* The forward and rear position lights as installed on the airplane shall show unbroken light within dihedral angles specified in paragraphs (a) through (c) of this section.

(a) Dihedral angle L (left) shall be considered formed by two intersecting vertical planes, one parallel to the longitudinal axis of the airplane and the other at 110° to the left of the first, when looking forward along the longitudinal axis.

(b) Dihedral angle R (right) shall be considered formed by two intersecting vertical planes, one parallel to the longitudinal axis of the airplane and the other at 110° to the right of the first, when looking forward along the longitudinal axis.

(c) Dihedral angle A (aft) shall be considered formed by two intersecting vertical planes making angles of 70° to the right and 70° to the left, respectively, looking aft along the longitudinal axis, to a vertical plane passing through the longitudinal axis.

3.702 *Position light distribution and intensities.*

(a) *General.* The intensities prescribed in this section are those to be provided by new equipment with all light covers and color filters in place. Intensities shall be determined with the light source operating at a steady value equal to the average luminous output of the light source at the normal operating voltage of the airplane. The light distribution and intensities of position lights shall comply with the provisions of paragraph (b) of this section.

(b) *Forward and rear position lights.* The light distribution and intensities of forward and rear position lights shall be expressed in terms of minimum intensities in the horizontal plane, minimum intensities in any vertical plane, and maximum intensities in overlapping beams within dihedral angles L, R, and A, and shall comply with the provisions of subparagraphs (1) through (3) of this paragraph.

(1) *Intensities in horizontal plane.* The intensities in the horizontal plane shall not be less than the values given in Figure 3-15. (The horizontal plane is the plane containing the longitudinal axis of the airplane and is perpendicular to the plane of symmetry of the airplane.)

(2) *Intensities above and below horizontal.* The intensities in any vertical plane shall not be less than the appropriate value given in Figure 3-16, where I is the minimum intensity prescribed in Figure 3-15 for the corresponding angles in the horizontal plane. (Vertical planes are planes perpendicular to the horizontal plane.)

(3) *Overlaps between adjacent signals.* The intensities in overlaps between adjacent signals shall not exceed the values given in Figure 3-17, except that higher intensities in the overlaps shall be acceptable with the use of main beam intensities substantially greater than the minima specified in Figures 3-15 and 3-16 if the overlap intensities in relation to the main beam intensities are such as not to affect adversely signal clarity.

Dihedral angle (light involved)	Angle from right or left of longitudinal axis, measured from dead ahead	Intensity (candles)
L and *R* (forward red and green)	0° to 10°	40
	10° to 20°	30
	20° to 110°	5
A (rear white)	110° to 180°	20

Figure 3–15.—Minimum Intensities in the Horizontal Plane of Forward and Rear Position Lights.

Angle above or below horizontal	Intensity
0°	1.00 I.
0° to 5°	.90 I.
5° to 10°	.80 I.
10° to 15°	.70 I.
15° to 20°	.50 I.
20° to 30°	.30 I.
30° to 40°	.10 I.
40° to 90°	At least 2 candles.

Figure 3–16.—Minimum Intensities in any Vertical Plane of Forward and Rear Position Lights.

Overlaps	Maximum intensity	
	Area A (candles)	Area B (candles)
Green in dihedral angle *L*	10	1
Red in dihedral angle *R*	10	1
Green in dihedral angle *A*	5	1
Red in dihedral angle *A*	5	1
Rear white in dihedral angle *L*	5	1
Rear white in dihedral angle *R*	5	1

Note: Area A includes all directions in the adjacent dihedral angle which pass through the light source and which intersect the common boundary plane at more than 10 degrees but less than 20 degrees. Area B includes all directions in the adjacent dihedral angle which pass through the light source and which intersect the common boundary plane at more than 20 degrees.

Figure 3–17.—Maximum Intensities in Overlapping Beams of Forward and Rear Position Lights.

3.702–1 *Rear position light installation* (*CAA interpretations which apply to sec. 3.702*). A single rear position light may be installed in a position displaced laterally from the plane of symmetry of an airplane if the axis of the maximum cone of illumination is parallel to the flight path in level flight, and if there is no obstruction aft of the light and between planes 70° to the right and left of the axis of maximum illumination.

(Supp. 10, 16 F. R. 3292, Apr. 14, 1951.)

[3.702–2 *Overlaps between high intensity forward position lights* (*CAA policies which apply to sec. 3.702* (*b*) (*3*)). When the peak intensity of the forward position lights is greater than 100 candles, the maximum overlap intensities between them may exceed the values given in figure 3–17 provided the overlap intensity in Area A is not greater than 10 percent of peak position light intensity and the overlap intensity in Area B is not greater than 2.5 percent of peak position light intensity.[8d]]

(23 F. R. 1001, Feb. 15, 1958, effective Mar. 10, 1958.)

3.703 *Color specifications.* The colors of the position lights shall have the International Commission on Illumination chromaticity coordinates as set forth in paragraphs (a) through (c) of this section.

(a) *Aviation red.*

y is not greater than 0.335,

z is not greater than 0.002;

(b) *Aviation green.*

x is not greater than $0.440-0.320y$,

x is not greater than $y-0.170$,

y is not less than $0.390-0.170x$;

(c) *Aviation white.*

x is not less than 0.350,

x is not greater than 0.540,

$y-y_0$ is not numerically greater than 0.01, y_0 being the y coordinate of the Planckian radiator for which $x_0=x$.

Riding Light

3.704 *Riding light.*

(a) When a riding (anchor) light is required for a seaplane, flying boat, or amphibian, it shall be capable of showing a white light for at least 2 miles at night under clear atmospheric conditions.

[[8d] Overlap intensities should be determined with the position lights installed in their actual aircraft locations, since adjacent aircraft structure will often provide some cutoff in the overlap area.]

(b) The riding light shall be installed to show the maximum unbroken light practicable when the airplane is moored or drifting on the water. Externally hung lights shall be acceptable.

Anti-collision Light System

3.705 *Anti-collision light system.* An airplane to be eligible for night operation shall have installed an anti-collision light system. Such system shall consist of one or more approved anti-collision lights so located that the emitted light will not be detrimental to the crew's vision and will not detract from the conspicuity of the position lights. The system shall comply with the provisions of paragraphs (a) through (d) of this section.

(a) *Field of coverage.* The system shall consist of such lights as will afford coverage of all vital areas around the airplane with due consideration to the physical configuration and the flight characteristics of the airplane. In any case, the field of coverage shall extend in all directions within 30° above and 30° below the horizontal plane of the airplane, except that a solid angle or angles of obstructed visibility totaling not more than .03 steradians shall be permissible within a solid angle equal to .15 steradians centered about the longitudinal axis in the rearward direction.

(b) *Flashing characteristics.* The arrangement of the system, i. e., number of light sources, beam width, speed of rotation, etc., shall be such as to give an effective flash frequency of not less than 40 and not more than 100 cycles per minute. The effective flash frequency shall be the frequency at which the airplane's complete anti-collision light system is observed from a distance, and shall apply to all sectors of light including the overlaps which might exist when the system consists of more than one light source. In overlaps, flash frequencies higher than 100 cycles per minute shall be permissible, except that they shall not be higher than 180 cycles per minute.

(c) *Color.* The color of the anti-collision lights shall be aviation red in accordance with the specifications of section 3.703 (a).

(d) *Light intensity.* The minimum light intensities in all vertical planes, measured with the red filter and expressed in terms of "effective" intensities, shall be in accordance with Figure 3-18. The following relation shall be assumed:

$$I_e=\frac{\int_{t_1}^{t_2} I(t)dt}{0.2+(t_2-t_1)};$$

where:

I_e=effective intensity (candles);
$I(t)$=instantaneous intensity as a function of time;
t_2-t_1=flash time interval (seconds).

NOTE: Normally, the maximum value of effective intensity is obtained when t_2 and t_1 are so chosen that the effective intensity is equal to the instantaneous intensity at t_2 and t_1.

Angle above or below horizontal plane	Effective intensity (candles)
0° to 5°	100
5° to 10°	60
10° to 20°	20
20° to 30°	10

Figure 3-18.—Minimum Effective Intensities for Anti-collision Lights.

3.705-1 *Anticollision light standards (CAA policies which apply to sec. 3.705).* The anticollision light standards in section 3.705 apply to aircraft for which an application for a type certificate is made on or after April 1, 1957. When anticollision lights are installed on aircraft for which an application for a type certificate was made before April 1, 1957, the applicant may conform either to section 3.705 or to the standards listed below:

(a) Anticollision lights (when installed) should be of the rotating beacon type installed on top of the fuselage or tail in such a location that the light will not be detrimental to the flight crew's vision and will not detract from the conspicuity of the position lights. If there is no acceptable location on top of the fuselage or tail, a bottom fuselage installation may be used.

(b) The color of the anticollision light should be aviation red in accordance with the specifications of section 3.703.

(c) The arrangement of the anticollision light, i. e., number of light sources, beam width, speed of rotation, etc., should be such as to give an effective flash frequency of not less than 40 and not more than 100 cycles per minute, with an on-off ratio not less than 1:75.

(22 F. R. 3653, May 24, 1957, effective June 15, 1957 as amended 22 F. R. 6883, Aug. 27, 1957, effective Sept. 15, 1957.)

Safety Equipment; Installation

3.711 *Marking.* Required safety equipment which the crew is expected to operate at a time of emergency, such as flares and automatic life raft releases, shall be readily accessible and plainly marked as to its method of operation. When such equipment is carried in lockers, compartments, or other storage places, such storage places shall be marked for the benefit of passengers and crew.

3.712 *De-icers.* When pneumatic de-icers are installed, the installation shall be in accordance with approved data. Positive means shall be provided for the deflation of the pneumatic boots.

3.713 *Flare requirements.* When parachute flares are required, they shall be of an approved type.

3.714 *Flare installation.* Parachute flares shall be releasable from the pilot compartment and so installed that danger of accidental discharge is reduced to a minimum. The installation shall be demonstrated in flight to eject flares satisfactorily, except in those cases where inspection indicates a ground test will be adequate. If the flares are ejected so that recoil loads are involved, structural provisions for such loads shall be made.

3.715 *Safety belts.* Safety belts shall be of an approved type. In no case shall the rated strength of the safety belt be less than that corresponding with the ultimate load factors specified in section 3.386 (a), taking due account of the dimensional characteristics of the safety belt installation for the specific seat or berth arrangement. Safety belts shall be attached so that no part of the anchorage will fail at a load lower than that corresponding with the ultimate load factors equal to those specified in section 3.386 (a) multiplied by a factor of 1.33. In the case of safety belts for berths, the forward load factor need not be applied.

Emergency Flotation and Signaling Equipment

3.716 *Rafts and life preservers.* Rafts and life preservers shall be of an approved type.

3.716–1 *Life rafts and life preservers (CAA rules which apply to sec. 3.716).*

(a) The minimum safety requirements for life preservers and life rafts which are intended for use in certificated civil aircraft engaged in over-water operations have been established by the Administrator in the following Technical Standard Orders:

(1) No. TSO–C12, "Life Rafts," effective August 1, 1948 (sec. 514.22 of this title).

(2) No. TSO–C13, "Life Preservers," effective August 1, 1948 (sec. 514.23 of this title).

(Supp. 10, 16 F. R. 3292, Apr. 14, 1951.)

3.717 *Installation.* When such emergency equipment is required, it shall be so installed as to be readily available to the crew and passengers. Rafts released automatically or by the pilot shall be attached to the airplane by means of a line to keep them adjacent to the airplane. The strength of the line shall be such that it will break before submerging the empty raft.

3.718 *Signaling device.* Signaling devices, when required by other parts of the regulations of this subchapter, shall be accessible, function satisfactorily, and be free from any hazard in their operation.

Radio Equipment; Installation

3.721 *General.* Radio equipment and installations in the airplane shall be free from hazards in themselves, in their method of operation, and in their effects on other components of the airplane.

3.721–1 *Radio equipment installation (CAA intrepretations which apply to sec. 3.721).* Engineering flight tests are not required for equipment installations unless a particular installation could conceivably interfere with flight operation of airplane or change the airplane configuration so that performance or flight characteristics became adversely affected.

(Supp. 10, 16 F. R. 3292, Apr. 14, 1951.)

[3.721–2 *Radio equipment installations (CAA policies which apply to sec. 3.721).*

[(a) *Wiring.* Radio installations should be wired to minimize the possibility of fire or smoke hazards or the cause of unsatisfactory operation of the radio equipment.

[(1) The radio installation should be connected to the airplane electrical system at a terminal strip or by a plug and receptacle connection.

[(i) If a terminal strip is used, it should be designed or mounted so that loose metallic objects cannot fall across the terminal posts. Posts should be No. 6 or larger to permit proper tightening of the nuts thus providing maximum current-carrying capacity without a danger of shearing the studs.

[(ii) If a plug and receptacle type of connection is used, the soldered connections of the wire to the plug and receptacle inserts should be individually insulated from each other and from other metallic parts of the plug and receptacle.

[(2) Junction boxes should be used for enclosure of terminal strips. The boxes should be made of either fire-resistant or nonabsorbent plastic material. They should be of sufficiently rigid construction to prevent "oil-canning" of the box sides thus avoiding the possibility of the sides causing internal shorts. They should be designed and installed to permit easy access to the enclosed terminals and to allow any loose metallic parts to fall away from the terminals. Sufficient space should be provided in the junction box so that it will not be necessary to bend the wires sharply as they leave the terminal strip. The terminal strip should be mounted within the box rather than on the inside of the box cover.

[(3) Interconnecting wires and cables between various pieces of radio equipment should be supported by insulated clamps so that they do not rub against the airplane or each other under vibration conditions encountered in flight.

[(b) *Location of radio equipment.* The equipment, controls, and indicators should be located where they can be satisfactorily operated and read respectively from the appropriate crew member station. The equipment should be so located that there is sufficient air circulation to avoid overheating of the equipment. Also, clearance should be provided between high temperature areas of the equipment and readily flammable parts of the airplane.

[(c) *Mounting of radio equipment.* The equipment should be attached to the airplane with self-locking devices to prevent loosening in service. Mechanical remote control devices should have the control cable so routed and so supported as to prevent kinking, binding or abrasion. Items mounted on shock mounts should have sufficient clearance for normal vibration and swaying of the equipment without hitting adjacent equipment or parts of the airplane. Electrical and mechanical cables to shock-mounted equipment should be routed and supported so that they will not be unduly stressed by motion of the equipment. In order that the occupants will not be endangered by moving equipment during minor crash landings, the equipment mounting and rack should be capable of withstanding ultimate accelerations of 9.0g in the forward direction and 1.5g in a sideward direction.

[(d) *Bonding.* Radio equipment should be bonded to the airplane in order to provide a low resistance ground circuit and to minimize radio interference from static electrical charges. Nonconducting finishes, such as paint and anodizing films should be carefully removed from the attachment surface under the bonding terminal. Bonding jumpers should be as short as practicable and be installed in such manner that the resistance of each connection does not exceed .003 ohm. Where a jumper is for radio-noise prevention only and not for current-carrying purposes, a resistance of .01 ohm is satisfactory. Aluminum alloy or tinned or cadmium-plated copper jumpers [9] should be used for bonding aluminum alloy parts and copper, brass, or bronze jumpers should be used to bond steel parts.

[(e) *Available power supply.* The radio equipment should operate satisfactorily throughout the voltage range of the airplane electrical system under taxi, takeoff, slow cruise, normal cruise, and landing operating conditions.

[(f) *Antennas, general.*[10] It is satisfactory to use one antenna for transmission and reception of communications provided such antenna is a satisfactory compromise for the frequencies to be used. In a single antenna installation of this type, the antenna should be connected to the receiver and be switched automatically to the transmitter when the microphone "push-to-talk" switch is actuated.

[(1) Fixed and trailing wire antenna installations should be tailored to fit the particular type of aircraft. Other types of antennas are

[[9] When aluminum or its alloys are in contact with most metals (exceptions are magnesium and zinc), current will flow from the aluminum to the other metals. The result will be that the aluminum will corrode. In order to minimize such corrosion, metals other than magnesium and zinc when in contact with aluminum should be cadmium-plated. Where contact between dissimilar metals cannot be avoided, care should be taken to minimize corrosion by putting a protective coating on the finished connection.]

[[10] Since all radio communication and navigation equipment necessitates the use of antennas for transmission or reception of radio frequency energy, care should be exercised in the design and installation of antennas and their coupling to the radio equipment. Consideration should be given to the fact that a transmitting antenna is a tuned circuit and that its ability to radiate radio frequency energy into space is governed primarily by the relationship between its length and the frequency of the power to be radiated. In general, the higher the frequency to be transmitted, the shorter the antenna necessary. Receiver antenna design and installation is controlled primarily by the degree of directivity necessary for the particular communication or navigation equipment.]

much more compact and may be installed as complete units on various types of aircraft.

