CIVIL AERONAUTICS MANUAL 3

U. S. Department of Commerce — Civil Aeronautics Administration

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This supplement provides manual holders with revised pages containing new sections 3.10-1, 3.241-2, 3.690-1 and new subparagraphs (1) and (2) to 3.291-1. A revision to Figure 1 of Appendix A is also provided.

Remove and destroy the following pages:	Insert in lieu thereof the following pages:
1 and 2	1 and 2
25 and 26	25 and 26
27 and 28	27 and 28, 28-1
49 and 50	49 and 50
69 and 70	69 and 70

Ink revisions:

Add the new sections in the proper places in the Table of Contents.

Note: New or revised material is indicated by brackets [ ]

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Director, Office
of Aviation Safety

Attachments

Part 3 -- Airplane Airworthiness; Normal, Utility, and Acrobatic Categories

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 This part, until superseded or rescinded, shall apply to all airplanes for which applications for type certification under this part were made between the effective date of this part (November 13,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 § 3.18.)

(b) *General design*—(1) *Standard atmosphere.* The standard atmosphere is an atmosphere 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 −67° F. is −0.003566° F./ft. and zero thereabove,

(v) The density ρ_0 at sea level under the above conditions is 0.002378 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 § 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 critica 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 § 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 § 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 § 3.73.)

(4) *Design maximum weight.* The design maximum weight is the maximum weight of the airplane at which compliance is shown with the structural loading conditions. (See § 3.181.)

(5) *Design minimum weight.* The design minimum weight is the minimum weight of the airplane at which compliance is shown with the structural loading conditions. (See § 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 § 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 airspeed indicator reading as installed in the airplane without correction for airspeed 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 § 3.184.)

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

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

(8) V_{fe}. The flap extended speed is a maximum speed with wing flaps in a prescribed extended position. (See § 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 § 3.111.)

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

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

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

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

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

(16) V_{s_1}. The stalling speed or the minimum steady flight speed obtained in a specified configuration. (See § 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 § 3.171.)

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

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

(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.

[CAR, Amdt. 3-7, 17 F. R. 1084, Feb. 5, 1952]

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.

[CAR, Amdt. 3-7, 17 F. R. 1085, Feb. 5, 1952]

[§ 3.10-1 *Substantiation of aircraft with wing tip-tanks. (CAA policy which applies to § 3.10).* The installation of wing tip-tanks is a feature which may create an unsafe condition if the dynamic loading on the wing is neglected in evaluating the design loads under the ground load conditions of § 3.241. Therefore, when an aircraft incorporates wing tip-tanks, substantiation of the wing, wing tip-tank, and wing-fuselage attachment structure should be accomplished in accordance with § 3.241-1.]

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

§ 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 § 1.24 of this subchapter, a change to a type certificate (see § 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 § 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 § 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

(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 § 3.216–1) at the time of approximately 0.15 seconds.

(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 § 3.216 (b))*. The condition given in § 3.216 (b) represents an "unchecked" push-down and is identical to § 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 § 3.216 (c))*. (a) The condition given in § 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 § 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 of 3.2 g. The condition occurring at this time (1.0 second) represents the critical up tail load condition of § 3.216 (c).

[Supp. 10, 16 F. R. 3287, Apr. 14, 1951]

§ 3.216–5 *Principles applicable to detailed analysis of conditions given in § 3.216 (CAA policies which apply to § 3.216)*. (a) The basic principles underlying detailed analysis for the conditions covered in § 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 § 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 § 3.216 (a) and that the specified upload condition is adequately taken care of by § 3.216 (b). For these reasons, the conditions of § 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 § 3.216)*. (a) The curves on Figure 3–3 (b) in § 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:

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

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

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

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

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

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

(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 following conditions:

(a) Positive and negative gusts of 30 feet per second nominal intensity at speed V_c, corresponding to flight condition § 3.187 (a) with flaps retracted.

NOTE: The average loadings of Figures 3–5 (a) and 3–5 (b) and the distribution of Figure 3–9 may be used for the total tail loading in this condition.

(b) Positive and negative gusts of 15 feet per second nominal intensity at speed V_f, corresponding to flight condition § 3.190 (b) with flaps extended. In determining the total load on the horizontal tail for these conditions, the initial balancing tail loads shall first be determined for steady unaccelerated flight at the pertinent design speeds V_c and V_f. The incremental tail load resulting from the gust shall then 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:

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

where:

Δt = the limit gust load increment on the tail in pounds;
K = gust coefficient K in § 3.188,
U = nominal gust intensity in feet per second,
V = airplane speed in miles 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,
R_w = aspect ratio of the wing.

