Designation: F3116/F3116M − 15

Standard Specification for Design Loads and Conditions[1]

This standard is issued under the fixed designation F3116/F3116M; the number immediately following the designation indicates the year of original adoption or, in the case of revision, the year of last revision. A number in parentheses indicates the year of last reapproval. A superscript epsilon (ε) indicates an editorial change since the last revision or reapproval.

1. Scope

1.1 This specification addresses the airworthiness requirements for the design loads and conditions of small airplanes.

1.2 This specification is applicable to small airplanes as defined in the F44 terminology standard. Use of the term airplane is used throughout this specification and will mean "small airplane."

1.3 The applicant for a design approval must seek individual guidance from their respective CAA body concerning the use of this standard as part of a certification plan. For information on which CAA regulatory bodies have accepted this standard (in whole or in part) as a means of compliance to their Small Airplane Airworthiness Rules (hereinafter referred to as "the Rules"), refer to ASTM F44 webpage (www.ASTM.org/COMMITTEE/F44.htm) which includes CAA website links.

1.4 *Units*—Currently there is a mix of SI and Imperial units. In many locations, SI units have been included otherwise units are as they appear in Amendment 62 of 14 CFR Part 23. In a future revision values will be consistently stated in SI units followed by Imperial units in square brackets. The values stated in each system may not be exact equivalents; therefore, each system shall be used independently of the other. Combining values from the two systems may result in nonconformance with the standard.

1.5 *This standard does not purport to address all of the safety concerns, if any, associated with its use. It is the responsibility of the user of this standard to establish appropriate safety and health practices and determine the applicability of regulatory limitations prior to use.*

2. Referenced Documents

2.1 *ASTM Standards:*[2]

F3060 Terminology for Aircraft

2.2 *U.S. Code of Federal Regulations:*[3]

14 CFR Part 23 Airworthiness Standards: Normal, Utility, Aerobatic and Commuter Category Airplanes (Amendment 62)

2.3 *European Aviation Safety Agency Regulations:*

Certification Specifications for Normal, Utility, Aerobatic, and Commuter Category Aeroplanes (CS-23, Amendment 3)

Certification Specifications for Very Light Aeroplanes (CS-VLA, Amendment 1)

3. Terminology

3.1 A listing of terms, abbreviations, acronyms, and symbols related to aircraft covered by ASTM Committees F37 and F44 airworthiness design standards can be found in Terminology F3060. Items listed below are more specific to this standard.

3.2 *Definitions of Terms Specific to This Standard:*

3.2.1 *chordwise, n*—directed, moving, or placed along the chord of an airfoil section.

3.2.2 *downwash, n*—the downward deflection of an airstream by an aircraft wing.

3.2.3 *flight envelope, n*—any combination of airspeed and load factor on and within the boundaries of a flight envelope that represents the envelope of the flight loading conditions specified by the maneuvering and gust criteria.

3.2.4 *flight load factor, n*—represents the ratio of the aerodynamic force component (acting normal to the assumed longitudinal axis of the airplane) to the weight of the airplane. A positive flight load factor is one in which the aerodynamic force acts upward, with respect to the airplane.

3.2.5 *propeller slipstream, n*—the airstream pushed back by a revolving aircraft propeller.

3.2.6 *spanwise, n*—directed, moving, or placed along the span of an airfoil.

3.2.7 *winglet, n*—a nearly vertical airfoil at an airplane's wingtip.

3.3 *Acronyms:*

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[1] This specification is under the jurisdiction of ASTM Committee F44 on General Aviation Aircraft and is the direct responsibility of Subcommittee F44.30 on Structures.

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[2] For referenced ASTM standards, visit the ASTM website, www.astm.org, or contact ASTM Customer Service at service@astm.org. For *Annual Book of ASTM Standards* volume information, refer to the standard's Document Summary page on the ASTM website.

[3] Available from U.S. Government Printing Office Superintendent of Documents, 732 N. Capitol St., NW, Mail Stop: SDE, Washington, DC 20401, http://www.access.gpo.gov.

3.3.1 *MCP*—maximum continuous power

3.4 *Symbols:*

C_{NA} = maximum airplane normal force coefficient

M_C = design cruising speed (Mach number)

V_E = design dive speed at zero or negative load factor

V_{SF} = stalling speed with flaps fully extended

4. Flight Loads

4.1 *Loads:*

4.1.1 Unless otherwise provided, prescribed loads are limit loads.

4.1.2 Unless otherwise provided, the air, ground, and water loads must be placed in equilibrium with inertia forces, considering each item of mass in the airplane. These loads must be distributed to conservatively approximate or closely represent actual conditions. Methods used to determine load intensities and distribution on canard and tandem wing configurations must be validated by flight test measurement unless the methods used for determining those loading conditions are shown to be reliable or conservative on the configuration under consideration.

4.1.3 If deflections under load would significantly change the distribution of external or internal loads, this redistribution must be taken into account.

4.1.4 Appendix X1 through Appendix X4 provides, within the limitations specified within the appendix, a simplified means of compliance with several of the requirements set forth in 4.2 to 4.26 and 7.1 to 7.9 that can be applied as one (but not the only) means to comply.

4.2 *General:*

4.2.1 Flight load factors, *n*, represent the ratio of the aerodynamic force component (acting normal to the assumed longitudinal axis of the airplane) to the weight of the airplane. A positive flight load factor is one in which the aerodynamic force acts upward, with respect to the airplane.

4.2.2 Compliance with the flight load requirements of this subpart must be shown:

4.2.2.1 At each critical altitude within the range in which the airplane may be expected to operate;

4.2.2.2 At each weight from the design minimum weight to the design maximum weight; and

4.2.2.3 For each required altitude and weight, for any practicable distribution of disposable load within the operating limitations specified in 14 CFR Part 23, Sections 23.1583 through 23.1589.

4.2.3 When significant, the effects of compressibility must be taken into account.

4.3 *Symmetrical Flight Conditions:*

4.3.1 The appropriate balancing horizontal tail load must be accounted for in a rational or conservative manner when determining the wing loads and linear inertia loads corresponding to any of the symmetrical flight conditions specified in 4.4 through 4.6.

4.3.2 The incremental horizontal tail loads due to maneuvering and gusts must be reacted by the angular inertia of the airplane in a rational or conservative manner.

4.3.3 Mutual influence of the aerodynamic surfaces must be taken into account when determining flight loads.

4.4 *Flight Envelope:*

4.4.1 *General*—Compliance with the strength requirements of this subpart must be shown at any combination of airspeed and load factor on and within the boundaries of a flight envelope (similar to the one in 4.4.4) that represents the envelope of the flight loading conditions specified by the maneuvering and gust criteria of 4.4.2 and 4.4.3 respectively.

4.4.2 *Maneuvering Envelope*—Except where limited by maximum (static) lift coefficients, the airplane is assumed to be subjected to symmetrical maneuvers resulting in the following limit load factors:

4.4.2.1 The positive maneuvering load factor specified in 4.5 at speeds up to V_D;

4.4.2.2 The negative maneuvering load factor specified in 4.5 at V_C; and

4.4.2.3 Factors varying linearly with speed from the specified value at V_C to 0.0 at V_D. For airplanes with a positive limit maneuvering load factor greater than 3.8, use a value of –1.0 at V_D.