[(2) Masts used to support a fixed-wire antenna should be as long as practicable to separate the antenna from the fuselage and/or wings in order to provide an effective antenna. Masts should be firmly attached to the airplane structure. If an antenna is attached at the trailing edge of wings and leading edge of horizontal stabilizers, the attachment should be made to lugs firmly fixed to the structure. The lugs should be welded, riveted, clamped, or bolted, whichever method is most suitable, to the structure.

[(g) *Range receiver antennas (200 to 400 kc).* A "T" or "V" type antenna should be used and mounted on the top or bottom of the airplane (see fig. 1, appendix B) with an approximate clearance of one foot from the fuselage and wings. The main leg of the "T" and each leg of the "V" antenna should be a minimum of 6 feet long.[11]

[(h) *Direction finding antennas (100 to 1750 kc).* Manual or automatic loop type antennas should be used with direction finder receivers. The loops are designed for use with a particular receiver. Connecting wires between the loops and receivers are also designed for the specific equipment. Accordingly, only components meeting the specification characteristics of the receiver manufacturer should be used.

[(1) Loops are usually enclosed in a streamlined housing for external mounting on an airplane. However, loops may be installed internally in the airplane when proper attention is given to avoiding interference from metallic structure and skin of the airplane.

[(2) The outstanding characteristic of a loop antenna is its directional sensitivity which makes it useful as an accurate navigational aid. Various things can reduce this accuracy and should be avoided. Metallic base paints should not be used on the housing. Location of the manual loop near an engine which has poor ignition shielding should be avoided since this location makes it difficult to detect the "null." Loops should not be located in positions where extensive metallic airplane structure is between them and the ground facilities. After installation, a loop should be compensated[12] to correct for any azimuth error due to its location with respect to any metal structure of the airplane.

[(i) *HF transmitting antenna—fixed.*

[(1) Two basic types of fixed antennas are used. One of these, the Hertz type, is tuned to its resonant frequency or a higher harmonically related frequency by being designed to a specific length. The lowest frequency at which this type antenna will be resonant is its fundamental frequency. At this frequency the antenna is approximately ½ wave length. This length may be determined by the formula:

$$\tfrac{1}{2} \text{ wave length (in feet)} = \frac{300{,}000{,}000 \times 3.28}{2 \times f}$$

in which 300,000,000 is the velocity of the propogation of radio waves in meters per second; 3.28 is the conversion factor from meters to feet; f is the frequency in cycles per second.

[(2) The other basic type of antenna is the Marconi. This type is readily adaptable for use on different frequencies and has one end grounded through the transmitter to the metallic aircraft structure. Thus, the aircraft acts as a part of the antenna and it is possible to use only ¼ wave length in the antenna proper. This length may be computed by the formula:

$$\tfrac{1}{4} \text{ wave length (in feet)} = \frac{300{,}000{,}000 \times 3.28}{4 \times f}.$$

An antenna of this type shorter than ¼ wave length may be tuned to resonance with the transmitter frequency by connecting an induction coil having appropriate characteristics in series with the antenna. A continuously variable coil or a stepped variable coil may be used for transmission on several different frequencies on one antenna.

[(j) *VHF transmitting antenna—"Whip" and streamlined blade.* (See fig. 2, appendix B.)

[(1) The "whip" antenna is resonant through a narrow frequency band and may be used with equipment operating in the 118 to 136 mc band. The proper length in inches should be determined from the formula $\frac{11{,}218}{4 \times f \text{ (mc)}}$. (For 125 mc the length [¼ wave length] of the antenna would be approximately 22.5 inches.)

[[11] Excessively long antennas are unsatisfactory because they are more directive in reception than relatively short ones. A trailing wire type of antenna will not prove suitable when used with a range receiver because it is highly directive; may cause a loss of distinct "cone of silence"; and changes in the airplane's direction, which may occur often in range flying, cause the antenna to "whip," resulting in breakage and loss of reception. The whipping action may also cause the course signals to shift continuously.]

[[12] Compensation of a loop for such error requires technical knowledge of the equipment and its operational use and should be accomplished by a qualified technician.]

[(2) When it is necessary to cover a broader frequency range than can be covered by the "whip" antenna, a blade type should be used, because it is resonant over a much broader frequency range. However, a broadband antenna is not as efficient as a small diameter "whip" antenna and, accordingly, should not be used with relatively low output transmitters (under 5 watts).

[(3) Antennas of both these types should be located so that there is a minimum of structure between them and the ground radio stations. Thus, the location will usually be a compromise. The antennas may be mounted on the top or bottom of the fuselage or on the cowl forward of the cockpit.

[(4) On fabric-covered aircraft, or aircraft with other types of nonmetallic skin, the manufacturer's recommendations should be followed in order to provide the necessary ground plane. An acceptable method of accomplishing this is by providing a number of metal foil strips in a radial position from the antenna base and secured under the fabric or wood skin of the aircraft as shown in figure 3, appendix B.

[(k) *VHF navigation receiving antennas.* Antennas for omnirange (VOR), and instrument landing system (ILS) localizer receivers should be located at a position on the airplane where they will have the greatest sensitivity for the desired signals and minimum response to undesired signals such as VHF energy radiated by the engine ignition system. The best location for the VOR-localizer receiving antenna on most small airplanes is over the forward part of the cabin. (See fig. 4, appendix B.) The rigid "V" type antenna should be mounted so that the apex of the V points forward and the plane of the V is level in normal flight. A list of satisfactory antenna installations on some popular small airplanes, giving distance in inches aft of the windshield, follows:

Piper	7	Swift	11
Ercoupe	26	Bellanca	4
Stinson	13	Luscombe	34
Cessna	10	Beech Bonanza	35

[(1) The most effective method of minimizing the engine ignition interference is to shield the ignition system. This involves enclosing in metal all parts of the circuits which might radiate noise. Ignition wires having a metallic-braid covering and special end connectors should be used between magnetos and spark plugs. Shielded type spark plugs should be used and a shielded metal cover for the magneto, if it is not of a shielded type, should be used. All connections in the shielding system should be tight metal-to-metal contact.

[(2) If it is not feasible to shield the engine ignition system, the engine ignition noise should be suppressed by replacing the spark plugs with resistor spark plugs of a type approved for the engine.

[(l) *Marker receiving antenna.* The marker receiver operates at a frequency of 75 mc. In order to keep to a minimum the number of antennas on an airplane, the marker receiver may utilize the same antenna as the range receiver. However, both receivers should include provisions to permit simultaneous operation without interference. A "whip" or other vertical type of antenna should not be used for marker reception since the ground facility transmits from a horizontally polarized antenna.

[(m) *Glide slope receiving antenna.* The glide slope receiver of the instrument landing system (ILS) utilizes a small, simple dipole type antenna which should be mounted at right angles to the longitudinal axis of the airplane and near the forward part of the airplane. Several types of localizer and glide slope antennas are shown in figure 5, appendix B.

[(n) *DME antenna.* The DME antenna should be mounted at an unobstructed location on the underside of the fuselage near the trailing edge of the wing, preferably at the lowest point on the aircraft when in level flight. (See fig. 6, appendix B.) During flight, the antenna should be as nearly vertical as possible. It should be mounted as far as possible from other antennas and at least 36 inches away from other obstructions. If separate antennas are used for transmission and reception, the antennas should be mounted on a line perpendicular to the longitudinal axis of the aircraft with a minimum spacing of 2 feet between antennas. Since transmission line losses are relatively high at these frequencies, the antenna connecting cables should be kept as short as possible (10 feet maximum).]

(22 F. R. 135, Jan. 5, 1937, effective Jan. 31, 1957.)

Miscellaneous Equipment; Installation

3.725 *Accessories for multiengine airplanes.* Engine driven accessories essential to the safe operation of the airplane shall be so distributed among two or more engines that the failure of any one engine will not impair the safe operation of the airplane by the malfunctioning of these accessories.

Hydraulic Systems

3.726 *General.* Hydraulic systems and elements shall be so designed as to withstand, without exceeding the yield point, any structural loads which might be imposed in addition to the hydraulic loads.

3.727 *Tests.* Hydraulic systems shall be substantiated by proof pressure tests. When proof tested, no part of the hydraulic system shall fail, malfunction, or experience a permanent set. The proof load of any system shall be 1.5 times the maximum operating pressure of that system.

3.728 *Accumulators.* Hydraulic accumulators or pressurized reservoirs shall not be installed on the engine side of the fire wall, except when they form an integral part of the engine or propeller.

Subpart G—Operating Limitations and Information

3.735 *General.* Means shall be provided to inform adequately the pilot and other appropriate crew members of all operating limitations upon which the type design is based. Any other information concerning the airplane found by the Administrator to be necessary for safety during its operation shall also be made available to the crew. (See secs. 3.755 and 3.777.)

Limitations

3.737 *Limitations.* The operating limitations specified in sections 3.738–3.750 and any similar limitations shall be established for any airplane and made available to the operator as further described in sections 3.755–3.780, unless its design is such that they are unnecessary for safe operation.

Air Speed

3.738 *Air speed.* Air-speed limitations shall be established as set forth in sections 3.739–3.743.

3.739 *Never-exceed speed* (V_{ne}). This speed shall not exceed the lesser of the following:

(a) 0.9 V_d chosen in accordance with section 3.184.

(b) 0.9 times the maximum speed demonstrated in accordance with section 3.159, but shall not be less than 0.9 times the minimum value of V_d permitted by section 3.184.

3.740 *Maximum structural cruising* (*speed* V_{no}). This operating limitation shall be:

(a) Not greater than V_c chosen in accordance with section 3.184.

(b) Not greater than 0.89 times V_{ne} established under section 3.739.

(c) Not less than the minimum V_c permitted in section 3.184.

3.741 *Maneuvering speed* (V_p). See section 3.184.

3.742 *Flaps-extended speed* (V_{fe}).

(a) This speed shall not exceed the lesser of the following:

(1) The design flap speed, V_f, chosen in accordance with section 3.190.

(2) The design flap speed chosen in accordance with section 3.223, but shall not be less than the minimum value of design flap speed permitted in sections 3.190 and 3.223.

(b) Additional combinations of flap setting, air speed, and engine power may be established, providing the structure has been proven for the corresponding design conditions.

3.743 *Minimum control speed* (V_{mc}). (See sec. 3.111.)

Power Plant

3.744 *Power plant.* The power plant limitations in sections 3.745 through 3.747 shall be established and shall not exceed the corresponding limits established as a part of the type certification of the engine and propeller installed in the airplane.

3.745 ***Take-off operation.***

(a) Maximum rotational speed (revolutions per minute).

(b) Maximum permissible manifold pressure (if applicable).

(c) The time limit upon the use of the corresponding power.

(d) Where the time limit of paragraph (c) of this section exceeds 2 minutes, the maximum allowable temperatures for cylinder head, oil, and coolant outlet if applicable.

3.746 ***Maximum continuous operation.***

(a) Maximum rotational speed (revolutions per minute).

(b) Maximum permissible manifold pressure (if applicable).

(c) Maximum allowable temperatures for cylinder head, oil, and coolant outlet if applicable.

3.747 ***Fuel octane rating.*** **The minimum octane rating of fuel required for satisfactory operation of the power plant at the limits of sections 3.745 and 3.746.**

Airplane Weight

3.748 *Airplane weight.* The airplane weight and center of gravity limitations are those required to be determined by section 3.71.

Minimum Flight Crew

3.749 *Minimum flight crew.* The minimum flight crew shall be established as that number of persons required for the safe operation of the airplane during any contact flight as determined by the availability and satisfactory operation of all necessary controls by each operator concerned.

Types of Operation

3.750 *Types of operation.* The type of operation to which the airplane is limited shall be established by the category in which it has been found eligible for certification and by the equipment installed. (See the appropriate operating parts of the Civil Air Regulations.)

Markings and Placards

3.755 *Markings and placards.*

(a) The markings and placards specified are required for all airplanes. Placards shall be displayed in a conspicuous place and both shall be such that they cannot be easily erased, disfigured, or obscured. Additional informational placards and instrument markings having a direct and important bearing on safe operation may be required by the Administrator when unusual design, operating, or handling characteristics so warrant.

(b) When an airplane is certificated in more than one category, the applicant shall select one category on which all placards and markings on the airplane shall be based. The placard and marking information for the other categories in which the airplane is certificated shall be entered in the Airplane Flight Manual. A reference to this information shall be included on a placard which shall also indicate the category on which the airplane placards and markings are based.

3.755–1 *Markings and placards for an airplane certificated in more than one category (CAA policies which apply to sec. 3.755 (b)).*

(a) The following suggestions are given to assist in making placards and markings as simple and straightforward as possible:

(1) The applicant (who may be the manufacturer or an individual operator) should select a "basic" category on which all markings and placards will be based and installed on a particular airplane. However, this does not prevent the selection of some other category as "basic" for the placarding and marking of other airplane of the same model.

(2) Placards of markings pertaining to other categories should be installed only when this can be done without confusing the placards or markings for the "basic" category. For example, previous attempts to put dual sets of markings on airspeed indicators have proven unsatisfactory. On the other hand, it may be desirable to install baggage capacity and number of persons placards which cover both normal and utility categories.

(3) A statement on the placard, required by sections 3.769 and 3.770, should refer the operator to the "Approved Airplane Flight Manual" for information on the placards and markings appropriate to the other categories in which the airplane is certificated.

(4) All placards should be arranged to present the necessary information to the pilot in as simple and practical a manner as possible. In many cases, it may be convenient to consolidate various placards.

(5) The following is an example of a possible (but not necessarily complete) form for a consolidated placard for an airplane certificated in Normal and Utility Categories, using the Normal Category as the "basic" category for purposes of placarding and marking.

> THIS AIRPLANE MUST BE OPERATED AS A NORMAL OR UTILITY CATEGORY AIRPLANE IN COMPLIANCE WITH THE APPROVED AIRPLANE FLIGHT MANUAL.
>
> All markings and placards on this airplane apply to its operation as a Normal Category Airplane. For Utility Category operations, refer to the Airplane Flight Manual.
>
> NO ACROBATIC MANEUVERS (INCLUDING SPINS) ARE APPROVED FOR NORMAL CATEGORY OPERATIONS.

(6) When the category selected for marking and placarding is the Utility Category, the appropriate placards for limiting the weight to the approved utility value should, of course, be posted. This may, for example, require placards on some of the seats, "Not to be occupied during Utility operations," and "Maximum baggage capacity during Utility Category operations, ------ pounds."

When the number of occupants permitted for the Utility Category is less than the number of seats, but the seating arrangement makes no difference, it may be more convenient to omit the seat placards and substitute a statement such as the following on the consolidated placard, "Maximum number of persons for Utility Category Operations, ------."

(7) For Utility Category maneuvering limitations, see section 3.20–2.

(Supp. 10, 16 F. R. 3292, Apr. 14, 1951.)

3.755–2 *Markings and placards for flap settings (CAA policies which apply to sec. 3.755 (a)).*

(a) *Flap settings as related to performance.* Instructions on flap settings relating to airplane performance should be included in the "performance information" section of the Manual, and should be identified with the corresponding performance data given in section 3.777–1 (g). If the applicant has demonstrated compliance with the pertinent performance requirements for a range of flap settings, the range may be given instead of a single setting. In this case, performance data should be shown for both extremes of the range, or for the critical setting within the range, plus explanation of the qualitative effect on performance of using other settings within the range.

(b) *Flap settings resulting in unsafe characteristics.* If improper setting of the flaps can result in dangerous characteristics, a suitable item should be included in the "operating limitations" section of the Flight Manual, and on a placard in view of the pilot.

Typical examples of "dangerous characteristics" would be cases in which a flap takeoff setting less than that marked on the flap indicator would cause unusual difficulty in takeoff by greatly extending the takeoff distance, or affecting controllability (e. g., porpoising, or inability to raise nose wheel). Reasonable and gradual variations in performance with change in flap setting would not be considered dangerous. Cases of obvious pilot error need not be considered such as takeoff with flaps in landing setting, provided the pertinent settings are adequately marked on the flap indicator.

(Supp. 10, 16 F. R. 3292, Apr. 14, 1951.)

Instrument Markings

3.756 *Instrument markings.* The instruments listed in sections 3.757–3.761 shall have the following limitations marked thereon. When these markings are placed on the cover glass of the instrument, adequate provision shall be made to maintain the correct alignment of the glass cover with the face of the dial. All arcs and lines shall be of sufficient width and so located as to be clearly and easily visible to the pilot.