§ 3.217–1 *Gust loads; horizontal tail surfaces (CAA policies which apply to § 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 relieving 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. 3287, 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.

(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 the sideslip.

NOTE: The average loading of Figure 3-3 and the distribution of Figure 3-7 may be used.

(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.

§ 3.219-1 *Vertical surface maneuvering loads (CAA policies which apply to § 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 1g 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 1g 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:

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

where:

$\overline{w}$ = 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.

§ 3.220-1 *Gust loads; vertical tail surfaces (CAA policies which apply to § 3.220).* (a) The K factor specified in § 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 § 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 § 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 §§ 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.* (a) In the symmetrical flight conditions (see §§ 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 § 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:

$$\delta_2 = \frac{V_p}{V_c}\delta_1 \quad \text{and} \quad \delta_3 = \frac{0.5V_p}{V_d}\delta_1$$

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:

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

where:

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

δ_{2i} and δ_{3i} 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 § 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 §§ 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 § 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 § 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 §§ 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 § 3.231 (a)*). The 125 percent factor on computed hinge moments provided in § 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 § 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 § 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 § 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 § 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.	

Surface	*K*
(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 § 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.

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 § 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 tangential force having a limit value equal to 1.25 times the couple force determined from the above criteria.
[4] D = wheel diameter.

FIG. 3-11—PILOT CONTROL FORCE LIMITS

GROUND LOADS

§ 3.241 *Ground loads.* The loads specified in the following conditions shall be considered as the external loads and inertia forces which would occur in an airplane structure if it were acting as a rigid body. 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 § 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 §§ 3.241 through 3.256, except for the following, should be considered applicable: §§ 3.245 (a), 3.246 (a), and 3.250 through 3.252.

(b) The conditions as specified in §§ 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 § 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 § 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.

(1) 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 § 3.241).* The assumption of the aircraft structure as a rigid body in applying the ground load conditions of §§ 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, § 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".[1]

[(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 weight used in the landing conditions shall not be less than the maximum weight for which certification is desired: *Provided, however,* That for multiengine airplanes meeting the one-engine-inoperative climb requirement of § 3.85 (b), the airplane may be designed for a design landing weight which is less than the maximum design weight, if compliance *is shown with the following sections* of Part 4b of this subchapter in lieu of the corresponding requirements of this part: the ground load requirements of § 4b.230, the landing gear requirements of §§ 4b.331 through 4b.336, and the fuel jettisoning system requirements of § 4b.437.

[Amdt. 03-0, 11 F. R. 11374, Nov. 9, 1946, as amended by Amdt. 3-9, 17 F. R. 11631, Dec. 20, 1952]

[1] Not filed for publication in the FEDERAL REGISTER.

§ 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:

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

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 § 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 §§ 3.245–3.247. (See Figs. 3–12 (a) and 3–12 (b) for acceptable landing conditions which are considered to conform with §§ 3.245–3.247.)

§ 3.244–1 *Landing cases and attitudes (CAA policies which apply to § 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 § 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:

the proper supply of energy to the remaining instruments or from the other source.

[Amdt. 3-7, 17 F. R. 1087, Feb. 5, 1952]

§ 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.

[Amdt. 3-7, 17 F. R. 1087, Feb. 5, 1952]

POWER-PLANT INSTRUMENTS

§ 3.670 *Operational markings.* Instruments shall be marked as specified in § 3.759.

§ 3.671 *Instrument lines.* Powerplant instrument lines shall comply with the provisions of § 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 § 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.

[Amdt. 3-4, 15 F. R. 8902, Dec. 15, 1950]

§ 3.672-1 *Means to indicate fuel quantity (CAA policies which apply to § 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 § 3.581 is demonstrated at a speed in excess of the speed of best rate of climb.

§ 3.676 *Carburetor air temperature indicating system.* A carburetor air temperature indicating system shall be provided for each altitude engine equipped with a preheater which is capable of providing a heat rise in excess of 60° F.

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 equipment 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 § 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]

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 § 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 pre-flight 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.

§ 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.* Generator voltage control equipment shall be capable of dependably regulating the generator output within rated limits.

§ 3.687 *Reverse current cut-out.* A generator reverse current cut-out shall 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.

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 *Stall warning indicator circuits (CAA policies which apply to § 3.688)*—(a) *Wiring of circuit by the master switch.* Airplanes on which the indicators are required for type certification as a result of the particular stall characteristics of the airplane, should have the indicator circuit by-pass the master switch. A circuit protector should be installed for the protection of the indicator wiring and this protector should be located as near as is practicable to the source of electric power.