4.4.3 *Gust Envelope:*

4.4.3.1 The airplane is assumed to be subjected to symmetrical vertical gusts in level flight. The resulting limit load factors must correspond to the conditions determined as follows:

(1) Positive (up) and negative (down) gusts of 15.24 m/s [50 fps] at V_C must be considered at altitudes between sea level and 6,096 m [20 000 ft]. The gust velocity may be reduced linearly from 15.24 m/s [50 fps] at 6096 m [20 000 ft] to 7.62 m/s [25 fps] at 15 240 m [50 000 ft]; and

(2) Positive and negative gusts of 7.62 m/s [25 fps] at V_D must be considered at altitudes between sea level and 6,096 m [20 000 ft]. The gust velocity may be reduced linearly from 7.62 m/s [25 fps] at 6096 m [20 000 ft] to 3.81 m/s [12.5 fps] at 15 240 m [50 000 ft].

(3) In addition, for level 4 airplanes, positive (up) and negative (down) rough air gusts of 20.12 m/s [66 fps] at V_B must be considered at altitudes between sea level and 6096 m [20 000 ft]. The gust velocity may be reduced linearly from 20.12 m/s [66 fps] at 6096 m [20 000 ft] to 11.58 m/s [38 fps] at 15 240 m [50 000 ft].

4.4.3.2 The following assumptions must be made:

(1) The shape of the gust is:

$$U = \frac{U_{de}}{2}\left(1 - \cos\frac{2\pi s}{25C}\right) \quad (1)$$

where:

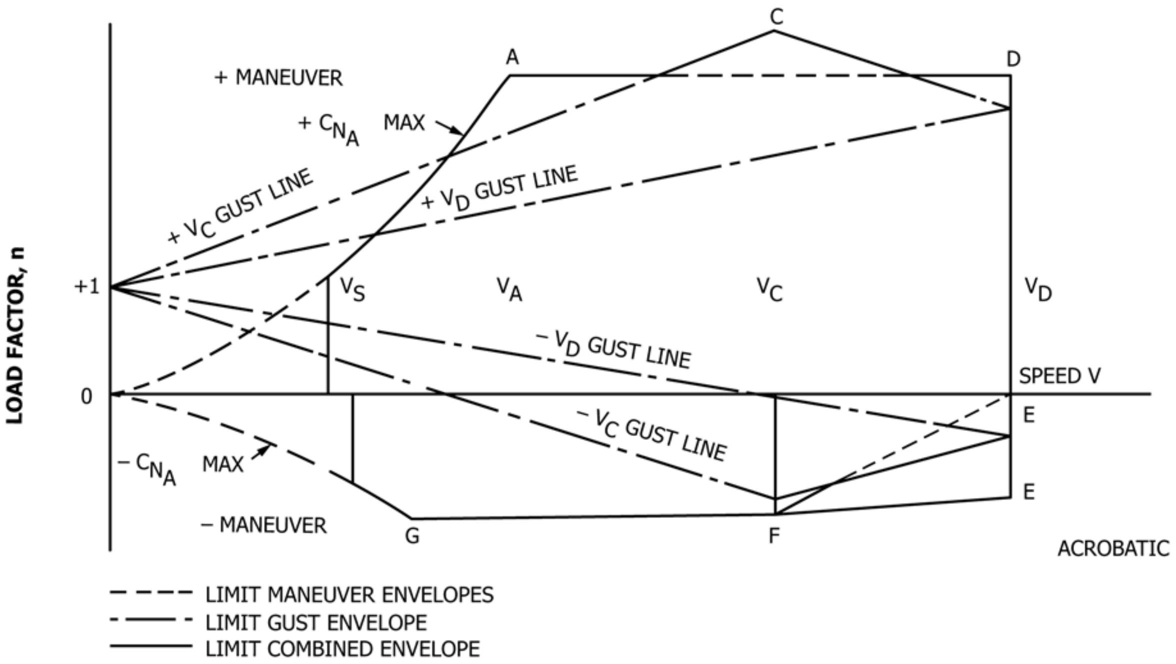
s = distance penetrated into gust (m or [ft]);
C = mean geometric chord of wing (m or [ft]); and
U_{de} = derived gust velocity referred to in 4.4.3.1 (m/s or [fps]).

(2) Gust load factors vary linearly with speed between V_C and V_D.

4.4.4 *Flight Envelope*—See Fig. 1.

4.5 *Limit Maneuvering Load Factors:*

4.5.1 The positive limit maneuvering load factor *n* may not be less than:



NOTE 1—Point G need not be investigated when the supplementary condition specified in 4.14 is investigated.

FIG. 1 Flight Envelope

4.5.1.1 $2.1 + \frac{24,000}{W + 10,000}$, where W = design maximum take-off weight (lb), except that n need not be more than 3.8;

4.5.1.2 6.0 for airplanes approved for aerobatics.

4.5.2 The negative limit maneuvering load factor may not be less than:

4.5.2.1 0.4 times the positive load factor;

4.5.2.2 0.5 times the positive load factor for airplanes approved for aerobatics.

4.5.3 Maneuvering load factors lower than those specified in this section may be used if the airplane has design features that make it impossible to exceed these values in flight.

4.6 *Gust Load Factors:*

4.6.1 Each airplane must be designed to withstand loads on each lifting surface resulting from gusts specified in 4.4.3.

4.6.2 The gust load factors for a canard or tandem wing configuration must be computed using a rational analysis, or may be computed in accordance with 4.6.3, provided that the resulting net loads are shown to be conservative with respect to the gust criteria of 4.4.3.

4.6.3 In the absence of a more rational analysis, the gust load factors must be computed as follows:

$$n = 1 + \frac{K_g U_{de} V a}{498\left(\frac{W}{S}\right)} \tag{2}$$

where:

$K_g = \frac{0.88\mu_g}{5.3 + \mu_g}$ = gust alleviation factor;

$\mu_g = \frac{2(W/S)}{\rho C a g}$ = airplane mass ratio;

U_{de} = derived gust velocities referred to in 4.4.3 (f.p.s.).

ρ = density of air (slugs/ft^3);

W/S = wing loading (p.s.f.) due to the applicable weight of the airplane in the particular load case;

C = mean geometric chord (ft);

g = acceleration due to gravity (ft/s^2);

V = airplane equivalent speed (knots); and

a = slope of the airplane normal force coefficient curve C_{NA} per radian if the gust loads are applied to the wings and horizontal tail surfaces simultaneously by a rational method. The wing lift curve slope C_L per radian may be used when the gust load is applied to the wings only and the horizontal tail gust loads are treated as a separate condition.

4.7 *Design Fuel Loads:*

4.7.1 The disposable load combinations must include each fuel load in the range from zero fuel to the selected maximum fuel load.

4.7.2 If fuel is carried in the wings, the maximum allowable weight of the airplane without any fuel in the wing tank(s) must be established as "maximum zero wing fuel weight," if it is less than the maximum weight.