3.757 *Air-speed indicator.*

(a) True indicated air speed shall be used:

(1) The never-exceed speed, V_{ne}—a radial red line (see sec. 3.739).

(2) The caution range—a yellow arc extending from the red line in (1) above to the upper limit of the green arc specified in (3) below.

(3) The normal operating range—a green arc with the lower limit at V_{s_1} as determined in section 3.82 with maximum weight, landing gear and wing flaps retracted, and the upper limit at the maximum structural cruising speed established in section 3.740.

(4) The flap operating range—a white arc with the lower limit at V_{s_0} as determined in section 3.82 at the maximum weight, and the upper limit at the flaps-extended speed in section 3.742.

(b) When the never-exceed and maximum structural cruising speeds vary with altitude, means shall be provided which will indicate the appropriate limitations to the pilot throughout the operating altitude range.

3.757–1 *White arc on air-speed indicator (CAA interpretations which apply to sec. 3.757 (a) (4)).* The white arc on the air-speed indicator should extend to the "basic" flaps extended speed specified in section 3.742. Additional combinations of flap setting, airspeed and power established in accordance with section 3.742 should be listed in the airplane flight manual and may be listed on a placard if the manufacturer desires.

(Supp. 10, 16 F. R. 3293, Apr. 14, 1951.)

3.758 *Magnetic direction indicator.* A placard shall be installed on or in close proximity to the magnetic direction indicator which contains the calibration of the instrument in a level flight attitude with engine(s) operating and radio receiver(s) on or off (which shall be stated). The calibration readings shall be those to known magnetic headings in not greater than 30-degree increments.

3.759 *Powerplant instruments.* All required powerplant instruments shall be marked with a red radial line at the maximum and minimum (if applicable) indications for safe operation. The normal operating ranges shall be marked with a green arc which shall not extend beyond the maximum and minimum limits for continuous operation. Take-off and precautionary ranges shall be marked with a yellow arc. Ranges of engine speed which are restricted as a result of excessive engine or propeller vibration shall be marked with a red arc.

3.759-1 *Powerplant instrument markings (CAA interpretations which apply to sec. 3.759).*

(a) Where the propeller is restricted against operation in a definite r. p. m. range, because of vibrating stress considerations, such restrictions should be indicated by a red arc on the tachometer extending from the low to the high engine r. p. m. speeds corresponding to the restricted propeller speed r. p. m. ranges. This policy follows the general practice of the regulations in prescribing the use of red markings instead of yellow markings in indicating restrictions that are more than precautionary.

(b) Tachometer dial should not be marked to indicate restricted operating range due to propeller vibratory stress considerations when this consideration applies only under certain conditions such as when landing gear is extended. It is considered satisfactory for a placard covering such restricted ranges to be provided.

(Supp. 10, 16 F. R. 3293, Apr. 14, 1951.)

3.760 *Oil quantity indicators.* Indicators shall be suitably marked in sufficient increments so that they will readily and accurately indicate the quantity of oil.

3.761 *Fuel quantity indicator.* When the unusable fuel supply for any tank exceeds 1 gallon or 5 percent of the tank capacity, whichever is greater, a red band shall be placed on the indicator extending from the calibrated zero reading (see sec. 3.437) to the lowest reading obtainable in the level flight attitude, and a suitable notation in the Airplane Flight Manual shall be provided to indicate to the flight personnel that the fuel remaining in the tank when the quantity indicator reaches zero cannot be used safely in flight. (See sec. 3.672.)

Control Markings

3.762 *General.* All cockpit controls, with the exception of the primary flight controls, shall be plainly marked as to their function and method of operation.

3.762-1 *Marking of button-type starter switches (CAA interpretations which apply to sec. 3.762).* Simple pushbutton type starter switches need not be marked to indicate method of operation.

(Supp. 10, 16 F. R. 3293, Apr. 14, 1951.)

3.763 *Aerodynamic controls.* The secondary controls shall be suitably marked to comply with sections 3.337 and 3.338.

3.764 *Power-plant fuel controls.*

(a) Controls for fuel tank selector valves shall be marked to indicate the position corresponding to each tank and to all existing cross feed positions.

(b) When more than one fuel tank is provided, and if safe operation depends upon the use of tanks in a specific sequence, the fuel tank selector controls shall be marked adjacent to or on the control to indicate to the flight personnel the order in which the tanks must be used.

(c) On multiengine airplanes, controls for engine valves shall be marked to indicate the position corresponding to each engine.

(d) The usable capacity of each tank shall be indicated adjacent to or on the fuel tank selector control.

3.765 *Accessory and auxiliary controls.*

(a) When a retractable landing gear is used, the indicator required in section 3.359 shall be marked in such a manner that the pilot can

ascertain at all times when the wheels are secured in the extreme positions.

(b) Emergency controls shall be colored red and clearly marked as to their method of operation.

Miscellaneous

3.766 ***Baggage compartments, ballast location, and special seat loading limitations.***

(a) Each baggage or cargo compartment and ballast location shall bear a placard which states the maximum allowable weight of contents and, if applicable, any special limitation of contents due to loading requirements, etc.

(b) When the maximum permissible weight to be carried in a seat is less than 170 pounds (see sec. 3.74), a placard shall be permanently attached to the seat structure which states the maximum allowable weight of occupants to be carried.

3.767 ***Fuel, oil, and coolant filler openings.*** **The following information shall be marked on or adjacent to the filler cover in each case:**

(a) The word "fuel," the minimum permissible fuel octane number for the engines installed, and the usable fuel tank capacity. (See sec. 3.437.)

(b) The word "oil" and the oil tank capacity.

(c) The name of the proper coolant fluid and the capacity of the coolant system.

3.768 ***Emergency exit placards.*** **Emergency exit placards and operating controls shall be colored red. A placard shall be located adjacent to the control(s) which clearly indicates it to be an emergency exit and describes the method of operation. (See sec. 3.387.)**

3.769 ***Approved flight maneuvers.***

(a) ***Category N.*** **A placard shall be provided in front of and in clear view of the pilot stating: "No acrobat maneuvers including spins approved."**

(b) ***Category U.*** **A placard shall be provided in clear view of the pilot stating: "Acrobatic maneuvers are limited to the following: (list approved maneuvers)."**

(c) ***Category A.*** **A placard shall be provided in clear view of the pilot which lists all approved acrobatic maneuvers and the recommended entry air speed for each. If inverted flight maneuvers are not approved, the placard shall bear a notation to this effect.**

3.770 ***Operating limitations placard.*** **A placard shall be provided in clear view of the pilot stating: "This airplane must be operated as a ---------- or ---------- category airplane in compliance with the operating limitations stated in the form of placards, markings, and manuals."**

3.771 ***Airspeed placards.*** **The following airspeed limitations shall be shown on placards in view of the pilot:**

(a) Maximum speed with landing gear extended, if the airplane is equipped with retractable landing gear.

(b) Minimum control speed with one engine inoperative, for multiengine airplanes.

(c) Rough air or maneuvering speed determined in accordance with section 3.741.

Airplane Flight Manual

3.777 ***Airplane Flight Manual.***

(a) An Airplane Flight Manual shall be furnished with each airplane, having a maximum certificated weight of more than 6,000 pounds. The portions of this document listed below shall be verified and approved by the Administrator, and shall be segregated, identified, and clearly distinguished from portions not so approved. Additional items of information having a direct and important bearing on safe operation may be required by the Administrator when unusual design, operating, or handling characteristics so warrant.

(b) For airplanes having a maximum certificated weight of 6,000 pounds or less an Airplane Flight Manual is not required; instead, the information prescribed in this part for inclusion in the Airplane Flight Manual shall be made available to the operator by the manufacturer in the form of clearly stated placards, markings, or manuals.

3.777–1 *Preparation of airplane flight manuals for airplanes in the normal, utility, and acrobatic categories (CAA policies which apply to sec. 3.777).*

(a) This section outlines an acceptable arrangement for the Airplane Flight Manual as required by section 3.777. It should be noted that the items outlined below for inclusion in the document will not all be necessary for a given airplane, and the Civil Aeronautics Administration is desirous of holding the document to the smallest practicable amount of material. Only the material required by this part should be included in the Civil Aeronautics Administration approved portion of the manual. However, if desired, the manufacturer may add other data in a distinctly separate section in the same cover. The portion of the material that is to be approved by the Civil Aeronautics Administration must be so marked and clearly separated from any other material so that no one could easily err in regard to the part that is approved.

(b) The page size for the Airplane Flight Manual will be left to the decision of the manufacturer. Some sort of a cover should be provided where more than one page is involved and should indicate the nature of the contents with the following title: "Airplane Flight Manual." Each page of the approved portion should bear the notation "CAA Approved" and the date of issuance. The material should be bound in some semipermanent fashion so that pages will not easily be lost, but should be so bound that revised pages can be inserted. In the case of small airplanes where the document consists of only one or two pages, superseding the entire document would be preferable to issuing revised pages. The aircraft specification will identify the manual by the approval date, and when different versions of the airplane (skiplanes, seaplanes, etc.) are covered in separate manuals, each will be listed. Also, the latest approved revisions will be shown.

(1) When an aircraft has tentative approval only, the following statement should appear on the inside of the front covering page of the manual:

The certificate of airworthiness issued to the aircraft described hereon, subject to the final issuance of a covering type certificate, is based upon tentative approval of aircraft of this model. Upon issuance of a covering type certificate, it may become necessary to make certain modifications or adjustments to the subject aircraft in order that the certificate of airworthiness may remain effective.

(c) The Airplane Flight Manual should contain as much of the material in paragraphs (d) through (h) of this section as is applicable to the individual model. It is suggested that the document be divided into sections as indicated in paragraphs (d) through (h) of this section. The sequence of sections and of items within sections should follow the outline in so far as is practicable. This will facilitate revising the document when an airplane is altered in the field.

(d) *Administrative section.* (This section will be unnecessary in the case of small uncomplicated airplanes where the limitations consist of only one or two pages. In such cases the data noted for inclusion on the title page can be placed at the top of the first page.)

(1) *Title page.* This page should include the manufacturer's name, airplane model and the registration number.

(2) *Table of contents.* This page will not be necessary where the document consists of only a few pages.

(3) *Log of revisions.* Should provide spaces in which to record revised pages and the date inserted. This page will not be necessary where the document is short and will be superseded completely if changes are necessary.

(e) *Limitations section*

(1) *Engine power and speed limits.* Should also list engine and propeller manufacturer and model.

(2) *Temperature and manifold pressure limits.* Include, if applicable, minimum climbing airspeeds for hot weather operation.

(3) *Fuel grade.* This item as well as subparagraphs (1) and (2) of this paragraph may, in the case of most airplanes, be covered together.

(4) *Propeller.* Should list propeller manufacturer and model.

(5) *Powerplant door and flap settings.* Pertains only when cowl flaps, cooler doors or other similar devices are installed.

(6) *Placards (powerplant only).* Should list all power-plant operating placards and explain their significance, where pertinent.

(7) *Instrument markings (power-plant instruments).* Should list all power-plant instrument markings.

(8) *Airspeed limitations.* Should include "never exceed speed," "maximum structural cruising speed," "maneuvering speed," "flaps

extended speed," and "landing gear extended speed" where applicable.

(9) *Flight load factors.* The pertinent load factors should be given in terms of accelerations.

(10) *Maximum weight.* This should list maximum weights.

(11) *C. G. Range.* The approved c. g. limits and datum should be listed in inches.

(12) *Maneuvers.* This should list the approved maneuvers with recommended entry speeds.

(13) *Placards (except power-plant placards).* Should list all flight placards and explain their significance where pertinent.

(14) *Instrument markings (except power-plant instruments).* Should list all flight instrument markings, and explain their significance. (In most cases this will involve only the airspeed indicator.)

(15) *Minimum crew.* This section should be used only when the minimum crew is more than one. Where used, the section should explain the basic duties of each crew member.

(f) *Procedures section.*

(1) *Normal operating procedures.* For the small conventional airplane where all procedures are conventional, this section will not be necessary. Only unconventional features and peculiarities of the particular airplane should be covered here, and, in the case of more complex airplanes, the following should be covered where pertinent.

(i) One engine inoperative. Applies only to multiengine types and should contain all necessary procedures for such operation.

(ii) Propeller featuring. Applies only to multiengine types equipped with feathering propellers. Should contain full instructions on feathering and unfeathering.

(iii) Circuit breakers. Should contain full information on the location and method of resetting all circuit breakers installed.

(iv) Fire procedures. Pertains only to airplanes equipped with a built-in fire extinguishingsystem. Should contain full instructions on the operation of such systems as well as associated fire protection equipment and procedures.

(v) Emergency procedures for flaps, landing gear, fuel dumping, etc.

(vi) Other special operating procedures (if any).

(g) *Performance information section.*

(1) *Takeoff data.* Should include distance to clear 50-foot obstacle, etc., at various altitudes and temperatures.

(2) *Climb data.* Should give normal rate of climb, balked landing climb (landing gear extended and wing flaps in landing position) and one-engine inoperative climb (for multiengine types) at various altitudes and temperatures.

(3) *Landing data.* Should give distance to complete landing over 50-foot obstacle and approach speed for various altitudes and temperatures.

(4) *Stalling data.* Should give stall speeds, stall warning indications and other pertinent data including stalling speeds at various angles of bank.

(h) *Weight and balance data section.* This section will not be included in the approved portion of the Airplane Flight Manual. It is the intention of the Civil Aeronautics Administration to place the responsibility for the control of weight and balance with the manufacturer and operator. The manufacturer will furnish a weight and balance report for each new airplane. The Civil Aeronautics Administration representative will not approve each individual report but will make only occasional spot checks to ascertain that the manufacturer's weight control procedure is adequate. The manufacturer will be expected to furnish complete information with the airplane, not only regarding its actual weight and balance, but also to include sketches, samples, and other data that will assist the operator in checking the balance after alterations. The Repair and Alteration Form (ACA–337) has been revised to include space for recording the new empty weight, empty weight c. g. and useful load on the form after each change. A copy of this form will be given to the owner and his file of such forms, together with the manufacturer's original data will afford the owner with a complete and up-to-date file. In cases where the permissible c. g. positions vary with gross weight, it is suggested that a note be included in the weight and balance report advising owners to contact the airplane manufacturer when any change is made to the airplane which would appreciably affect the location of the empty c. g. or location of useful load items. The manufacturer is asked to cooperate in an educational program to inform the owner of his responsibility and the means whereby he can discharge it. To this end, a statement substantially as follows should be prominently displayed in the weight and balance section:

NOTE: It is the responsibility of the airplane owner and the pilot to insure that the airplane is loaded properly. The empty weight, empty weight c. g. and useful load are noted below for this airplane as delivered from the factory. If the airplane has been altered, refer to the latest approved Repair and Alteration Form (ACA-337) for this information.

(1) *Weight limits.* Should list and explain (where necessary) the various weight limits. In the case of a small airplane, only the maximum (gross) weight would be applicable.

(2) *C. G. limits.* Approved operating c. g. range.

(3) *Empty weight c. g. limits where practicable.* This applies to the empty weight c. g. range which will automatically assure compliance with the operating c. g. limits under the most adverse loading conditions.

(4) *Empty weight and empty weight c. g. location.*

(5) *Equipment list.* All equipment included in the empty weight. Equipment items should normally be identified by the item number and name used in the Aircraft Specification.

(6) *Weight computations.* The computations necessary to determine the empty weight c. g. location and the check of forward and aft c. g. locations where applicable.

(7) *Loading schedule.* Supply where necessary.

(8) *Loading schedule instructions.* Complete instructions in the use of the loading schedule.

(9) *Unconventional airplanes.* The material in paragraph (h) of this section is believed to be complete and adequate for a conventional airplane. In the case of unconventional airplanes or airplanes with special features, the foregoing should be modified or amplified as necessary to cover the case.

(i) *Number of copies.* Three copies of the above material, less the Weight and Balance Data Section, should be submitted to the appropriate Civil Aeronautics Administration regional office by the manufacturer for an original approval. One copy will be signed by the Chief, Aircraft Division, and returned to the manufacturer. Revisions to the manual will be approved in the regional office. One copy of the Weight and Balance Data Section should be included in the manual by the manufacturer for each airplane at the time of certification.

(j) *Sample of airplane flight manual.* A sample of an Airplane Flight Manual that fulfills the requirements in the case of a small uncomplicated airplane is given in paragraph (k) of this section.