(b) *Wiring of circuit through the master switch.* Where the indicator is installed as an accessory but not as required equipment, it is permissible to wire the indicator through the master switch or direct to the source of power. A circuit protector should be installed for the protection of the indicator wiring and this protector should be located as near as is practicable to the source of the electric power.

[Supp. 10, 16 F. R. 3292, Apr. 14, 1951]

§ 3.689 *Master switch installation.* The master switch or its controls shall be so installed that it is easily discernible and accessible to a member of the crew in flight.

PROTECTIVE DEVICES

§ 3.690 *Fuses or circuit breakers.* If electrical equipment is installed, protective devices (fuses or circuit breakers) shall be installed in the circuits to all electrical equipment, except that such items need not be installed in the main circuits of starter motors or in other circuits where no hazard is presented by their omission.

[§ 3.690–1 *Automatic reset circuit breakers (CAA policies which apply to § 3.690).* Automatic reset circuit breakers (which automatically reset themselves periodically) should not be applied as circuit protective devices.[1] They may be used as integral protectors for electrical equipment (e. g., thermal cut-outs) provided that circuit protection is also installed to protect the cable to the equipment.]

(19 F. R. 8140, Dec. 10, 1954. Effective Dec. 15, 1954.)

§ 3.691 *Protective devices installation.* Protective devices in circuits essential to safety in flight shall be so located and identified that fuses may be replaced or circuit breakers reset readily in flight.

§ 3.692 *Spare fuses.* If fuses are used, one spare of each rating or 50 percent spare fuses of each rating, whichever is greater, shall be provided.

ELECTRIC CABLES

§ 3.693 *Electric cables.* If electrical equipment is installed, the connecting cables used shall be in accordance with recognized standards for electric cable of a slow burning type and of suitable capacity.

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 § 3.696).* The use of the cabin dome light is not considered adequate to comply with the provision of § 3.696.

[Supp. 10, 16 F. R. 3292, 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 §§ 3.700 through 3.703 shall be applicable to the position light system as a whole, and shall be complied with if a single circuit type system is installed.[1] The single circuit system shall include the items specified in paragraphs (b) through (f) 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) *Flasher.* If employed, an approved position light flasher for a single circuit system shall be installed. The flasher shall be such that the system is energized automatically at a rate of not less than 60 nor more than 120 flashes per minute with an on-off ratio between 2.5:1 and 1:1. Unless the flasher is of a fail-safe type, means shall be provided in the system to indicate to the pilot when there is a failure of the flasher and a further means shall be provided for turning the lights on steady in the event of such failure.

(f) *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 of light transmission during normal use.

[Amdt. 3–4, 15 F. R. 8902, Dec. 15, 1950, as amended by Amdt. 3–9, 17 F. R. 11631, Dec. 20, 1952]

§ 3.700–1 *Red passing lights (CAA policies which apply to § 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]

§ 3.700–2 *Fail-safe position light flashers (single-circuit) (CAA policies which apply to § 3.700 (e)).* (a) When subjected to the conditions of failure specified in paragraph (b) of this section, a position light flasher is considered to have "failed-safe" when:

(1) The position light circuit is closed continuously, or

(2) The position light circuit is alternately closed and opened in such manner that,

(i) The frequency is not less than 40 cycles per minute and not greater than 120 cycles per minute,

(ii) The ratio of the closed circuit interval to the open circuit interval is not less than 1:1 and not greater than 2:1.

(b) Conditions of failure are as follows:

(1) At room ambient conditions, when the supply voltage is adjusted to the minimum value at which perceptible light is emitted by the position light bulb.

(2) At nominal supply voltage and at room ambient conditions,

(i) With any one position light branch circuit open, or

(ii) After any single malfunction within the flasher timing device, such as open circuit, short circuit, or jamming of a contact in its open or closed position.

[Supp. 12, 16 F. R. 6743, July 12, 1951]

§ 3.700–3 *Anti-collision light (CAA policies which apply to § 3.700 (a)).* Anti-collision 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 would not be detrimental to the crew's vision and would not detract from the conspicuity of the position lights. The color of the anti-collision light should be aviation red in accordance with the specifications of § 3.703. The arrangement of the anti-collision 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.

[Supp. 18, 18 F. R. 7339, Nov. 20, 1953]

§ 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,

[[1] Circuit protective devices are normally installed to limit the hazardous consequences of overloaded or faulted circuits. These devices are resettable (circuit breakers) or replaceable (fuses) to permit the crew to restore service when nuisance trips occur or when the abnormal circuit condition can be corrected in flight. If the abnormal circuit condition can *not* be corrected in flight, the decision to restore power to the circuit involves a careful analysis of the flight situation. It is necessary to weigh the essentiality of the circuit for continued safe flight against the hazards of resetting on a possibly faulted circuit. Such evaluation is properly an aircraft crew function which can not be performed by automatic reset circuit breakers. To assure crew supervision over the reset operation, circuit protective devices should be of such design that a manual operation is required to restore service after tripping.]