4.7.3 For level 4 airplanes, a structural reserve fuel condition, not exceeding fuel necessary for 45 min of operation at maximum continuous power, may be selected. If a structural reserve fuel condition is selected, it must be used as the minimum fuel weight condition for showing compliance with the flight load requirements prescribed in this part and:

4.7.3.1 The structure must be designed to withstand a condition of zero fuel in the wing at limit loads corresponding to:

(1) 90 % of the maneuvering load factors defined in 4.5, and

(2) Gust velocities equal to 85 % of the values prescribed in 4.4.3.

4.7.3.2 The fatigue evaluation of the structure must account for any increase in operating stresses resulting from the design condition of 4.7.3.1.

4.7.3.3 The flutter, deformation, and vibration requirements must also be met with zero fuel in the wings.

4.8 *High Lift Devices:*

4.8.1 If wing flaps or similar high lift devices are installed for use in take-off, approach, or landing, the airplane, with the flaps fully deflected at V_F, is assumed to be subjected to symmetrical maneuvers and gusts resulting in limit load factors within the range determined by:

4.8.1.1 Maneuvering, to a positive limit load factor of 2.0; and

4.8.1.2 Positive and negative gust of 7.62 m/s [25 fps] acting normal to the flight path in level flight.

4.8.1.3 However, if an automatic flap load limiting device is used, the airplane may be designed for the critical combinations of airspeed and flap position allowed by that device.

4.8.2 V_F must be assumed to be not less than 1.4 V_S or 1.8 V_{SF}, whichever is greater, where:

4.8.2.1 V_S is the 1g computed stalling speed with flaps retracted at the design weight; and

4.8.2.2 V_{SF} is the 1g computed stalling speed with flaps fully extended at the design weight.

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

4.8.4 The flaps, their operating mechanism, and their supporting structures, must be designed for the conditions prescribed in 4.8.1. In addition, with the flaps fully extended at V_F, the following conditions, taken separately, must be accounted for:

4.8.4.1 A head-on gust having a velocity of 7.62 m/s [25 fps] (EAS), combined with propeller slipstream corresponding to 75 % of maximum continuous power; and

4.8.4.2 The effects of propeller slipstream corresponding to maximum takeoff power.

4.8.4.3 For the investigation of slipstream effects, the load factor may be assumed to be 1.0.

4.9 *Unsymmetrical Flight Conditions:*

4.9.1 The airplane is assumed to be subjected to the unsymmetrical flight conditions of 4.10 and 4.11. Unbalanced aerodynamic moments about the center of gravity must be reacted in a rational or conservative manner, considering the principal masses furnishing the reacting inertia forces.

4.9.2 Airplanes approved for aerobatics must be designed for additional asymmetric loads acting on the wing and the horizontal tail.

4.10 *Rolling Conditions*—The wing and wing bracing must be designed for the following loading conditions:

4.10.1 Unsymmetrical wing loads. Unless the following values result in unrealistic loads, the rolling accelerations may be obtained by modifying the symmetrical flight conditions in 4.4.4 as follows:

4.10.1.1 In Condition A, assume that 100 % of the semispan wing airload acts on one side of the airplane and 70 % of this load acts on the other side. For airplanes of more than 454 kg [1000 lb] design weight, the latter percentage may be increased linearly with weight up to 75 % at 5670 kg [12 500 lb].

4.10.1.2 For airplanes approved for aerobatics, in conditions A and F, assume that 100 % of the semispan wing airload acts on one side of the plane of symmetry and 60 % of this load acts on the other side.

4.10.2 The loads resulting from the aileron deflections and speeds specified in 4.25, in combination with an airplane load factor of at least two thirds of the positive maneuvering load factor used for design. Unless the following values result in unrealistic loads, 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 determined in 4.4.4:

$$\Delta c_m = -0.01\ \delta \tag{3}$$

where:

Δc_m = the moment coefficient increment; and
δ = the down aileron deflection in degrees in the critical condition.

4.11 *Yawing Conditions*—The airplane must be designed for yawing loads on the vertical surfaces resulting from the loads specified in 4.20 through 4.22.

4.12 *Pressurized Cabin Loads*—For each pressurized compartment, the following applies:

4.12.1 The airplane structure must be strong enough to withstand the flight loads combined with pressure differential loads from zero up to the maximum relief valve setting.

4.12.2 The external pressure distribution in flight, and any stress concentrations, must be accounted for.

4.12.3 If landings may be made with the cabin pressurized, landing loads must be combined with pressure differential loads from zero up to the maximum allowed during landing.

4.12.4 The airplane structure must be strong enough to withstand the pressure differential loads corresponding to the maximum relief valve setting multiplied by a factor of 1.33, omitting other loads.

4.12.5 If a pressurized cabin has two or more compartments separated by bulkheads or a floor, the primary structure must be designed for the effects of sudden release of pressure in any compartment with external doors or windows. This condition must be investigated for the effects of failure of the largest opening in the compartment. The effects of intercompartmental venting may be considered.

4.13 *Unsymmetrical Loads Due to Engine Failure:*

4.13.1 Multi-engine airplanes must be designed for the unsymmetrical loads resulting from the failure of the critical engine including the following conditions in combination with a single malfunction of the propeller drag limiting system, considering the probable pilot corrective action on the flight controls:

4.13.1.1 At speeds between V_{MC} and V_D, the loads resulting from power failure because of fuel flow interruption are considered to be limit loads.

4.13.1.2 At speeds between V_{MC} and V_C, the loads resulting from the disconnection of the engine compressor from the turbine or from loss of the turbine blades are considered to be ultimate loads.

4.13.1.3 The time history of the thrust decay and drag buildup occurring as a result of the prescribed engine failures must be substantiated by test or other data applicable to the particular engine-propeller combination.

4.13.1.4 The timing and magnitude of the probable pilot corrective action must be conservatively estimated, considering the characteristics of the particular engine-propeller-airplane combination.

4.13.2 Pilot corrective action may be assumed to be initiated at the time maximum yawing velocity is reached, but not earlier than 2 s after the engine failure. The magnitude of the corrective action may be based on the limit pilot forces specified in 7.4 except that lower forces may be assumed where it is shown by analysis or test that these forces can control the yaw and roll resulting from the prescribed engine failure conditions.

4.14 *Rear Lift Truss:*

4.14.1 If a rear lift truss is used, it must be designed for conditions of reversed airflow at a design speed of:

$$V = 8.7\sqrt{\frac{W}{S}} + 8.7 \text{ (knots)} \tag{4}$$

where:

W/S = wing loading (lb/ft^2) at design maximum takeoff weight.

4.14.2 Either aerodynamic data for the particular wing section used, or a value of C_L equalling –0.8 with a chordwise distribution that is triangular between a peak at the trailing edge and zero at the leading edge, must be used.

4.15 *Speed Control Devices*—If speed control devices (such as spoilers and drag flaps) are incorporated for use in enroute conditions:

4.15.1 The airplane must be designed for the symmetrical maneuvers and gusts prescribed in 4.4, 4.5, and 4.6, and the yawing maneuvers and lateral gusts in 4.20 and 4.21, with the device extended at speeds up to the placard device extended speed; and

4.15.2 If the device has automatic operating or load limiting features, the airplane must be designed for the maneuver and gust conditions prescribed in 4.15.1 at the speeds and corresponding device positions that the mechanism allows.

4.16 *Balancing Loads:*

4.16.1 A horizontal surface balancing load is a load necessary to maintain equilibrium in any specified flight condition with no pitching acceleration.