(k) *Sample sheet.*

3.777-2 *Calculated effects of temperature and altitude variations (CAA policies which apply to sec. 3.777).* See section 3.780-1.

(Supp. 10, 16 F. R. 3295, Apr. 14, 1951.)

3.777-3 *Performance data for altered airplanes of this part (CAA policies which apply to sec. 3.777).* See section 3.780-2.

(Supp. 10, 16 F. R. 3295, Apr. 14, 1951).

3.777-4 *Performance data and flight tests for ski installations on airplanes of this part (CAA policies which apply to sec. 3.777).* See section 3780-3.

(Supp. 10, 16 F. R. 3295, Apr. 14, 1951.)

3.778 *Operating limitations.*

(a) *Air-speed limitations.* Sufficient information shall be included to permit proper marking of the air-speed limitations on the indicator as required in section 3.757. It shall also include the design, maneuvering speed, and the maximum safe air speed at which the landing gear can be safely lowered. In addition to the above information, the significance of the air speed limitations and of the color coding used shall be explained.

(b) *Power-plant limitations.* Sufficient information shall be included to outline and explain all power-plant limitations (see sec. 3.744) and to permit marking the instruments as required in section 3.759.

(c) *Weight.* The following information shall be included:

(1) Maximum weight for which the airplane has been certificated,

SAMPLE SHEET

(This document must be kept in airplane at all times)

C. A. A. Approved January 1, 1947.

RONSON AIRCRAFT CO.

LOS ANGELES, MAINE

CAA Identification No. ------

RONSON 98
NORMAL AND UTILITY CATEGORIES

AIRPLANE FLIGHT MANUAL

1. LIMITATIONS

The following limitations must be observed in the operation of this airplane:

Engine: Motors model 150A.
Engine limits: For all operations—2500 rpm, 150 hp.
Fuel: 73 minimum octane aviation gasoline.
Propeller: Hamilton Standard constant speed, hub 2D30, blades 6167A-15. Pitch settings, high 29°, low 14° at 42 in. sta.
Power instrument: (a) Fuel quantity gauge: Fuel remaining in tank when indicator is in the region marked in RED cannot safely be used in flight.
(b) Oil temperature: Unsafe if indicator exceeds RED line (200° F)
(c) Tachometer: RED line at rated engine speed. DO NOT EXCEED.

		Normal category	*Utility category*
Airspeed limits (true indicated airspeed):	Never exceed	160 mph	165 mph
	Max. structural cruising	120 mph	120 mph
	Maneuvering	110 mph	110 mph
	Flaps extended	90 mph	90 mph
Flight load factors:	Max. positive load factors	3.9	4.5
	Max. negative load factor	No inverted maneuvers approved.	
Maximum weight:		2,100 lbs.	1,900 lbs.

C. G. range: (+11.0) to (+22.4)
Datum—L. E. of wing.

NOTE: It is the responsibility of the airplane owner and the pilot to insure that the airplane is properly loaded.

(a) No acrobatic maneuvers approved for Normal Category operation.
(b) The following maneuvers are approved for operation in the Utility Category only, with recommended entry speeds shown:
CHANDELLES—110 MPH TIAS LAZY EIGHTS—105 MPH TIAS
STEEP TURNS—100 MPH TIAS SPINS
STALLS (EXCEPT WHIP STALLS)

Airspeed instrument markings and their significance:
(a) Radial RED line marks the never exceed speed which is the maximum safe airspeed.
(b) YELLOW arc on indicator denotes range of speeds in which operations should be conducted with caution and only in smooth air.
(c) GREEN arc denotes normal operating speed range.
(d) WHITE arc denotes speed range in which flaps may safely be lowered.

NOTE: Maneuvers involving approach to stalling angle or full application of elevator rudder or aileron should be confined to speeds below maneuvering speed.

2. PROCEDURES

Normal operating procedures: (a) Rear seat must not be occupied when airplane is operated in the Utility Category.

3. PERFORMANCE INFORMATION

The following performance figures were obtained during Civil Aeronautics Administration type tests and may be realized under conditions indicated with the airplane and engine in good condition and with average piloting technique.

All performance is given for 2100 lbs. weight, with no wind and on level, paved runways. In using the following data, allowance for actual conditions must be made.

Item	Altitude	Outside air temperatures				
		0° F.	25° F.	50° F	75° F.	100° F.
Take-off distance (in feet): Distance required to take-off and climb 50 feet. Full throttle 80 mph climb speed. Flaps down 10°.	Sea level	625	750	900	1,000	1,075
	2,000 feet	1,125	1,275	1,475	1,600	1,700
	4,000 feet	1,625	1,875	2,050	2,200	2,550
	6,000 feet	2,100	2,300	2,500	2,700	3,000
Landing distance (in feet): Distance required to land over 50 feet obstacle and stop. Flaps full down. Approach at 75 mph.	Sea level	1,200	1,250	1,300	1,350	1,400
	2,000 feet	1,260	1,320	1,380	1,440	1,500
	4,000 feet	1,340	1,400	1,450	1,510	1,570
	6,000 feet	1,420	1,480	1,520	1,580	1,640
Normal rate of climb (ft. per min.): Full throttle. Flaps up. Air speed 80 mph.	Sea level	800	760	720	680	640
	2,000 feet	680	640	600	560	520
	4,000 feet	550	510	470	430	390
	6,000 feet	420	380	340	300	260
Balked landing climb (feet per minute). Full throttle. Flaps down. Airspeed 80 mph.	Sea level	600	570	530	500	470
	2,000 feet	490	450	420	380	350
	4,000 feet	380	340	300	270	240
	6,000 feet	260	230	190	160	120
Stalling speeds (mph). Power off	Angle of bank	0°	20°	40°	50°	60°
	Flaps up	55	57	64	70	80
	Flaps full down	50				

Following the performance information would be the section on weight and balance. The manufacturer may merely append his regular weight and balance forms if he desires).

[Supp. 10, 16 F. R. 3293, Apr. 14, 1950]

Figure 3.—Sample sheet.

(2) Airplane empty weight and center of gravity location,

(3) Useful load,

(4) The composition of the useful load, including the total weight of fuel and oil with tanks full.

(d) *Load distribution.*

(1) All authorized center of gravity limits shall be stated. If the available space for loading the airplane is adequately placarded or so arranged that any reasonable distribution of the useful load listed in weight above will not result in a center of gravity location outside of the stated limits, this section need not include any other information than the statement of center of gravity limits.

(2) In all other cases this section shall also include adequate information to indicate satisfactory loading combinations which will assure maintaining the center of gravity position within approved limits.

(e) *Maneuvers.* All authorized maneuvers and the appropriate air-speed limitations as well as all unauthorized maneuvers shall be included in accordance with the following:

(1) *Normal category.* All acrobatic maneuvers, including spins, are unauthorized. If the airplane has been demonstrated to be characteristically incapable of spinning in accordance with section 3.124 (d), a statement to this effect shall be entered here.

(2) *Utility category.* All authorized maneuvers demonstrated in the type flight tests shall be listed, together with recommended entry speeds. All other maneuvers are not approved. If the airplane has been demonstrated to be characteristically incapable of spinning in accordance with section 3.124 (d), a statement to this effect shall be entered here.

(3) *Acrobatic category.* All approved flight maneuvers demonstrated in the type flight tests shall be included, together with recommended entry speeds.

(f) *Flight load factor.* The positive limit load factors made good by the airplane's structure shall be described here in terms of accelerations.

(g) *Flight crew.* When a flight crew of more than one is required to operate the airplane safely, the number and functions of the minimum flight crew shall be included.

3.779 *Operating procedures.* This section shall contain information concerning normal and emergency procedures and other pertinent information peculiar to the airplane's operating characteristics which are necessary to safe operation.

3.780 *Performance information.*

(a) For airplanes with a maximum certificated takeoff weight of more than 6,000 lbs., information relative to the items of performance set forth in subparagraphs (1) through (5) of this paragraph shall be included:

(1) The stalling speed, V_{s_0}, at maximum weight,

(2) The stalling speed, V_{s_1}, at maximum weight and with landing gear and wing flaps retracted,

(3) The take-off distance determined in accordance with section 3.84, including the air speed at the 50-foot height, and the airplane configuration, if pertinent,

(4) The landing distance determined in accordance with section 3.86, including the airplane configuration, if pertinent,

(5) The steady rate of climb determined in accordance with section 3.85 (a), (c), and, as appropriate, (b), including the air speed, power, and airplane configuration, if pertinent.

(b) The effect of variation in paragraph (a) (2) of this section with angle of bank up to 60 degrees shall be included.

(c) The calculated approximate effect of variations in paragraph (a) (3), (4) and (5) of this section with altitude and temperature shall be included.

3.780-1 *Calculated effects of temperature and altitude variations (CAA policies which apply to sec. 3.780).* Section 3.780 requires that the calculated effects of variations in temperature and altitude on the takeoff distance (sec. 3.84 (a) (2)), the landing distance (sec. 3.86), and the steady rate of climb (sec. 3.85 (a), (b), and (c)), shall be included in the Airplane Flight Manual. The following ranges of these variables will be considered acceptable by the Administrator:

(a) The altitudes and temperatures for which performance in takeoff distance, landing distance, takeoff climb and balked landing climb shall be calculated are sea level to 7,000 feet and 0° F. to 100° F. respectively, except that take-off and landing distances for a seaplane need not show temperatures below 30° F. at altitudes above 1,000 feet.

(b) For multiengined aircraft, the climb with the critical engine inoperative shall be calculated for an altitude range of sea level to absolute ceiling and a temperature range from 60° F. below the standard temperature to 40° F. above the standard temperature at the altitude involved.

(Supp. 1, 12 F. R. 3438, May 28, 1947, as amended by Amdt. 1, 14 F. R. 36, Jan. 5, 1949.)

3.780–2 *Performance data for altered airplanes of this part (CAA policies which apply to sec. 3.780).* Performance data for altered airplanes of this part must be changed in the Airplane Flight Manual if the alteration decreases the performance below that given in the existing manual. If performance can be shown to equal or exceed original values then a statement in the manual to this effect is sufficient.

(Supp. 10, 16 F. R. 3295, Apr. 14, 1951.)

3.780–3 *Performance data and flight tests for ski installations on airplanes of this part (CAA policies which apply to sec. 3.780).*

(a) *Take-off and landing distances.* It will not be necessary, in complying with section 3.780 (a) (3) and (4), to make takeoff and landing distance tests on skiplane installations where landplane distances are given in the Airplane Flight Manual. The following, or similar, statements should be given in the performance information section of the Airplane Flight Manual.

(1) *Takeoff.* Under the most favorable conditions of smooth packed snow at temperatures approximating 32° F. the skiplane takeoff distance is approximately 10 percent greater than that shown for the landplane.

NOTE: In estimating takeoff distances for other conditions caution should be exercised in that lower temperatures or other snow conditions will usually increase these distances.

(2) *Landing.* Under the most favorable conditions of smooth packed snow at temperatures approximating 32° F. the skiplane landing distance is approximately 20 percent greater than that shown for the landplane.

NOTE: In estimating landing distances for other conditions caution should be exercised in that other temperatures or other snow conditions may either decrease or increase these distances.

(b) *Climb performance.* In cases where the landing gear is fixed (both landplane and skiplane), where the climb requirements are not critical, and the climb reduction is small (30 to 50 feet per minute), the CAA will accept a statement of the approximate reduction in climb performance placed in the Airplane Flight Manual performance information section. For larger variations in climb performance, or where the minimum requirements are critical, or where the landing gear of the landplane was retractable, appropriate climb data should be obtained to determine the changes, and new curves, tables, or a note should be incorporated in the Airplane Flight Manual.

(c) *Flight and handling tests.* At least a general flight check should be made prior to approval. This should include more than one landing to determine the ground handling characteristics as well as takeoff and landing characteristics. Note should be taken of ski angle at landing contact during tail high and tail low landings to avoid having the ski dig in or fail from localized stress. Ground control should be sufficient to satisfactorily complete a landing run with a turn off at slow speed in cases where brakes are not provided. In flight the ski should ride steady with no unusual drag and produce no unsatisfactory flight characteristics. Spin checks should be made on all aircraft in which spins are an approved maneuver. When spins are approved under section 3.124 (a), investigation with ski installations need not be made unless the spin characteristics of the type are known to be marginal.

(Supp. 10, 16 F. R. 3295, Apr. 14, 1951.)

Subpart H—Identification Data

3.791 *Identification plate.* A fireproof identification plate shall be securely attached to the structure in an accessible location where it will not likely be defaced during normal service. The identification plate shall not be

placed in a location where it might be expected to be destroyed or lost in the event of an accident. The identification plate shall contain the identification data required by section 1.50 of this subchapter.

3.792 *Airworthiness certificate number.* The identifying symbols and registration numbers shall be permanently affixed to the airplane structure in compliance with section 1.100 of this subchapter.

Appendix A

Simplified Design Load Criteria for Airplanes Having a Design Weight Equal to or Less Than 6,000 Pounds

1.0 General.

1.1 The design load criteria contained in this Appendix A have been determined by the Administrator under Section 3.171 (c) of CAR Part 3 to result in design loads not less than those prescribed in Sections 3.181 through 3.234 of Part 3 of the Civil Air Regulations. The use of these criteria is restricted to conventional single engine airplanes having a design weight equal to or less than 6000 pounds. Careful consideration has been given to the comments and suggestions received and where it was possible to do so, within the limits of Part 3, these comments and suggestions have been incorporated into this appendix.

1.2 It is the option of the designer whether or not he wishes to use this appendix. However, should he elect to use the appendix, he should use it in its entirety as a direct and equivalent substitute for Sections 3.181 through 3.234 of Part 3. Generally speaking, the light plane designer will find that the use of these simplified criteria will substantially reduce the amount of engineering work required to determine the basic design loads for his airplane under present airworthiness standards. In addition, it is easier for him to obtain simultaneous certification of his airplane in more than one category.

2.0 Design Criteria.

2.1 *Flight Envelope.* The flight envelope used is a combination of the maneuvering and gust envelopes of Part 3. For most designs only the four corners of the V-n diagram need be investigated. Provisions, however, are made for those cases in which the gust load factor at V_c is greater than the manuevering load factor. The basic positive load factors of Part 3 have been retained but it was found necessary to increase the negative load factor of Part 3 from .4 to .5 of the positive value in order to make the criteria for the several categories consistent. In addition, the right hand lower corner has been squared off to reduce the number of design conditions.

2.2 *Minimum Design Air Speeds.* The minimum design speeds are almost exactly the same as the minimum design speeds now required by Part 3, with the exception of $V_{C\ min}$. It was necessary in this latter case to increase the minimum design speed 0%, 7.5% and 13.5%, for the N. U and A categories respectively, over the Part 3 minimums. $V_{C\ min}$, however need not exceed $.9V_h$ actually obtained at sea level. $V_{F\ min}$ is equal to 1.4 V_s where V_s was computed using $C_L=1.35$.

2.3 *Control Surface Loads.*

2.30 *Horizontal Tail.* The horizontal tail control surface design loads are determined primarily by the down gust load requirements of Part 3 for the normal category. A study was made of present day airplanes to determine whether or not the up loads on the horizontal tail could be reduced. The magnitude of the loads obtained for the Checked Maneuver Condition (Section 3.216 (c) of Part 3) indicated that in some instances it would be unconservative to reduce the up loads below the down loads, therefore the same design curve is used for the up and down loads.

2.31 *Vertical Tail.* The vertical tail design loads are based on the gust requirements and curve A of Figure 3-3 (b), of Part 3. The gust criteria used in developing the vertical tail loading curve is the same as Part 3 gust criteria corrected for an aspect ratio of 2.0. The selection of an aspect ratio=2.0 is well substantiated by referring to the basic data used in present day design of light airplanes, also by the fact that AR=2.0 was the minimum in

Part 04. The gust alleviation factor, K, was conservatively selected as being equal to 1.2.

2.32 *Aileron.* The aileron design loads are exactly the same as those given by curve B of Figure 3–3 (b) in Part 3.

2.33 *Flaps and Tab.* Since Part 3 does not contain empirical curves for flap and tab design, these curves were developed for this Appendix by selecting a value of $C_{N_F}=1.6$ and $V_F=12.5\sqrt{n\ w/s}$ for the flap and $C_{N_T}=.80$ and $V_C=19.5\sqrt{n\ w/s}$ for the tab. These values of C_N were selected as being reasonable and conservative, based on data from NACA Technical Reports 360, 498, 571, 574, and 633.

2.4 *Center of Gravity.* Except as noted in Section 5.12 all of the loadings specified in this Appendix are independent of the center of gravity position of the aircraft. A c. g. range is needed, however, to establish operating limitations in accordance with Section 3.778 (d) of Part 3 and therefore should be selected by the designer.