[1] Requirements for dual circuit position light systems are contained in Part 4b of this subchapter.

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.

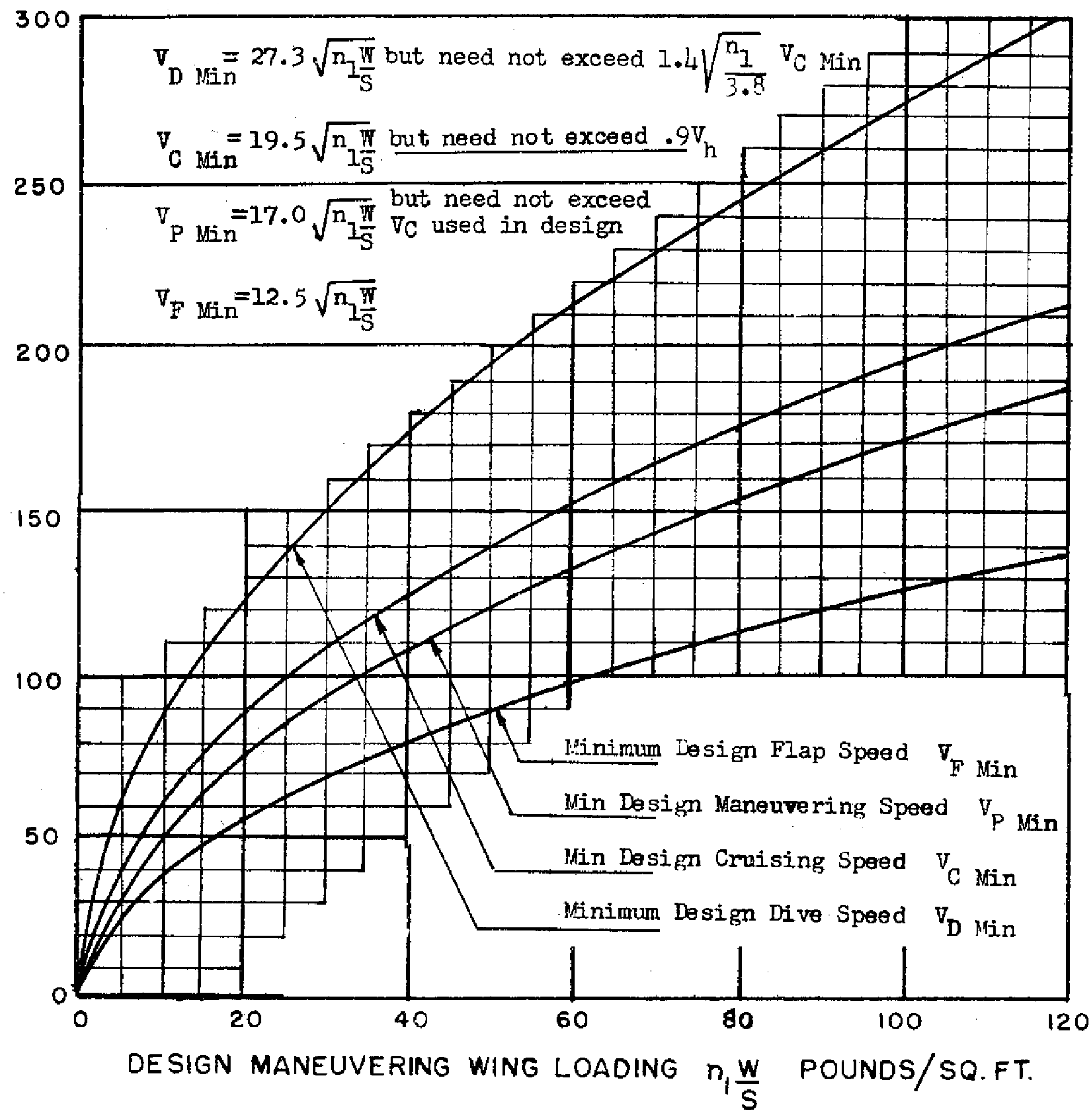
8.11 Acceptable maximum and minimum pilot loads for elevator, aileron, and rudder controls are as shown in Figure 3-11 of Part 3. These pilot loads should be assumed to act at the appropriate control grips or pads in a manner simulating flight conditions and to be reacted at the attachments of the control 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 I
MINIMUM DESIGN AIR SPEEDS



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