4.16.2 Horizontal balancing surfaces must be designed for the balancing loads occurring at any point on the limit maneuvering envelope and in the flap conditions specified in 4.8.

4.16.3 For airplanes meeting the limitations of X4.1, the distribution in Fig. X4.5 of Appendix X4 may be used.

4.17 *Maneuvering Loads for Horizontal Surfaces*—Each horizontal surface and its supporting structure, and the main wing of a canard or tandem wing configuration, if that surface has pitch control, must be designed for the maneuvering loads imposed by conditions 4.17.1 and 4.17.2. For airplanes meeting the limitations of X4.1, either condition 4.17.3 or condition 4.17.4 can be used instead of the loads determined in conditions 4.17.1 and 4.17.2.

4.17.1 A sudden movement of the pitching control at the speed V_A,

4.17.1.1 to the maximum aft movement (upward deflection), and

4.17.1.2 the maximum forward movement (downward deflection), as limited by the control stops, or pilot effort, whichever is critical.

4.17.1.3 For airplanes meeting the limitations of X4.1, the average loading of X4.3 of Appendix X4 and the distribution in Fig. X4.6 of Appendix X4 may be used.

4.17.2 A sudden aft movement of the pitching control at speeds above V_A, followed by a forward movement of the pitching control resulting in the following combinations of normal and angular acceleration:

Condition	Normal acceleration (n)	Angular acceleration (radian/s^2)
Nose-up pitching (down load)	1.0	$+\frac{39}{V}n_m(n_m - 1.5)$
Nose-down pitching (up load)	n_m	$-\frac{39}{V}n_m(n_m - 1.5)$

where:

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

V = initial speed in knots.

4.17.2.1 The conditions in this section involve loads corresponding to the loads that may occur in a "checked maneuver" (a maneuver in which the pitching control is suddenly displaced in one direction and then suddenly moved in the opposite direction). The deflections and timing of the "checked maneuver" must avoid exceeding the limit maneuvering load factor. The total horizontal surface load for both nose-up and nose-down pitching conditions is the sum of the balancing loads at V and the specified value of the normal load factor n, plus the maneuvering load increment due to the specified value of the angular acceleration. For airplanes meeting the limitations of X4.1, the maneuvering load increment in Fig. X4.2 of Appendix X4 and the distributions in Fig. X4.6 (for down loads) and in Fig. X4.7 (for up loads) of Appendix X4 may be used.

4.17.3 A sudden deflection of the elevator, the following cases must be considered:

4.17.3.1 Speed V_A, maximum upward deflection;

4.17.3.2 Speed V_A, maximum downward deflection;

4.17.3.3 Speed V_D, one-third maximum upward deflection;

4.17.3.4 Speed V_D, one-third maximum downward deflection.

4.17.3.5 The following assumptions must be made:

(1) The airplane is initially in level flight, and its attitude and air speed do not change.

(2) The loads are balanced by inertia forces.

4.17.4 A sudden deflection of the elevator such as to cause the normal acceleration to change from an initial value to a final value, the following cases being considered (see Fig. 2):

Speed	Initial Condition	Final Condition	Load Factor Increment
V_A	A_1	A	n1 − 1
	A	A_1	1 − n1
	A_1	G	n4 − 1
	G	A_1	1 − n4
V_D	D_1	D	n2 − 1
	D	D_1	1 − n2
	D_1	E	n3 − 1
	E	D_1	1 − n3

4.17.5 For the purpose of this calculation the difference in air speed between V_A and the value corresponding to point G on the maneuvering envelope can be ignored. The following assumptions must be made:

4.17.5.1 The airplane is initially in level flight, and its attitude and airspeed do not change;

4.17.5.2 The loads are balanced by inertia forces;

4.17.5.3 The aerodynamic tail load increment is given by:

$$\Delta P = \Delta n M g \left[\frac{X_{cg}}{l_t} - \frac{S_{ht}}{S} \frac{a_{ht}}{a} \left(1 - \frac{d\epsilon}{d\alpha} \right) - \frac{\rho_0}{2} \left(\frac{S_{ht} a_{ht} l_t}{M} \right) \right] \quad (5)$$

where:

ΔP = horizontal tail load increment, positive upwards (N),
Δn = load factor increment,
M = mass of the airplane (kg),
g = acceleration due to gravity (m/s^2),
X_{cg} = longitudinal distance of airplane c.g. aft of aerodynamic center of airplane less horizontal tail (m),
S_{ht} = horizontal tail area (m^2),
a_{ht} = slope of horizontal tail lift curve per radian,
$\frac{d\epsilon}{d\alpha}$ = rate of change of downwash angle with angle of attack,
ρ_0 = density of air at sea-level (kg/m^3),
l_t = tail arm (m),
S = wing area (m^2), and
a = slope of wing lift curve per radian.

4.18 *Gust Loads for Horizontal Surfaces:*

4.18.1 Each horizontal surface, other than a main wing, must be designed for loads resulting from:

4.18.1.1 Gust velocities specified in 4.4.3 with flaps retracted; and

4.18.1.2 Positive and negative gusts of 7.62 m/s [25 f.p.s.] nominal intensity at V_F, corresponding to the flight conditions specified in 4.8.1.2.

4.18.2 For airplanes meeting the limitations of X4.1, the average loadings in Fig. X4.3 and the distribution of Fig. X4.7 may be used to determine the incremental gust loads for the requirements of 4.18.1 applied as both up and down increments for 4.18.3.

4.18.3 When determining the total load on the horizontal surfaces for the conditions specified in 4.18.1, the initial balancing loads for steady unaccelerated flight at the pertinent design speeds V_F, V_C, and V_D must first be determined. The incremental load resulting from the gusts must be added to the initial balancing load to obtain the total load.

4.18.4 In the absence of a more rational analysis, the incremental load due to the gust must be computed as follows only on airplane configurations with aft-mounted, horizontal surfaces, unless its use elsewhere is shown to be conservative:

$$\Delta L_{ht} = \frac{K_g U_{de} V a_{ht} S_{ht}}{498} \left(1 - \frac{d_e}{d_\propto} \right) \quad (6)$$

where:

ΔL_{ht} = incremental horizontal tail load (lb);
K_g = gust alleviation factor defined in 4.6;
U_{de} = derived gust velocity (f.p.s.);
V = airplane equivalent speed (knots);
a_{ht} = slope of aft horizontal tail lift curve (per radian);
S_{ht} = area of aft horizontal lift surface (ft^2); and
$\left(1 - \frac{d_e}{d_\propto} \right)$ = downwash factor.

4.19 *Unsymmetrical Loads:*

4.19.1 Horizontal surfaces other than main wing and their supporting structure must be designed for unsymmetrical loads arising from yawing and slip-stream effects, in combination with the loads prescribed for the flight conditions set forth in 4.16 through 4.18.