3.0 Definitions.

3.1 Except as noted below, the nomenclature and symbols used in this Appendix are the same as the corresponding nomenclature and symbols used in Part 3.

n_1=Airplane Positive Maneuvering Limit Load Factor

n_2=Airplane Negative Maneuvering Limit Load Factor

n_3=Airplane Positive 30 fps Gust Limit Load Factor at V_C

n_4=Airplane Negative 30 fps Gust Limit Load Factor at V_C

n_{flap}=Airplane Positive Limit Load Factor with Flaps fully Extended at V_F

*$V_{F\ min}$=Minimum Design Flap Speed= $12.5\sqrt{n_1\ W/S}$

*$V_{P\ min}$=Minimum Design Maneuvering Speed=$17.0\sqrt{n_1\ W/S}$

*$V_{C\ min}$=Minimum Design Cruising Speed $=19.5\sqrt{n_1\ W/S}$

*$V_{D\ min}$=Minimum Design Dive Speed= $27.3\sqrt{n_1\ W/S}$

*Also see Section 5.3

4.0 Certification in More than one Category.

The criteria in this Appendix permit simultaneous certification in more than one category (N, U or A). When certification in more than one category is desired, the design category weights should be selected such that the term "n_1W" is constant for all categories or is greater for one desired category than others. The wings and control surfaces (including wing flaps and tabs) need be investigated only for the maximum value of "n_1W" or the category corresponding to the maximum design weight in the event "n_1W" is constant. If the acrobatic category is one of the categories selected, a special unsymmetrical flight load investigation in accordance with Sections 6.31 and 7.31 should be completed. The wing, wing carry-through and the horizontal tail structure should be checked for this condition. The basic fuselage structure need be investigated only for the highest load factor design category. The local supporting structure for dead weight items need be designed only for the highest load factor imposed when the particular item is installed in the airplane. The engine mount, however, must be substantiated for a higher side load factor when certification in the acrobatic category is desired than is required for certification in the normal and utility categories. The landing gear and the airplane as a whole under landing loads, need only be investigated for the category corresponding to the maximum design weight. These simplifications apply in general to single engine aircraft of conventional type for which experience is available, and the Administrator reserves the right to require additional investigations for aircraft incorporating unusual design features.

5.0 Flight Loads.

5.1 *General.* The flight loads may be considered independent of altitude and, except for the local supporting structure for dead weight items, only the maximum design weight conditions need be investigated. Values of n_1, n_2, n_3, n_4 should be determined from Table 1 and Figures **[6]** and **[7]** for the particular *maximum design weights* appropriate to the category or categories for which approval is desired.

5.10 Values of n_3 and n_4 corresponding to the *minimum flying weight* should also be determined using Figures **[6]** and **[7]** and if these load factors are greater than the load factors at the design weight, the supporting structure for dead weight items should be substantiated for the resulting higher load factors.

5.11 In all cases the loads and loading conditions specified in Sections 5.2 through 5.4 are the minimum for which strength should be provided in the structure.

5.12 The specified wing and tail loadings are independent of center of gravity range. The designer, however, should select a c. g. range and the basic fuselage structure should be investigated for the most adverse dead weight loading conditions corresponding to the c. g. range selected.

5.2 *Airplane Equilibrium.* Vertical wing loads may be found directly from vertical airplane loads by multiplying the airplane loads (as determined from Sections 6.2 and 6.3) by a factor of 1.05 for the positive flight conditions and 1.0 for the negative. It should be noted that the vertical wing load is considered to be the wing load vertical to the relative wind. This load, depending on the maximum high angle of attack will have a chordwise component may be as much as 25% of the vertical wing lead. This chordwise load should be taken into consideration.

5.3 *Minimum Design Air Speeds.* The minimum design airspeeds may be chosen by the designer except that they should not be less than the minimum speeds found using Figure [4]. In addition, $V_{C\min}$ need not exceed $.9V_h$ actually obtained at sea level for the lowest design weight category for which certification is desired. For purposes of computing these minimum design speeds, n_1 should not be less than 3.8.

5.4 *Flight Load Factor.* The limit flight load factors specified in Table 1, represent the acceleration component in terms of the gravitational constant, g, normal to the assumed longitudinal axes of the airplane, and equal in magnitude and opposite in direction to the airplane inertia load factor at the center of gravity.

6.0 Flight Conditions

6.1 *General.* The design conditions specified in Sections 6.2 and 6.3 are intended to provide strength for all conditions of speed and load factor on or within the boundary of a V-n diagram for the aircraft similar to the one shown in Figure [5]. This V-n diagram should also be used in determining the airplane structural operating limitations as specified in Sections 3.735 through 3.743 and Section 3.748 of Part 3.

6.2 *Symmetrical Flight Conditions.*

6.20 The airplane should be designed for at least the four Basic Flight Conditions, "A", "D", "E", and "G" as noted on the flight envelope, Figure [5].

6.201 The design limit flignt load factors, corresponding to conditions "D" and "E" should be at least as great as those specified in Table 1 and Figure [5] and the design speed for these conditions should be at least equal to the value of V_D found from Figure 1.

6.202 For conditions "A" and "G" the load factors should correspond to those specified in Table 1 and the design speeds should be those computed using these load factors with the maximum static lift coefficient (C_{N_A}) determined by the designer. In the absence of more precise computations, these latter conditions may be based on a value of $C_{N_A} = \pm 1.35$. The design speed for condition "A" may be a speed less than $V_{P\min}$.

6.203 Conditions "C" and "F" need be investigated only when n_3 W/S or n_4 W/S are greater than n_1 W/S or n_2 W/S respectively (see Figures [6] and [7].

6.21 When flaps or similar high lift devices intended for use at the relatively low air speeds of approach, landing and take-off are installed, the airplane should be designed for the two flight conditions corresponding to the values of limit flap-down factors specified in Table 1 with the flaps fully extended at not less than the design flap speed $V_{F_{\min}}$ from Figure [4].

6.3 *Unsymmetrical Flight Conditions.* The affected structure as noted, should be designed for the unsymmetrical loadings specified in Sections 6.30 through 6.32.

6.30 The aft fuselage to wing attachment should be designed for the critical vertical surface load from Sections 7.30 and 7.31.

6.31 The wing and wing carry-through structure should be designed for 100% of Condition "A" loading on one side of the plane of symmetry and 70% on the opposite side for certification in the normal and utility categories or 60% on the opposite side for certification in the acrobatic category.

6.32 The wing and wing carry-through structure should be designed for the loads resulting from a combination of 75% of the positive maneuvering wing loading on both sides of the plane of symmetry combined with the maximum wing torsion resulting from aileron displacement. The effect of aileron displacement on wing torsion at V_C or V_a using the basic airfoil moment coefficient modified over the aileron portion of the span, is computed as follows:

$$C_m = C_m + .01\delta_u \text{ (up aileron side)}$$
wing basic airfoil
$$C_m = C_m - .01\ \delta_d \text{ (down aileron side)}$$
wing basic airfoil

δ_u is the up aileron deflection and δ_d is the down aileron deflection. The sum of $\delta_u + \delta_d = \Delta$ critical. The method of computing Δ critical is shown below.

(1) compute $\Delta_a = \frac{V_p}{V_c} \times \Delta_p$

and $\Delta_b = 0.5\frac{V_p}{V_d} \times \Delta_p$

where Δ_p = the maximum total deflection (sum of both aileron deflections) at V_p. V_p, V_c, and V_d are described in Section 5.3.

(2) Determine K from the following formula:

$$K = \frac{(C_m - .01\delta_b)V_d^2}{(C_m - .01\delta_a)V_c^2}$$

Where δ_a is the down aileron deflection corresponding to Δ_a and δ_b is the down aileron deflection corresponding to Δ_b as computed in step (1).

If K is less than 1.0, Δ_a is Δ critical and should be used to determine δ_u and δ_d. In this case V_c is the critical speed to be used in computing the wing torsion loads over the aileron span.

If K is equal to or greater than 1.0, Δ_b is Δ critical and should be used to determine δ_u and δ_d. In this case V_d is the critical speed to be used in computing the wing torsion loads over the aileron span.

6.4 *Supplementary Conditions.* At least the conditions specified in Sections 6.41 and 6.42 should be investigated.

6.40 *Special Conditions for Rear Lift Truss.* In lieu of an investigation of condition G, Figure [5], the special condition specified in Section 3.194 of Part 3 may be investigated. In such event and if certification in more than one category (see Section 4.0) is desired, the value of W/S used in the formula appearing in Section 3.194 of Part 3 should be that for the category corresponding to the Maximum gross weight.

6.41 *Engine Torque Effects.* Engine mounts and their supporting structure should be designed for the maximum limit torque corresponding to METO power and propeller speed, acting simultaneously with the limit loads resulting from the maximum positive maneuvering flight load factor n_1. The limit torque should be obtained by multiplying the mean torque by a factor of 1.33 in the case of engines having five or more cylinders. For 4, 3, and 2 cylinder engines the factor should be 2, 3, and 4 respectively.

6.42 *Side Load on Engine Mount.* Engine mounts and their supporting structure should be designed for the loads resulting from a lateral limit load factor not less than 1.47 for N & U categories and 2.0 for the acrobatic category.

7.0 Control Surface Loads

7.1 *General.* Control surface loads should be determined using the criteria of Section 7.2 and within the simplified loadings of Section 7.3.

7.2 *Pilot Effort.* In the control surface loading conditions described in Sections 7.3. through 7.5, the airloads on the movable surfaces and the corresponding deflections need not exceed those which could be obtained in flight by employing the maximum pilot control forces specified in Figure 3-11 of Part 3. In cases where the surface loads are limited on the basis of maximum pilot effort, the tabs should either be considered to be deflected to their maximum travel in the direction which would assist the pilot or the deflection should correspond to the maximum expected degree of "out of trim" at the speed for the condition under consideration. The tab load, however, need not exceed the value specified in Table 2.

7.3 *Surface Loading Conditions.*

7.30 Simplified limit surface loadings and distributions for the horizontal tail, vertical tail, aileron, wing flaps and trim tabs are specified in Table 2 and Figures [8] and [9]. Where more than one distribution is given, each distribution should be investigated.

7.31 When certification in the Acrobatic category is desired the horizontal tail shall be investigated for an unsymmetrical load of 100% $\overline{w}$ on one side of the airplane center line and 50% on the other side of the airplane center line.

7.4 *Outboard Fins.* See Section 3.221 of Part 3.

7.5 *Special Devices.* See Section 3.225 of Part 3.

8.0 Control System Loads

8.1 *Primary Flight Controls and Systems.*

8.10 Flight control systems and supporting structures should be designed for loads corresponding to 125 percent of the computed hinge moments of the movable control surface in the conditions prescribed in Section 7.0 subject to the following maxima and minima:

8.101 The system limit loads need not exceed those which could be produced by the pilot and automatic devices operating the controls.

8.102 The loads should in any case be sufficient to provide a rugged system for service use, including consideration of jamming, ground gusts, taxying tail to wind, control inertia, and friction.

8.11 Acceptable maximum and minimum pilot loads for elevator, aileron, and rudder controls are shown in Figure 3–11 of Part 3. These pilot loads should be assumed to act at the appropriate control grips or pads in a manner simulating flight conditions and to be reacted at the attachments of the control system to the control surface horn.

8.2 *Dual Controls.* When dual controls are provided the systems should be designed for the pilots operating in opposition, using individual pilot loads equal to 75 percent of those obtained in accordance with Section 8.1 except that the individual pilot loads should not be less than the minimum loads shown in Figure 3–11, of Part 3.

8.3 *Ground Gust Conditions.* See Section 3.233 of Part 3.

8.4 *Secondary Controls and Systems.* See Section 3.234 of Part 3.

of symmetry and 70% on the opposite side for certification in the normal and utility categories or 60% on the opposite side for certification in the acrobatic category.

6.32 The wing and wing carry-through structure should be designed for the loads resulting from a combination of 75% of the positive maneuvering wing loading on both sides of the plane of symmetry combined with the maximum wing torsion resulting from aileron displacement. The effect of aileron displacement on wing torsion at V_C or V_α using the basic airfoil moment coefficient modified over the aileron portion of the span, is computed as follows:

$$C_m = C_m + .01\delta_u \text{ (up aileron side)}$$
wing basic airfoil
$$C_m = C_m - .01\ \delta_d \text{ (down aileron side)}$$
wing basic airfoil

δ_u is the up aileron deflection and δ_d is the down aileron deflection. The sum of $\delta_u + \delta_d = \Delta$ critical. The method of computing Δ critical is shown below.

(1) compute $\Delta_a = \frac{V_p}{V_c} \times \Delta_p$

and $\Delta_b = 0.5\frac{V_p}{V_d} \times \Delta_p$

where Δ_p = the maximum total deflection (sum of both aileron deflections) at V_p. V_p, V_c, and V_d are described in Section 5.3.

(2) Determine K from the following formula:

$$K = \frac{(C_m - .01\delta_b)V_d^2}{(C_m - .01\delta_a)V_c^2}$$

Where δ_a is the down aileron deflection corresponding to Δ_a and δ_b is the down aileron deflection corresponding to Δ_b as computed in step (1).

If K is less than 1.0, Δ_a is Δ critical and should be used to determine δ_u and δ_d. In this case V_c is the critical speed to be used in computing the wing torsion loads over the aileron span.

If K is equal to or greater than 1.0, Δ_b is Δ critical and should be used to determine δ_u and δ_d. In this case V_d is the critical speed to be used in computing the wing torsion loads over the aileron span.

6.4 *Supplementary Conditions.* At least the conditions specified in Sections 6.41 and 6.42 should be investigated.

6.40 *Special Conditions for Rear Lift Truss.* In lieu of an investigation of condition G, Figure [5], the special condition specified in Section 3.194 of Part 3 may be investigated. In such event and if certification in more than one category (see Section 4.0) is desired, the value of W/S used in the formula appearing in Section 3.194 of Part 3 should be that for the category corresponding to the Maximum gross weight.

6.41 *Engine Torque Effects.* Engine mounts and their supporting structure should be designed for the maximum limit torque corresponding to METO power and propeller speed, acting simultaneously with the limit loads resulting from the maximum positive maneuvering flight load factor n_1. The limit torque should be obtained by multiplying the mean torque by a factor of 1.33 in the case of engines having five or more cylinders. For 4, 3, and 2 cylinder engines the factor should be 2, 3, and 4 respectively.

6.42 *Side Load on Engine Mount.* Engine mounts and their supporting structure should be designed for the loads resulting from a lateral limit load factor not less than 1.47 for N & U categories and 2.0 for the acrobatic category.

7.0 Control Surface Loads

7.1 *General.* Control surface loads should be determined using the criteria of Section 7.2 and within the simplified loadings of Section 7.3.

7.2 *Pilot Effort.* In the control surface loading conditions described in Sections 7.3. through 7.5, the airloads on the movable surfaces and the corresponding deflections need not exceed those which could be obtained in flight by employing the maximum pilot control forces specified in Figure 3–11 of Part 3. In cases where the surface loads are limited on the basis of maximum pilot effort, the tabs should either be considered to be deflected to their maximum travel in the direction which would assist the pilot or the deflection should correspond to the maximum expected degree of "out of trim" at the speed for the condition under consideration. The tab load, however, need not exceed the value specified in Table 2.

7.3 *Surface Loading Conditions.*

7.30 Simplified limit surface loadings and distributions for the horizontal tail, vertical tail, aileron, wing flaps and trim tabs are specified in Table 2 and Figures [8] and [9]. Where more than one distribution is given, each distribution should be investigated.

7.31 When certification in the Acrobatic category is desired the horizontal tail shall be investigated for an unsymmetrical load of 100% $\overline{w}$ on one side of the airplane center line and 50% on the other side of the airplane center line.

7.4 *Outboard Fins.* See Section 3.221 of Part 3.

7.5 *Special Devices.* See Section 3.225 of Part 3.

8.0 Control System Loads

8.1 *Primary Flight Controls and Systems.*

8.10 Flight control systems and supporting structures should be designed for loads corresponding to 125 percent of the computed hinge moments of the movable control surface in the conditions prescribed in Section 7.0 subject to the following maxima and minima:

8.101 The system limit loads need not exceed those which could be produced by the pilot and automatic devices operating the controls.

8.102 The loads should in any case be sufficient to provide a rugged system for service use, including consideration of jamming, ground gusts, taxying tail to wind, control inertia, and friction.