4.19.2 In the absence of more rational data for airplanes that are conventional in regard to location of engines, wings, horizontal surfaces other than main wing, and fuselage shape:

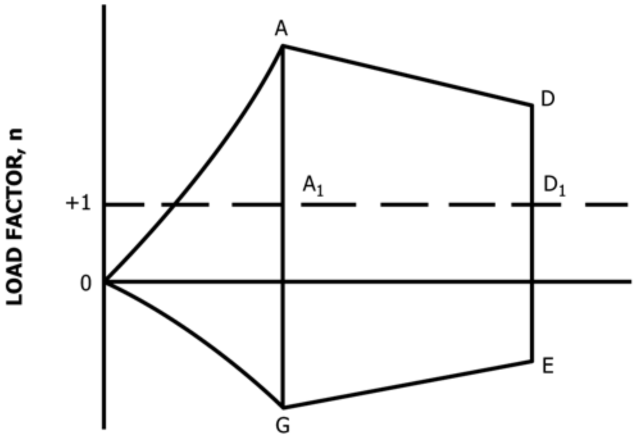


FIG. 2 Pitching Maneuvers

4.19.2.1 100 % of the maximum loading from the symmetrical flight conditions may be assumed on the surface on one side of the plane of symmetry; and

4.19.2.2 The following percentage of that loading must be applied to the opposite side: Percent = 100 – 10 (n – 1), where n is the specified positive maneuvering load factor, but this value may not be more than 80 %.

4.19.3 For airplanes that are not conventional (such as airplanes with horizontal surfaces other than main wing having appreciable dihedral or supported by the vertical tail surfaces) the surfaces and supporting structures must be designed for combined vertical and horizontal surface loads resulting from each prescribed flight condition taken separately.

4.20 *Maneuvering Loads for Vertical Surfaces:*

4.20.1 At speeds up to V_A, the vertical surfaces must be designed to withstand the following conditions. In computing the loads, the yawing velocity may be assumed to be zero:

4.20.1.1 With the airplane in unaccelerated flight at zero yaw, it is assumed that the rudder control is suddenly displaced to the maximum deflection, as limited by the control stops or by limit pilot forces.

4.20.1.2 With the rudder deflected as specified in 4.20.1.1, it is assumed that the airplane yaws to the overswing sideslip angle. In lieu of a rational analysis, an overswing angle may be assumed equal to 1.5 times the static sideslip angle of 4.20.1.3.

4.20.1.3 A yaw angle of 15° with the rudder control maintained in the neutral position (except as limited by pilot strength).

4.20.2 For airplanes meeting the limitations of X4.1, the average loading of Appendix X4, X4.3 and Fig. X4.1 of Appendix X4 and the distribution in Fig. X4.5, Fig. X4.6 and Fig. X4.7 of Appendix X4 may be used instead of the requirements of 4.20.1.2, 4.20.1.1, and 4.20.1.3, respectively.

4.20.3 For level 4 airplanes, the loads imposed by the following additional maneuver must be substantiated at speeds from V_A to V_D/M_D. When computing the tail loads:

4.20.3.1 The airplane must be yawed to the largest attainable steady state sideslip angle, with the rudder at maximum deflection caused by any one of the following:

(1) Control surface stops;
(2) Maximum available booster effort;
(3) Maximum pilot rudder force as shown in Fig. 3.

4.20.3.2 The rudder must be suddenly displaced from the maximum deflection to the neutral position.

4.20.4 The yaw angles specified in 4.20.1.3 may be reduced if the yaw angle chosen for a particular speed cannot be exceeded in:

4.20.4.1 Steady slip conditions;

4.20.4.2 Uncoordinated rolls from steep banks; or

4.20.4.3 For multi-engine airplanes, the sudden failure of the critical engine with delayed corrective action.

4.21 *Gust Loads for Vertical Surfaces:*

4.21.1 Vertical surfaces must be designed to withstand, in unaccelerated flight at speed V_C, lateral gusts or the values prescribed for V_C in 4.4.3.

4.21.2 In addition, for level 4 airplanes, the airplane is assumed to encounter derived gusts normal to the plane of symmetry while in unaccelerated flight at V_B, V_C, V_D, and V_F. The derived gusts and airplane speeds corresponding to these conditions, as determined by 4.6 and 4.8, must be investigated. The shape of the gust must be as specified in 4.4.3.2*(1)*.

4.21.3 In the absence of a more rational analysis, the gust load must be computed as follows:

$$L_{vt} = \frac{K_{gt} U_{de} V a_{vt} S_{vt}}{498} \tag{7}$$

where:

L_{vt} = vertical surface loads (lb);
$K_g = \frac{0.88\mu_g}{5.3+\mu_{gt}}$ = gust alleviation factor;

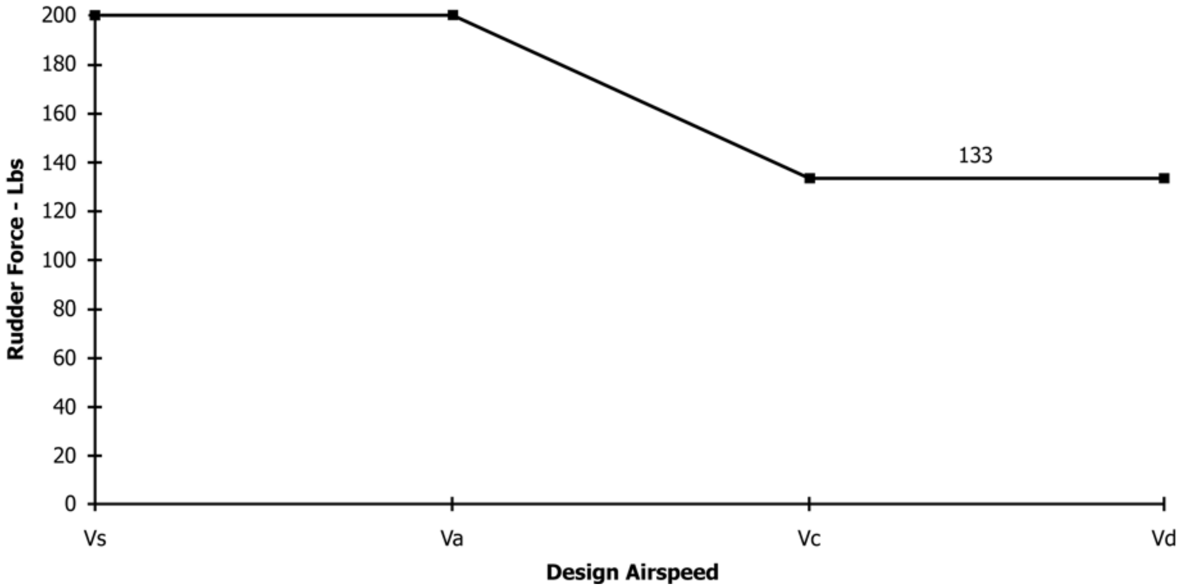


FIG. 3 Maximum Pilot Rudder Force

$\mu_{gt} = \frac{2W}{\rho \bar{c}_t g a_{vt} S_{vt}} \frac{K^2}{l_{vt}}$ = lateral mass ratio;

U_{de} = derived gust velocity (f.p.s.);

ρ = air density (slugs/ft^3);

W = the applicable weight of the airplane in the particular load case (lb);

S_{vt} = area of vertical surface (ft^2);

$\bar{c}_t$ = mean geometric chord of vertical surface (ft);

a_{vt} = lift curve slope of vertical surface (per radian);

K = radius of gyration in yaw (ft);

l_{vt} = distance from airplane c.g. to lift center of vertical surface (ft);

g = acceleration due to gravity (ft/s^2); and

V = equivalent airspeed (knots).

4.21.4 For airplanes meeting the limitations of X4.1, the average loading in Fig. X4.4 and the distribution in Fig. X4.7 of Appendix X4 may be used.

4.22 *Outboard Fins or Winglets:*

4.22.1 If outboard fins or winglets are included on the horizontal surfaces or wings, the horizontal surfaces or wings must be designed for their maximum load in combination with loads induced by the fins or winglets and moments or forces exerted on the horizontal surfaces or wings by the fins or winglets.

4.22.2 If outboard fins or winglets extend above and below the horizontal surface, the critical vertical surface loading (the load per unit area as determined under 4.20 and 4.21) must be applied to:

4.22.2.1 The part of the vertical surfaces above the horizontal surface with 80 % of that loading applied to the part below the horizontal surface; and

4.22.2.2 The part of the vertical surfaces below the horizontal surface with 80 % of that loading applied to the part above the horizontal surface.

4.22.3 The end plate effects of outboard fins or winglets must be taken into account in applying the yawing conditions of 4.20 and 4.21 to vertical surfaces in 4.22.2.

4.22.4 When rational methods are used for computing loads, the maneuvering loads of 4.20 on the vertical surfaces and the one-g horizontal surface load, including induced loads on the horizontal surface and moments or forces exerted on the horizontal surfaces by the vertical surfaces, must be applied simultaneously for the structural loading condition.

4.23 *Combined Loads on Tail Surfaces:*

4.23.1 With the airplane in a loading condition corresponding to point A or D in the V-n diagram (whichever condition leads to the higher balance load) the loads on the horizontal tail must be combined with those on the vertical tail as specified in 4.20.

4.23.2 75% of the loads according to 4.17 for the horizontal tail and 4.20 for the vertical tail must be assumed to be acting simultaneously.

4.24 *Additional Loads Applicable to V-tails*—An airplane with V-tail must be designed for a gust acting perpendicularly with respect to one of the tail surfaces at speed V_E. This case is supplemental to the equivalent horizontal and vertical tail cases specified. Mutual interference between the V-tail surfaces must be adequately accounted for.

4.25 *Ailerons:*

4.25.1 The ailerons must be designed for the loads to which they are subjected:

4.25.1.1 In the neutral position during symmetrical flight conditions; and

4.25.1.2 By the following deflections (except as limited by pilot effort), during unsymmetrical flight conditions:

(1) Sudden maximum displacement of the aileron control at V_A. Suitable allowance may be made for control system deflections.

(2) Sufficient deflection at V_C, where V_C is more than V_A, to produce a rate of roll not less than obtained in 4.25.1.2.

(3) Sufficient deflection at V_D to produce a rate of roll not less than one-third of that obtained in 4.25.1.2.

4.25.2 For airplanes meeting the limitations of X4.1, the average loading in Appendix X4, X4.3 and Fig. X4.1 of Appendix X4 and the distribution in Fig. X4.8 of Appendix X4 may be used.

4.26 *Special Devices*—The loading for special devices using aerodynamic surfaces (such as slots, slats, and spoilers) must be determined from test data.

5. Design Airspeeds

5.1 *Design Airspeeds*—Except as provided in 5.1.1.4, the selected design airspeeds are equivalent airspeeds (EAS).

5.1.1 *Design Cruising Speed, V_C*—For V_C, the following apply:

5.1.1.1 Where *W/S* = wing loading at the design maximum takeoff weight (lb/ft^2), V_C (in knots) may not be less than:

(1) $33\sqrt{W/S}$; and

(2) $36\sqrt{W/S}$ (for airplanes approved for aerobatics).

5.1.1.2 For values of *W/S* more than 20, the multiplying factors may be decreased linearly with *W/S* to a value of 28.6 where *W/S* = 100.

5.1.1.3 V_C need not be more than 0.9 V_H at sea level.

5.1.1.4 At altitudes where an M_D is established, a cruising speed M_C limited by compressibility may be selected.

5.1.2 *Design Dive Speed, V_D*—For V_D, the following apply:

5.1.2.1 V_D/M_D may not be less than 1.25 V_C/M_C: and

5.1.2.2 With V_C min, the required minimum design cruising speed, V_D (in knots) may not be less than:

(1) 1.40 V_C min; and

(2) 1.55 V_C min (for airplanes approved for aerobatics).

5.1.2.3 For values of *W/S* more than 20, the multiplying factors in 5.1.2.2 may be decreased linearly with *W/S* to a value of 1.35 where *W/S* = 100.

5.1.2.4 Compliance with 5.1.2.1 and 5.1.2.2 need not be shown if V_D/M_D is selected so that the minimum speed margin between V_C/M_C and V_D/M_D is the greater of the following:

(1) The speed increase resulting when, from the initial condition of stabilized flight at V_C/M_C, the airplane is assumed to be upset, flown for 20 s along a flight path 7.5° below the initial path, and then pulled up with a load factor of 1.5 (0.5 g. acceleration increment). At least 75 % maximum continuous

power for reciprocating engines, and maximum cruising power for turbines, or, if less, the power required for V_C/M_C for both kinds of engines, must be assumed until the pullup is initiated, at which point power reduction and pilot-controlled drag devices may be used; and either:

(2) Mach 0.05 (at altitudes where M_D is established); or

(3) Mach 0.07 for level 4 airplanes (at altitudes where M_D is established) unless a rational analysis, including the effects of automatic systems, is used to determine a lower margin. If a rational analysis is used, the minimum speed margin must be enough to provide for atmospheric variations (such as horizontal gusts), and the penetration of jet streams or cold fronts), instrument errors, airframe production variations, and must not be less than Mach 0.05.

5.1.3 *Design Maneuvering Speed V_A*—For V_A, the following applies:

5.1.3.1 V_A may not be less than $V_S\sqrt{n}$ where:

(1) V_S is a 1g computed stalling speed with flaps retracted (normally based on the maximum airplane normal force coefficients, C_{NA}) at either *(1)* the particular weight under consideration or *(2)* the design maximum takeoff weight; and

(2) n is the limit maneuvering load factor used in design.

5.1.3.2 The value of V_A need not exceed the value of V_C used in design.

5.1.4 *Design Speed for Maximum Gust Intensity, V_B*—For V_B, the following apply:

5.1.4.1 V_B may not be less than the speed determined by the intersection of the line representing the maximum positive lift, C_{NMAX}, and the line representing the rough air gust velocity on the gust V-n diagram, or $V_{S_1}\sqrt{n_g}$, whichever is less, where:

(1) n_g is the positive airplane gust load factor due to gust, at speed V_C (in accordance with 4.6), and at the particular weight under consideration; and

(2) V_{S_1} is the 1g stalling speed with the flaps retracted at the particular weight under consideration.