8.11 Acceptable maximum and minimum pilot loads for elevator, aileron, and rudder controls are shown in Figure 3-11 of Part 3. These pilot loads should be assumed to act at the appropriate control grips or pads in a manner simulating flight conditions and to be reacted at the attachments of the control system to the control surface horn.

8.2 *Dual Controls.* When dual controls are provided the systems should be designed for the pilots operating in opposition, using individual pilot loads equal to 75 percent of those obtained in accordance with Section 8.1 except that the individual pilot loads should not be less than the minimum loads shown in Figure 3-11, of Part 3.

8.3 *Ground Gust Conditions.* See Section 3.233 of Part 3.

8.4 *Secondary Controls and Systems.* See Section 3.234 of Part 3.

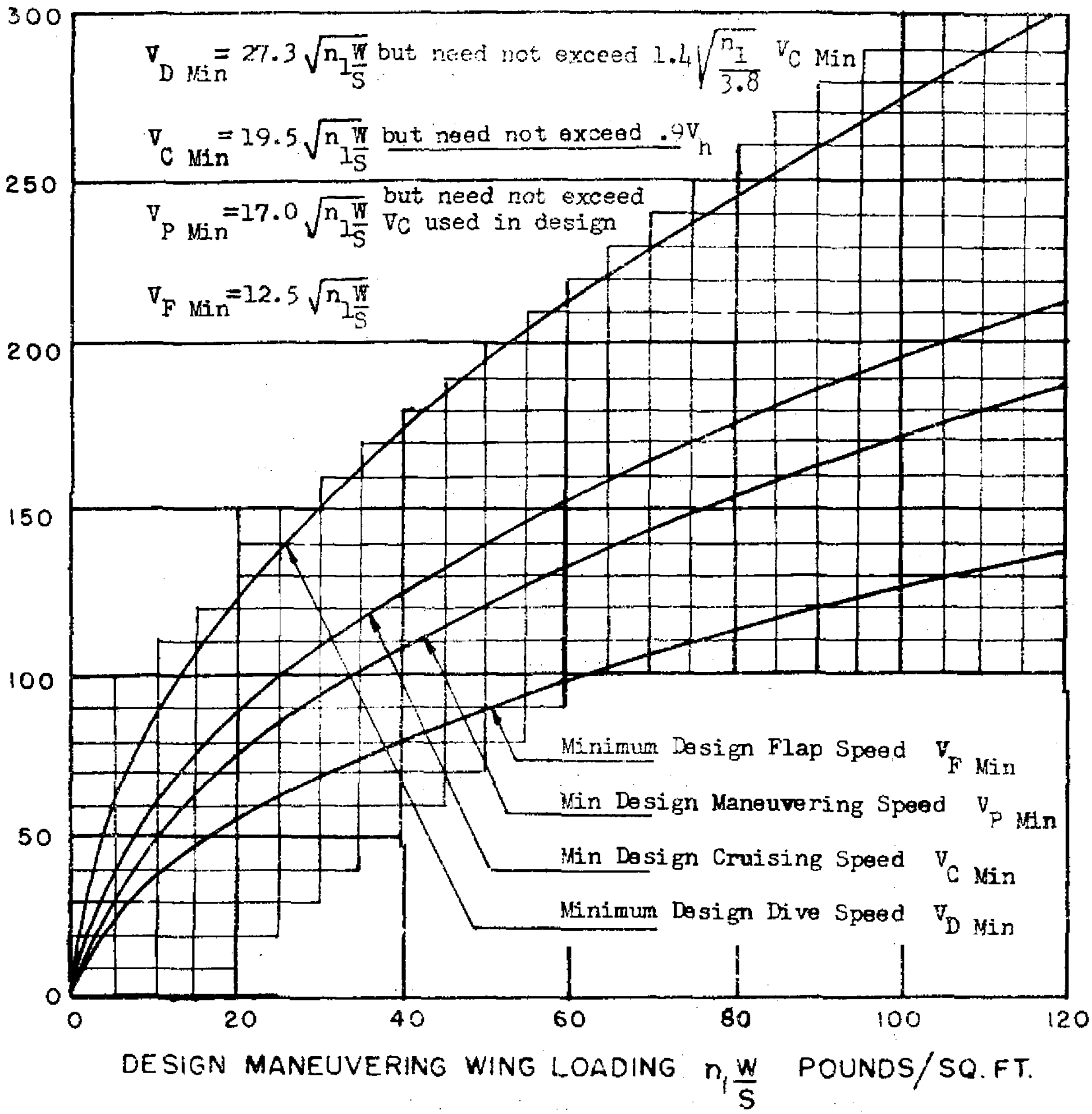


Figure 4.—Minimum design air speeds.

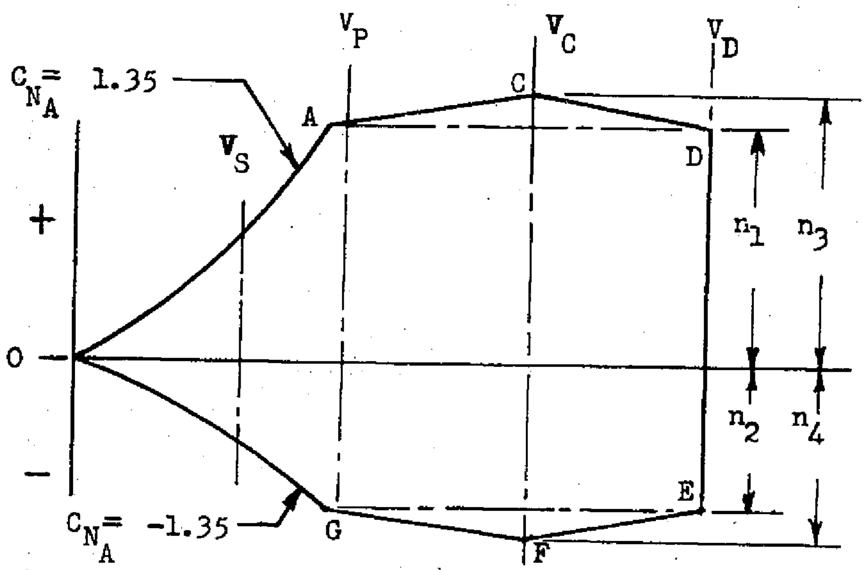


Figure 5.—(V-n) Diagram (flight envelope).

Notes: 1. Conditions "C" or "F" need be investigated only when $n_3 \frac{W}{S}$ or $n_4 \frac{W}{S}$ are greater than $n_1 \frac{W}{S}$, or $n_2 \frac{W}{S}$, respectively.

2. Condition "G" *need not* be investigated, when the supplementary condition specified in CAR 3.194 is investigated.

LIMIT FLIGHT LOAD FACTORS					
			N	U	A
FLIGHT Load Factors	Flaps up	n_1	3.8*	4.4	6.0
		n_2	$-0.5n_1$		
		n_3	Find n_3 from Fig. [6]		
		n_4	Find n_4 from Fig. [7]		
	Flaps Down	n_{flap}	$0.5n_1$		
		n_{flap}	Zero**		

Table 1.—Limit flight load factors.

***3.5 for spin-proof airplanes**

****Vertical wing load may be assumed equal to zero and only the flap portion of the wing need be checked for this condition.**

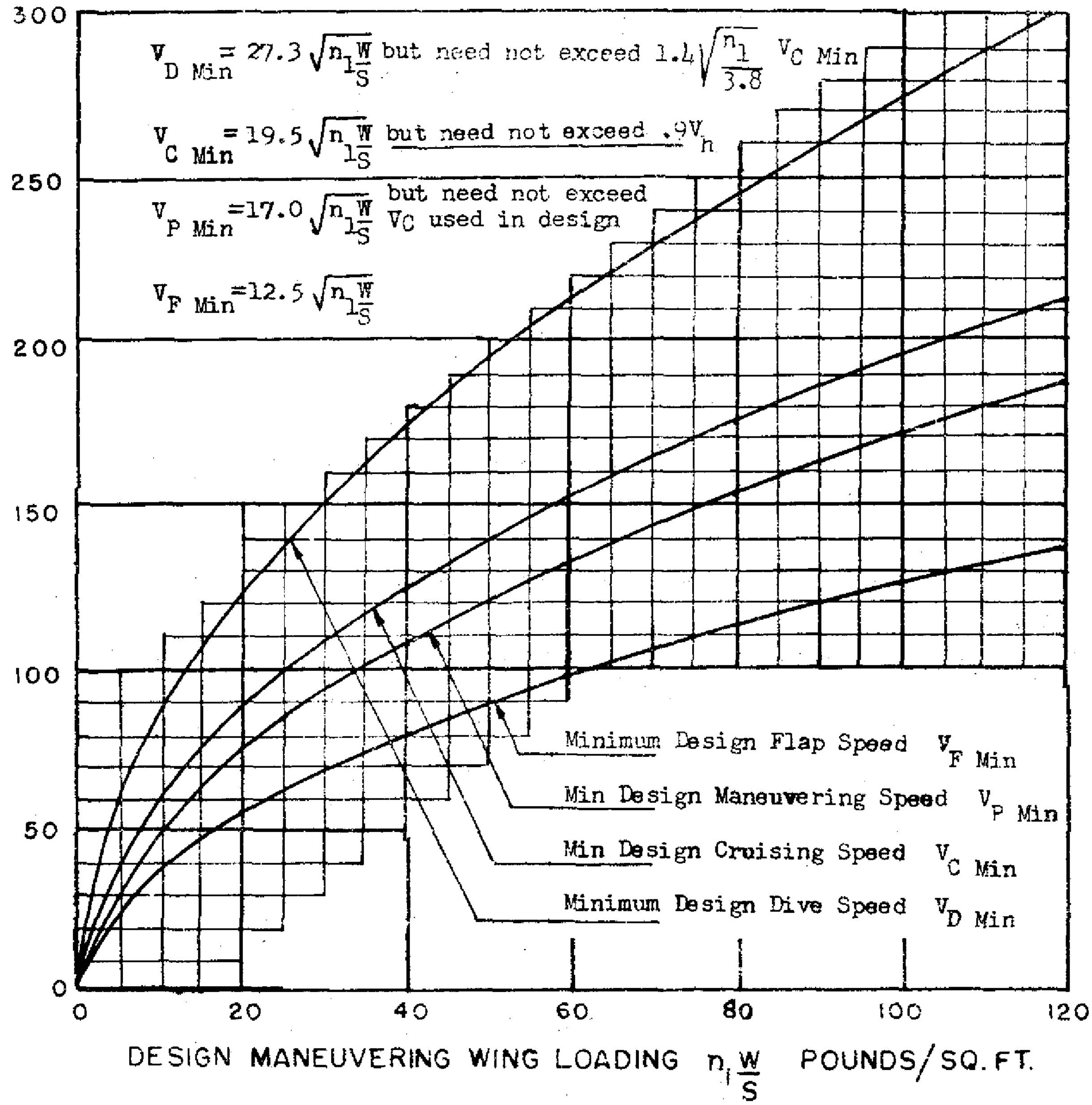


Figure 4.—Minimum design air speeds.

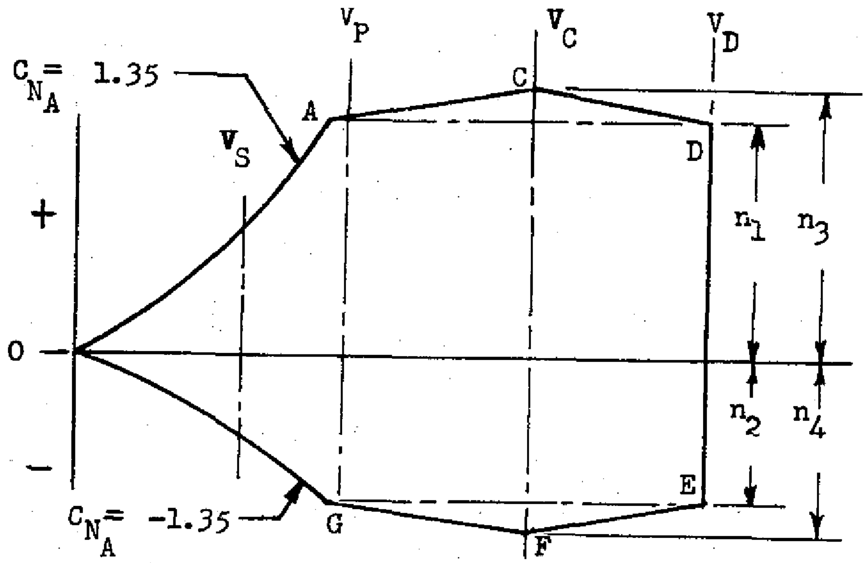


Figure 5.—(V-n) Diagram (flight envelope).

Notes: 1. Conditions "C" or "F" need be investigated only when $n_3 \frac{W}{S}$ or $n_4 \frac{W}{S}$ are greater than $n_1 \frac{W}{S}$, or $n_2 \frac{W}{S}$, respectively.

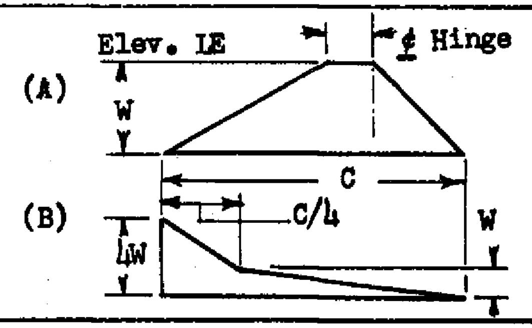
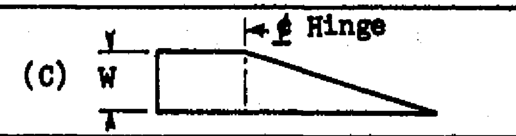
2. Condition "G" *need not* be investigated, when the supplementary condition specified in CAR 3.194 is investigated.

LIMIT FLIGHT LOAD FACTORS					
			N	U	A
FLIGHT Load Factors	Flaps up	n_1	3.8*	4.4	6.0
		n_2	$-0.5n_1$		
		n_3	Find n_3 from Fig. [6]		
		n_4	Find n_4 from Fig. [7]		
	Flaps Down	n_{flap}	$0.5n_1$		
		n_{flap}	Zero**		

Table 1.—Limit flight load factors.

*3.5 for spin-proof airplanes

**Vertical wing load may be assumed equal to zero and only the flap portion of the wing need be checked for this condition.

AVERAGE LIMIT CONTROL SURFACE LOADING			
SURFACE	DIRECTION OF LOADING	MAGNITUDE OF LOADING	CHORDWISE DISTRIBUTION
HORIZONTAL TAIL I	a) Up and Down	Figure [8] Curve (2)	
	b) Unsymmetrical loading (Up and Down)	100% $\bar{W}$ on one side airplane ¢. 65% $\bar{W}$ on other side of airplane ¢ for N and U categories. For A category see 7.3	
VERTICAL TAIL II	a) Right and Left	Figure [8] Curve (1)	Same as (A) above
	b) Right and Left	Figure [8] Curve (1)	Same as (B) above
AILERON III	a) Up and Down	Figure [9] Curve (5)	
WING FLAP IV	a) Up	Figure [9] Curve (4)	
	b) Down	.25 x Up Load (a)	
TRIM TAB V	a) Up and Down	Figure [9] Curve (3)	Same as (D) above

NOTE: The surface loadings I, II, III, and V above are based on speeds V_p min and V_c min. The loading of IV is based on V_f min. If values of speeds greater than these minimums are selected for design the appropriate surface loadings shall be multiplied by the ratio $\left[\frac{V_{selected}}{V_{minimum}}\right]^2$. For conditions I, II, III and V the multiplying factor used shall be the higher of $\left[\frac{V_{p\ sel.}}{V_{p\ min.}}\right]^2$ or $\left[\frac{V_{c\ sel.}}{V_{c\ min.}}\right]^2$

Table 2.—Average limit control surface loading.