5.1.4.2 V_B need not be greater than V_C.

6. Engine Mount Loads

6.1 *Engine Torque:*

6.1.1 Each engine mount and its supporting structure must be designed for the effects of:

6.1.1.1 A limit engine torque corresponding to takeoff power and, if applicable, propeller speed acting simultaneously with 75 % of the limit loads from flight condition A of 4.4.4;

6.1.1.2 The limit engine torque as specified in 6.1.3 acting simultaneously with the limit loads from flight condition A of 4.4.4; and

6.1.1.3 For turbo-propeller installations, in addition to the conditions specified in 6.1.1.1 and 6.1.1.2, a limit engine torque corresponding to takeoff power and propeller speed, multiplied by a factor accounting for propeller control system malfunction, including quick feathering, acting simultaneously with 1g level flight loads. In the absence of a rational analysis, a factor of 1.6 must be used.

6.1.2 For turbine engine installations, the engine mounts and supporting structure must be designed to withstand each of the following:

6.1.2.1 A limit engine torque load imposed by sudden engine stoppage due to malfunction or structural failure (such as compressor jamming).

6.1.2.2 A limit engine torque load imposed by the maximum acceleration of the engine.

6.1.3 The limit engine torque to be considered under 6.1.1 must be obtained by multiplying the mean torque for maximum continuous power by a factor determined as follows:

6.1.3.1 1.25 for turbo-propeller installations;

6.1.3.2 For four-stroke engines:

(1) 1.33 for engines with five or more cylinders,

(2) 2, 3, 4, or 8 for engines with four, three, two, or one cylinders, respectively.

6.1.3.3 For two-stroke engines:

(1) 2 for engines with three or more cylinders,

(2) 3 or 6, for engines with two or one cylinders respectively.

6.2 *Side Load on Engine Mount:*

6.2.1 Each engine mount and its supporting structure must be designed for a limit load factor in a lateral direction, for the side load on the engine mount, of not less than:

6.2.1.1 1.33, or

6.2.1.2 One-third of the limit load factor for flight condition A.

6.2.2 The side load prescribed in 6.2.1 may be assumed to be independent of other flight conditions.

6.3 *Gyroscopic and Aerodynamic Loads:*

6.3.1 Each engine mount and its supporting structure must be designed for the gyroscopic, inertial, and aerodynamic loads that result, with the engine(s) and propeller(s), if applicable, at maximum continuous r.p.m., under either:

6.3.1.1 The conditions prescribed in 4.11 and 4.22; or

6.3.1.2 All possible combinations of the following:

(1) A yaw velocity of 2.5 radians per second;

(2) A pitch velocity of 1.0 radian per second;

(3) A normal load factor of 2.5; and

(4) Maximum continuous thrust.

6.3.2 For airplanes approved for aerobatics, each engine mount and its supporting structure must meet the requirements of 6.3.1 and be designed to withstand the load factors expected during combined maximum yaw and pitch velocities.

6.3.3 For level 4 airplanes, each engine mount and its supporting structure must meet the requirements of 6.3.1 and the gust conditions specified in 4.6.

7. Flight Control Loads

7.1 *Control Surface Loads:*

7.1.1 The control surface loads specified in 4.16 through 4.26 and 7.4 through 7.9 are assumed to occur in the conditions described in 4.3 through 4.11.

7.1.2 For airplanes meeting the limitations of X4.1 and if allowed by the following paragraphs, the values of control surface loading in Appendix X4 may be used to determine the detailed rational requirements of 4.16 through 4.26 and 7.4 through 7.9, unless these values result in unrealistic loads.

7.2 *Loads Parallel to Hinge Line:*

7.2.1 Control surfaces and supporting hinge brackets must be designed to withstand inertial loads acting parallel to the hinge line.

7.2.2 In the absence of more rational data, the inertia loads may be assumed to be equal to KW, where:

7.2.2.1 K = 24 for vertical surfaces;

7.2.2.2 K = 12 for horizontal surfaces; and

7.2.2.3 W = weight of the movable surfaces.

7.3 *Control System Loads:*

7.3.1 Each flight control system and its supporting structure must be designed for loads corresponding to at least 125 % of the computed hinge moments of the movable control surface in the conditions prescribed in 4.16 through 4.26 and 7.1 through 7.9. In addition, the following apply:

7.3.1.1 The system limit loads need not exceed the higher of the loads that can be produced by the pilot and automatic devices operating the controls. However, autopilot forces need not be added to pilot forces. The system must be designed for the maximum effort of the pilot or autopilot, whichever is higher. In addition, if the pilot and the autopilot act in opposition, the part of the system between them may be designed for the maximum effort of the one that imposes the lesser load. Pilot forces used for design need not exceed the maximum forces prescribed in 7.4.2.

7.3.1.2 The design must, in any case, provide a rugged system for service use, considering jamming, ground gusts, taxiing downwind, control inertia, and friction. Compliance with this subparagraph may be shown by designing for loads resulting from application of the minimum forces prescribed in 7.4.2.

7.3.2 A 1.25 factor on computed hinge moments must be used to design elevator, aileron, and rudder systems. However, a factor as low as 1.0 may be used if hinge moments are based on accurate flight test data, the exact reduction depending upon the accuracy and reliability of the data.

7.3.3 Pilot forces used for design are assumed to act at the appropriate control grips or pads as they would in flight, and to react at the attachments of the control system to the control surface horns.

7.3.4 For airplanes meeting the limitations of X4.1, the rudder control system must be designed to a load of 1000 N [225 lb] per pedal, acting simultaneously on both pedals in the forward direction.

7.4 *Limit Control Forces and Torques:*

7.4.1 In the control surface flight loading condition, the air loads on movable surfaces and the corresponding deflections need not exceed those that would result in flight from the application of any pilot force within the ranges specified in 7.4.2. In applying this criterion, the effects of control system boost and servo-mechanisms, and the effects of tabs must be considered. The automatic pilot effort must be used for design if it alone can produce higher control surface loads than the human pilot.

7.4.2 The limit pilot forces and torques are as follows:

Control	Maximum forces or torques for design maximum takeoff weight, W, equal to or less than 2268 kg [5000 lb]^{A}	Minimum forces or torques^{B}
Aileron:		
Stick	298 N [67 lb]	178 N [40 lb]
Wheel^{C}	222 D Nm [50 D in. lb]^{D}	178 D Nm [40 D in.- lb]^{D}
Elevator:		
Stick	743 N [167 lb]	445 N [100 lb]
Wheel (symmetrical)	890 N [200 lb]	445 N [100 lb]
Wheel (unsymmetrical)^{E}		445 N [100 lb]
Rudder	890 N [200 lb]	667 N [150 lb]

[A] For design maximum takeoff weight (W) more than 2268 kg [5000 lb], the specified maximum values must be increased linearly with weight to 1.35 times the specified values at a design maximum takeoff weight of 8618 kg [19 000 lb].

[B] If the design of any individual set of control systems or surfaces makes these specified minimum forces or torques inapplicable, values corresponding to the present hinge moments obtained under 7.9, but not less than 0.6 of the specified minimum forces or torques, may be used.

[C] The critical parts of the aileron control system must also be designed for a single tangential force with a limit value of 1.25 times the couple force determined from the above criteria.

[D] D = wheel diameter (meters [inches]).

[E] The unsymmetrical force must be applied at one of the normal handgrip points on the control wheel.

7.5 *Dual Control System:*

7.5.1 Each dual control system must be designed to withstand the force of the pilots operating in opposition, using individual pilot forces not less than the greater of:

7.5.1.1 0.75 times those obtained under 7.3; or

7.5.1.2 The minimum forces specified in 7.4.2.

7.5.2 Each dual control system must be designed to withstand the force of the pilots applied together, in the same direction, using individual pilot forces not less than 0.75 times those obtained under 7.3.