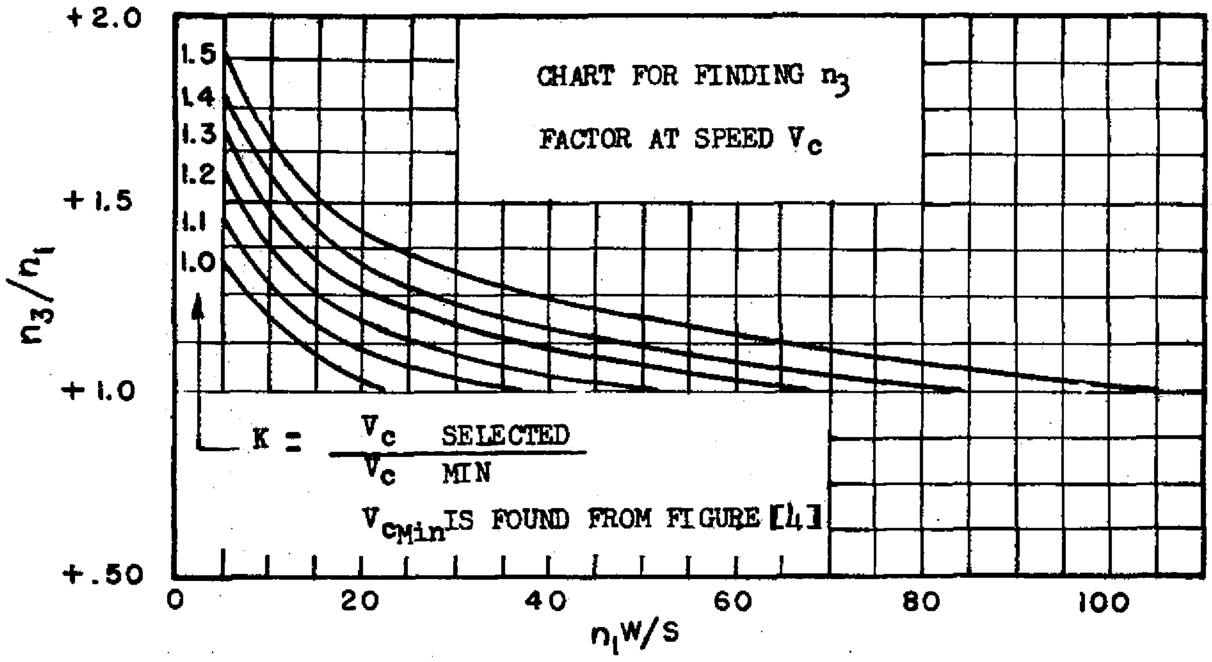


Figure 6.—Chart for finding n_3 factor at speed V_c.

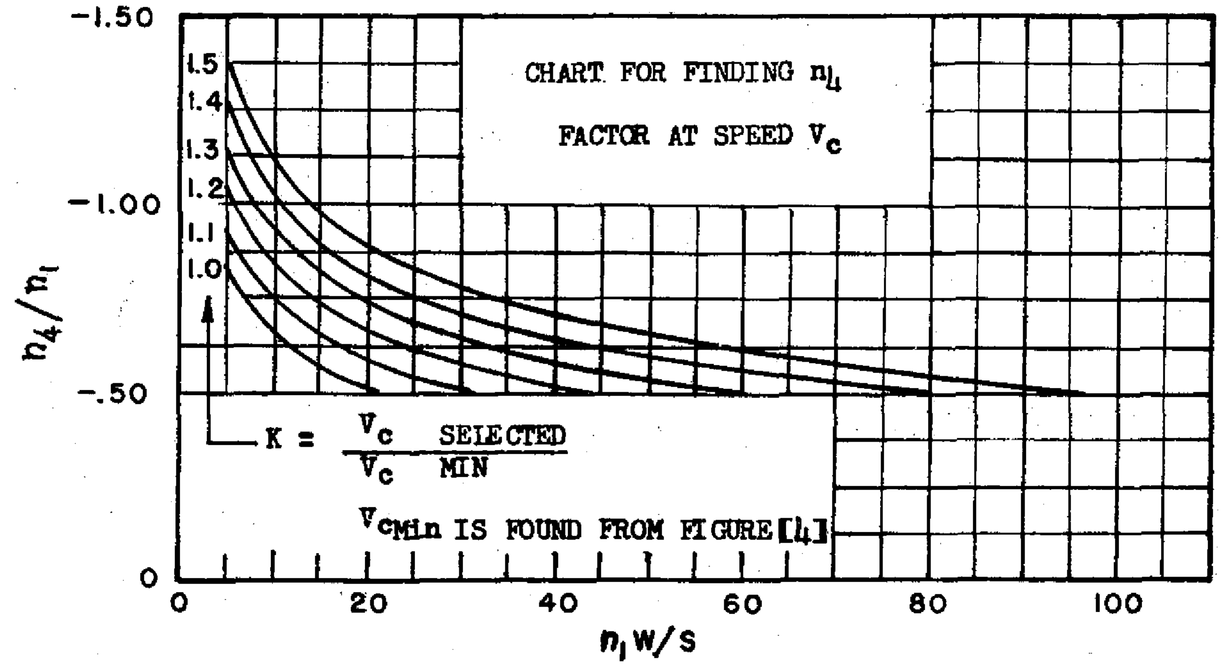


Figure 7.—Chart for find n_4 factor at speed V_c.

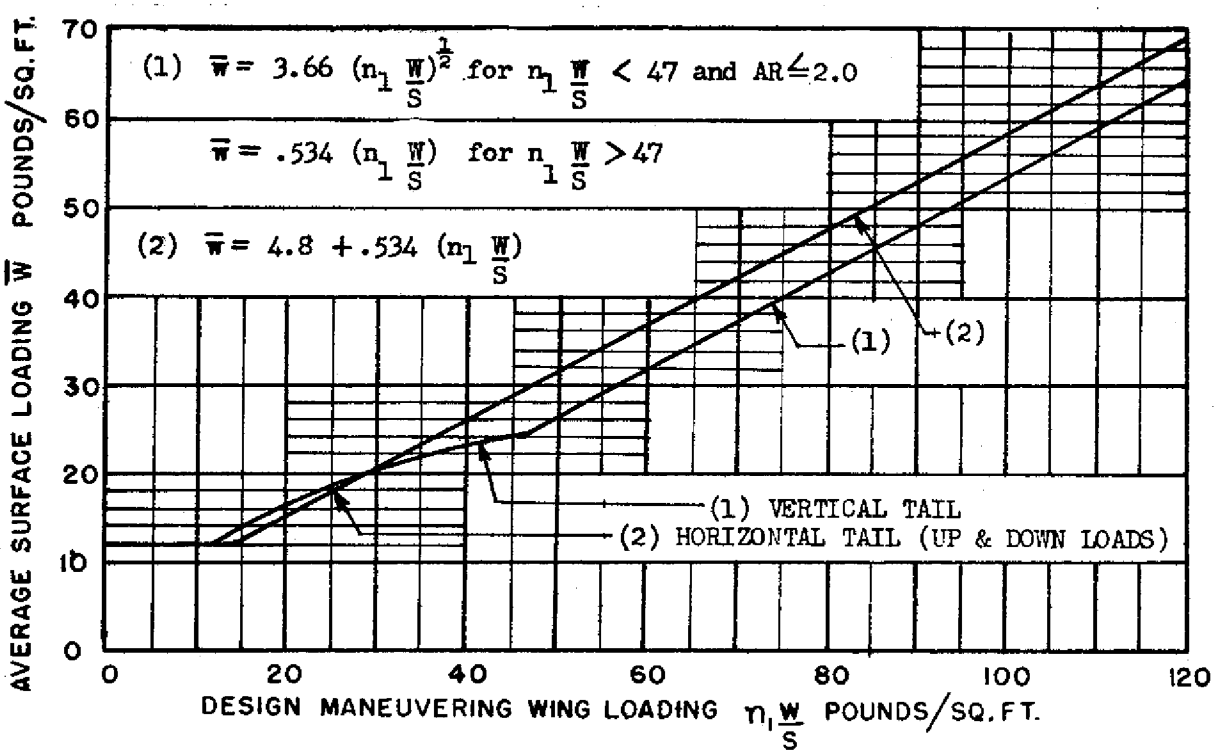


Figure 8.—Average limit control surface loading.

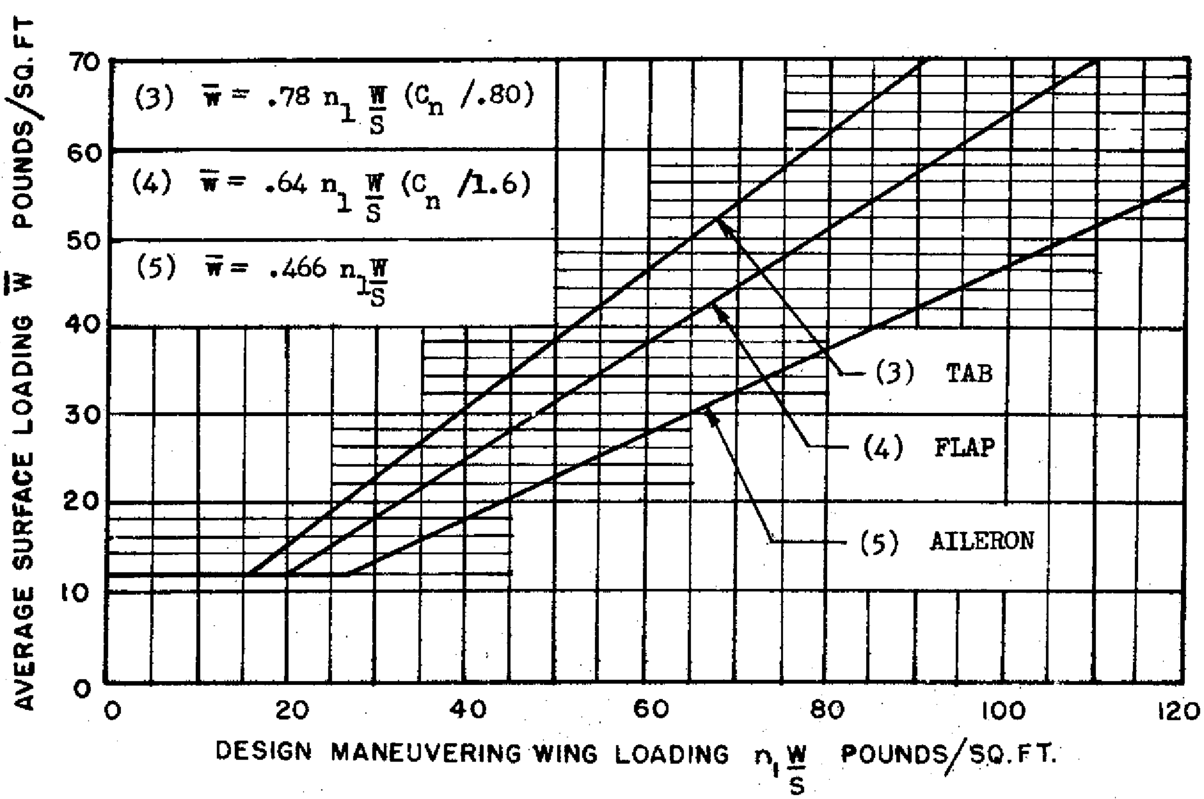
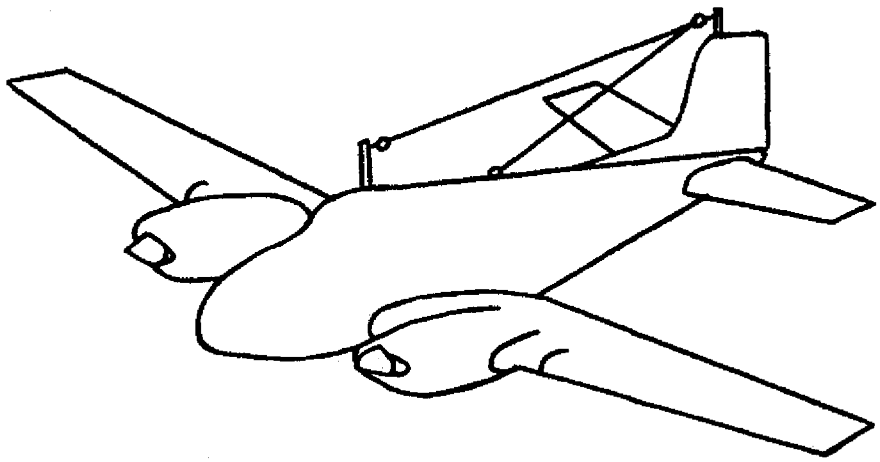
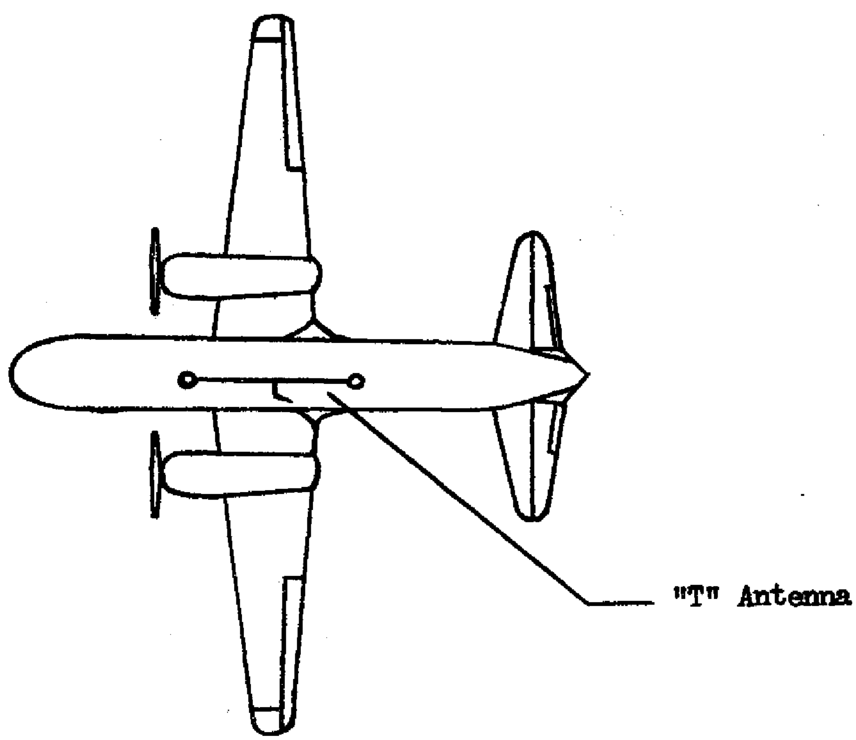


Figure 9.—Average limit control surface loading.

Appendix B



VERTICAL "V" ANTENNA



"T" OR "V" TYPE ANTENNA

Figure 1

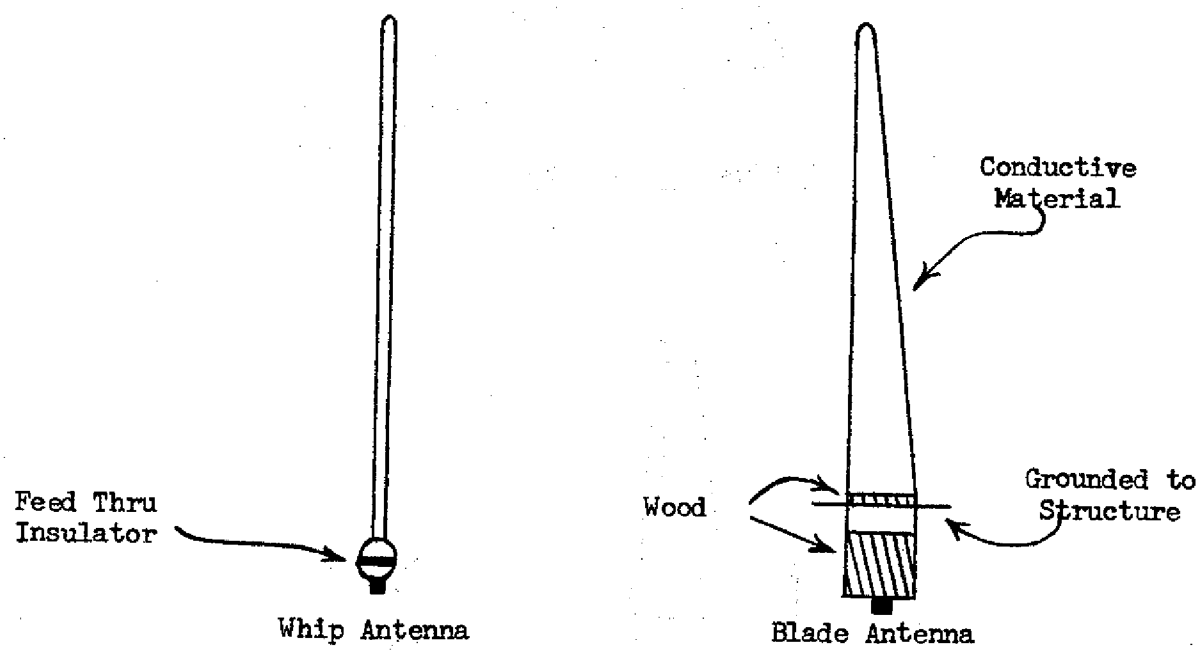

Feed Thru
Insulator
Whip Antenna
Conductive
Material
Wood
Grounded to
Structure
Blade Antenna

Figure 2

ANTENNA GROUND PLANE
FOR
NON-METALLIC AIRPLANES

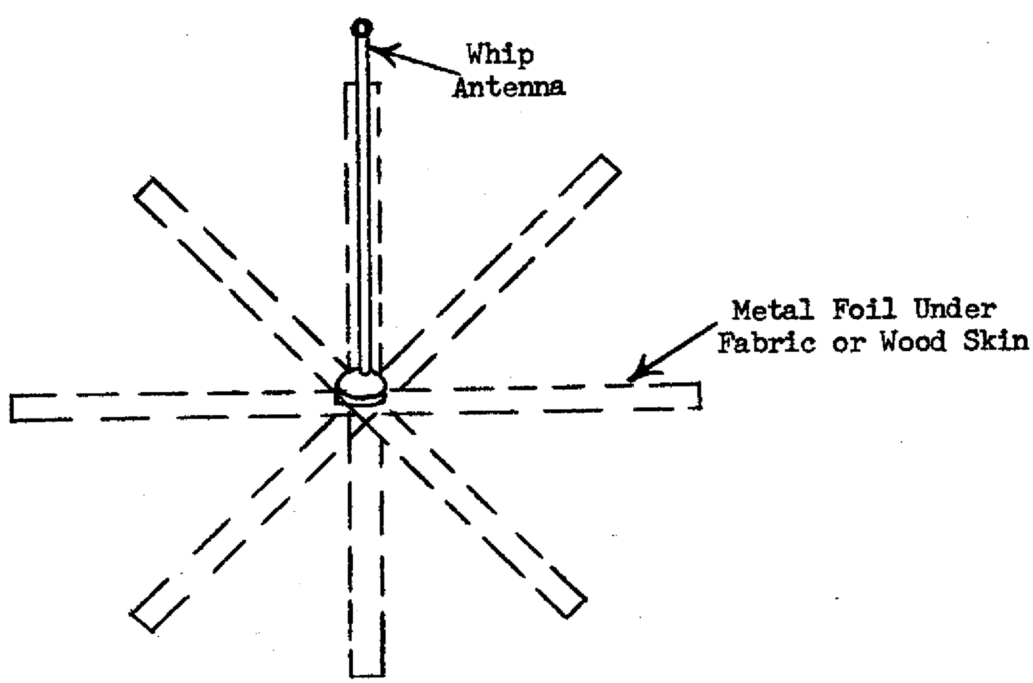
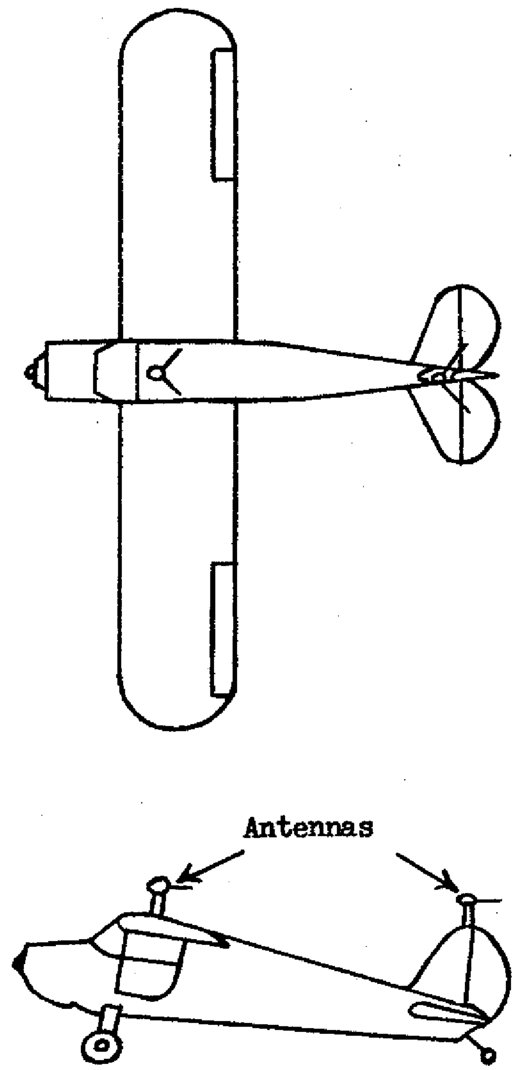
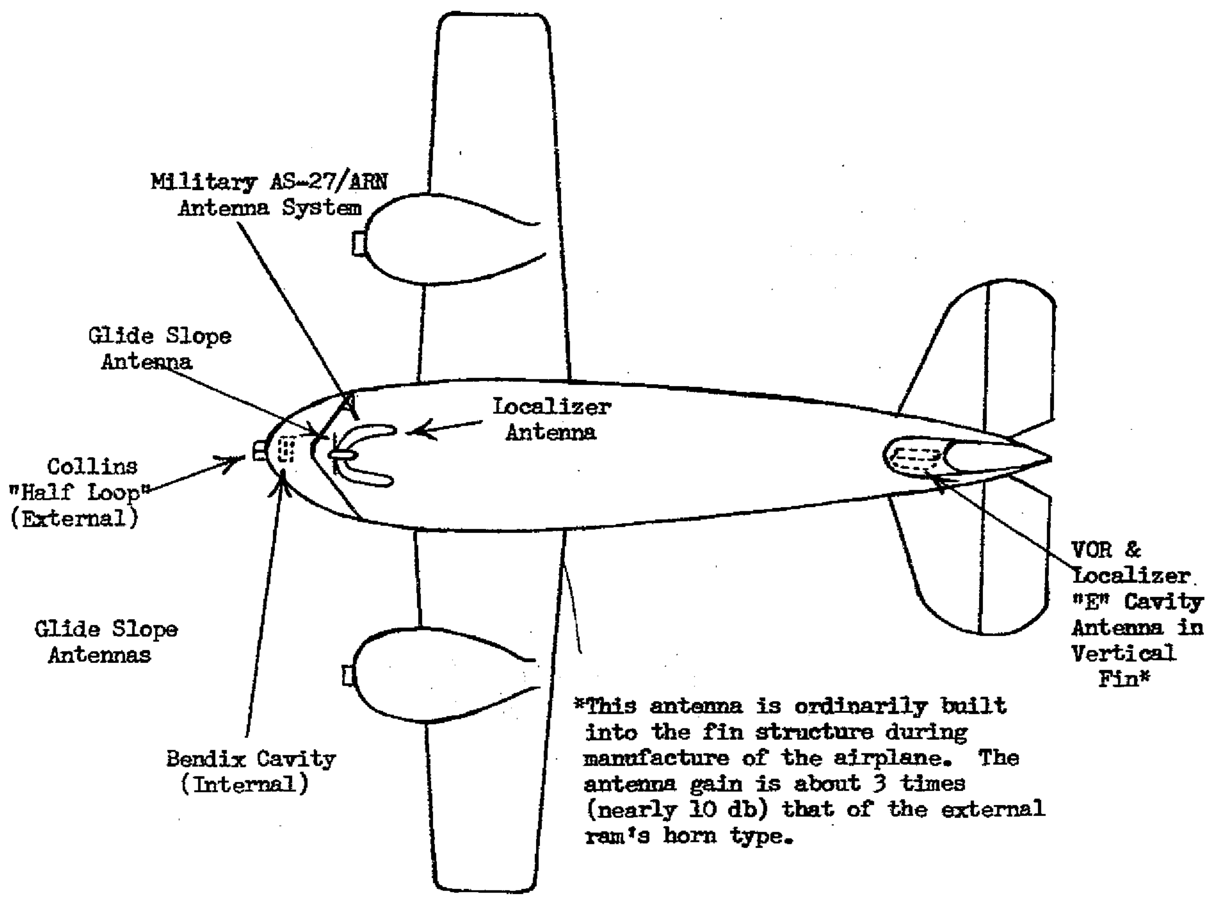


Figure 3



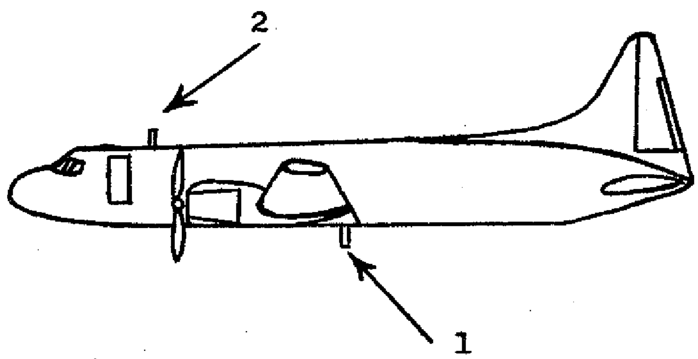
TWO PREFERRED LOCATIONS FOR VOR ANTENNAS
TO PROVIDE GOOD SIGNAL PICK-UP AND MINIMUM
ENGINE IGNITION INTERFERENCE

Figure 4



INSTRUMENT LANDING SYSTEM
GLIDE SLOPE & LOCALIZER RECEIVER ANTENNAS

Figure 5



PREFERRED POSITIONS
OF DISTANCE MEASURING ANTENNAS
IN ORDER OF PREFERENCE

Figure 6

[Appendix C

[Special Civil Air Regulations Which Affect Part 3

[SPECIAL CIVIL AIR REGULATION NO. SR-392B

[Effective: February 25, 1957
[Adopted: February 25, 1957

[Facilitation of Experiments with Exterior Lighting Systems

[Special Civil Air Regulation No. SR-392A adopted June 29, 1955, permits air carriers, subject to the approval of the Administrator, to install and use experimentally, on a limited number of their airplanes, exterior lighting systems which do not conform to the specifications contained in Part 4b of the Civil Air Regulations. The purpose of SR-392A was to permit experimentation on large airplanes while retaining their standard airworthiness certification. Prior to that time such experimentation was conducted either on Government-owned aircraft or on private aircraft limited in operations to the conditions of an experimental certificate.

[SR-392A does not extend the permission for experimentation with exterior lights to non-air-carrier aircraft because at the time of its adoption only air carrier operators indicated interest in this activity. Recently, however, new experimental developments in anti-collision light systems have aroused the interest of private and corporate operators to the extent that some of the operators apparently wish to install the new systems on their aircraft for purposes of experimentation. The Board sees no valid reason why operators other than air carriers should not be permitted to participate, if they wish, in experiments intended to improve the effectiveness of aircraft exterior lighting, provided that the number of such aircraft is reasonably limited.

[Since future experimentation is to be conducted more widely and by private individuals, the Board believes that conditions should be imposed which will assure that the experimental exterior lights are in fact installed for purposes of bona fide experimentation and that the results of such experimentation become available to the Government and to all other interested persons.

[Interested persons have been afforded an opportunity to participate in the making of this regulation (21 F. R. 3388), and due consideration has been given to all relevant matter presented. Since this regulation imposes no additional burden on any person, it may be made effective on less than 30 days' notice.

[In consideration of the foregoing, the Civil Aeronautics Board hereby makes and promulgates the following Special Civil Air Regulation, effective February 25, 1957.

[Contrary provisions of the Civil Air Regulations notwithstanding, experimental exterior lighting equipment which does not comply with the relevant specifications contained in the Civil Air Regulations may, subject to the approval of the Administrator, be installed and used on aircraft for the purpose

of experimentation intended to improve exterior lighting for a period not to exceed six months: *Provided,* That

[(1) The Administrator may grant approval for additional periods if he finds that the experiments can be reasonably expected to contribute to improvements in exterior lighting;

[(2) Not more than 15 aircraft possessing a U. S. certificate of airworthiness may have installed at any one time experimental exterior lighting equipment of one basic type;

[(3) The Administrator shall prescribe such conditions and limitations as may be necessary to insure safety and avoid confusion in air navigation;

[(4) The person engaged in the operation of the aircraft shall disclose publicly the deviations of the exterior lighting from the relevant specifications contained in the Civil Air Regulations at times and in a manner prescribed by the Administrator; and

[(5) Upon application for approval to conduct experimentation with exterior lighting, the applicant shall advise the Administrator of the specific purpose of the experiments to be conducted; and at the conclusion of the approved period of experimentation, he shall advise the Administrator of the detailed results thereof.

[*This regulation supersedes Special Civil Air Regulation No.* SR-392A and shall terminate February 25, 1962, unless sooner superseded or rescinded.]

[Appendix D

[Simplified Method for Determining Air Loads and Air Load Distributions for Wing-Store Combinations

[The following simplified approach may be used to determine the loads and load distributions resulting from the use of tip stores for low speed, low altitude (design Mach number less than 0.4; design altitude less than 15,000 ft.) aircraft with small amounts of sweep (i. e., mid-chord angles of sweep less than 15 degrees).

[*Lift distribution on wing-store combination*

$L_{TOTAL}=L_W+\Delta L.$

where:

L_{TOTAL}=total spanwise load distribution expressed in terms of unit wing loading parameter $\frac{C_l c}{C_L \bar{c}}$ for the wing-store combination.

L_W=spanwise load distribution for wing alone expressed in terms of $\left(\frac{C_l c}{C_L \bar{c}}\right)$ which is obtained by standard methods.

ΔL=additional lift due to tank alone and induced effects of tank on the wing and the wing on the tank expressed in terms of $\left(\frac{\Delta C_l c}{C_L \bar{c}}\right)$. (Obtain from fig. 1.)

[See figure 2 for typical wing with tip-tank air load distribution as determined by above simplified method. For this example, a typical air load distribution for a wing with aspect ratio of 6.17 and span of 400 inches was used. To this wing, a 20-inch diameter tip-tank 80 inches in length was added. It was further assumed that the wing-tip-tank configuration and the wing torsional rigidity are such that deflections under load will not significantly change the air load distribution (i. e., sec. 3.171 (b) is complied with). Note that the ordinates of the $\frac{C_l c}{C_L \bar{c}}+\frac{\Delta C_l c}{C_L \bar{c}}$ curve should be modified so that the area under the new curve will equal the area under the original wing alone curve (i. e., total lift to remain constant).

[*Moment due to stores*

$M_T=M_1+M_2.$

where M_T=total aerodynamic store moment about the wing tip elastic axis.

$M_1=2\ K_1 q \alpha V_o$=moment of load on store which is independent of the wing.

$M_2=.4\ L_5\ C_t$=moment of load on store which is induced by the wing.

K_1=factor proportional to length of tank expressed in diameters. (See table 1.)

q=dynamic pressure (p. s. f.).

α=angle of attack (radians).

V_o=volume of store.

L_5=total load on store induced by wing obtained from distrubution of tank load of figure 1.

C_t=tip chord.

NOTE: Moment coefficient, C_M, of the wing may be assumed not to be changed by presence of tip stores.

[*Lift curve slope*

[The slope of the lift curve for the wing-store combination may be approximated by the formula:

$m_T=[K_2(2D/b)+1]m_w.$

where m_T=slope of the lift curve of the wing-store combination.

m_w=slope of the lift curve of the wing alone.

D=store diameter.

b=wing span.

K_2=1.58 for aspect ratio 5.

K_2=1.25 for aspect ratio 7.

K_2=1.12 for aspect ratio 10 with straight line variation for intermediate aspect ratios.

[TABLE 1.—*Aerodynamic factor* K_1

L/D [1]	K_1	L/D [1]	K_1
1	0	6.01	0.873
1.5	.316	6.97	.897
2.0	.493	8.01	.916
2.51	.607	9.02	.930
2.99	.681	9.97	.939
3.99	.778	∞	1.000
4.99	.836		

[1] L=length of store (ft.).
D=diameter of store (ft.).

[The requirements of section 3.171(b) concerning deflections under load which could change the distribution of external or internal loads should be considered in evaluating each wing-tip-tank configuration. Inertia loads of tip-tanks with varying fuel quantities, different fuel c. g. locations, and the wing torsional rigidity should be considered.

[For stores whose cross section is other than circular (e. g., elliptical), an equivalent store with circular cross section should be assumed. For example, for an elliptical cross section, the equivalent diameter may be assumed equal to the sum of the minor and major diameters divided by 2.

[The simplified procedures are based on the theoretical methods outlined in the following documents:

NACA Research Memorandum RM L53B18, "A Method for Calculating the Aerodynamic Loading on Wing-Tip-Tank Combinations in Subsonic Flow" by S. W. Robinson and M. Zlotnick.

Aeronautic Research Council Report No. 2469, "Theoretical Load Distributions on Wings with Cylindrical Bodies at the Tips" by D. E. Hartley. (British.)