7.6 *Secondary Control System*—Secondary controls, such as wheel brakes, spoilers, and tab controls, must be designed for the maximum forces that a pilot is likely to apply to those controls.

7.7 *Trim Tab Effects*—The effects of trim tabs on the control surface design conditions must be accounted for only where the surface loads are limited by maximum pilot effort. In these cases, the tabs are considered to be deflected in the direction that would assist the pilot. These deflections must correspond to the maximum degree of "out of trim" expected at the speed for the condition under consideration.

7.8 *Tabs*—Control surface tabs must be designed for the most severe combination of airspeed and tab deflection likely to be obtained within the flight envelope for any usable loading condition.

7.9 *Ground Gust Conditions:*

7.9.1 The control system must be investigated as follows for control surface loads due to ground gusts and taxiing downwind:

7.9.1.1 If an investigation of the control system for ground gust loads is not required by 7.9.1.2, but the applicant elects to design a part of the control system for these loads, these loads need only be carried from control surface horns through the nearest stops or gust locks and their supporting structures.

7.9.1.2 If pilot forces less than the minimums specified in 7.4.2 are used for design, the effects of surface loads due to ground gusts and taxiing downwind must be investigated for the entire control system according to the formula:

$$H = K\ c\ S\ q \qquad (8)$$

where:

H = limit hinge moment (ft.-lb);

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

S = area of control surface aft of the hinge line (sq. ft.);

q = dynamic pressure (p.s.f.) based on a design speed not less than $14.6\sqrt{W/S} + 14.6$ (f.p.s) where W/S = wing loading at design maximum weight, except that the design speed need not exceed 88 (f.p.s.); and

K = limit hinge moment factor for ground gusts derived in 7.9.2. (For ailerons and elevators, a positive value of K indicates a moment tending to depress the surface and a negative value of K indicates a moment tending to raise the surface).

7.9.2 The limit hinge moment factor K for ground gusts must be derived as follows:

Surface	K	Position of controls
(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) Elevator	±0.75	(c) Elevator full up (–)
(d) Elevator		(d) Elevator full down (+)
(e) Rudder	±0.75	(e) Rudder in neutral
(f) Rudder		(f) Rudder at full throw

7.9.3 At all weights between the empty weight and the maximum weight declared for tie-down stated in the appropriate manual, any declared tie-down points and surrounding structure, control system, surfaces and associated gust locks, must be designed to withstand the limit load conditions that exist when the airplane is tied down and that result from wind speeds up to 65 knots horizontally from any direction.

8. Ground Loads

8.1 *General*—The limit ground loads specified in this subpart are considered to be external loads and inertia forces that act upon an airplane structure. In each specified ground load condition, the external reactions must be placed in equilibrium with the linear and angular inertia forces in a rational or conservative manner.

8.2 *Ground Load Conditions and Assumptions:*

8.2.1 The ground load requirements of this subpart must be complied with at the design maximum weight except that 8.4, 8.5, and 8.6 may be complied with at a design landing weight (the highest weight for landing conditions at the maximum descent velocity) allowed under 8.2.2 and 8.2.3.

8.2.2 The design landing weight may be as low as:

8.2.2.1 95 % of the maximum weight if the minimum fuel capacity is enough for at least one-half hour of operation at maximum continuous power plus a capacity equal to a fuel weight which is the difference between the design maximum weight and the design landing weight; or

8.2.2.2 The design maximum weight less the weight of 25 % of the total fuel capacity.

8.2.3 The design landing weight of a multi-engine airplane may be less than that allowed under 8.2.2 if:

8.2.3.1 The airplane meets the one-engine-inoperative climb requirements of 14 CFR Part 23, Sec. 23.67 (b)(1) or (c) and

8.2.3.2 Compliance is shown with the fuel jettisoning system requirements of 14 CFR Part 23, Sec. 23.1001.

8.2.4 The selected limit vertical inertia load factor at the center of gravity of the airplane for the ground load conditions prescribed in this subpart may not be less than that which would be obtained when landing with a descent velocity (V), in feet per second equal to 4.4 $(W/S)^{1/4}$, except that this velocity need not be more than 10 ft/s and may not be less than 7 ft/s.

8.2.5 Airplane lift not exceeding two-thirds of the weight of the airplane may be assumed to exist throughout the landing impact and to act through the center of gravity. The ground reaction load factor may be equal to the inertia load factor minus the ratio of the above assumed wing lift to the airplane weight.

8.2.6 If energy absorption tests are made to determine the limit load factor corresponding to the required limit descent velocities, these tests must be made under 14 CFR Part 23, Sec. 23.723 (a).

8.2.7 No inertia load factor used for design purposes may be less than 2.67, nor may the limit ground reaction n load factor be less than 2.0 at design maximum weight, unless these lower values will not be exceeded in taxiing at speeds up to takeoff speed over terrain as rough as that expected in service.

8.3 *Landing Gear Arrangement*—Sections 8.4 through 8.6, or the conditions in Appendix X5, apply to airplanes with conventional arrangements of main and nose gear, or main and tail gear.

8.4 *Level Landing Conditions:*

8.4.1 For a level landing, the airplane is assumed to be in the following attitudes:

8.4.1.1 For airplanes with tail wheels, a normal level flight attitude.

8.4.1.2 For airplanes with nose wheels, attitudes in which:

(1) The nose and main wheels contact the ground simultaneously; and

(2) The main wheels contact the ground and the nose wheel is just clear of the ground.

8.4.1.3 The attitude used in 8.4.1.2*(1)* of this section may be used in the analysis required under 8.4.1.2*(2)*.

8.4.2 When investigating landing conditions, the drag components simulating the forces required to accelerate the tires and wheels up to the landing speed (spin-up) must be properly combined with the corresponding instantaneous vertical ground reactions, and the forward-acting horizontal loads resulting from rapid reduction of the spin-up drag loads (spring-back) must be combined with vertical ground reactions at the instant of the peak forward load, assuming wing lift and a tire-sliding coefficient of friction of 0.8. However, the drag loads may not be less than 25 % of the maximum vertical ground reactions (neglecting wing lift).

8.4.3 In the absence of specific tests or a more rational analysis for determining the wheel spin-up and spring-back loads for landing conditions, the method set forth in Appendix X6 of this part must be used. If Appendix X6 of this part is used, the drag components used for design must not be less than those given by Appendix X5 of this part.

8.4.4 For airplanes with tip tanks or large overhung masses (such as turbo-propeller or jet engines) supported by the wing, the tip tanks and the structure supporting the tanks or overhung masses must be designed for the effects of dynamic responses under the level landing conditions of either 8.4.1.1 or 8.4.1.2*(2)*. In evaluating the effects of dynamic response, an airplane lift equal to the weight of the airplane may be assumed.

8.5 *Tail Down Landing Conditions:*

8.5.1 For a tail down landing, the airplane is assumed to be in the following attitudes:

8.5.1.1 For airplanes with tail wheels, an attitude in which the main and tail wheels contact the ground simultaneously.

8.5.1.2 For airplanes with nose wheels, a stalling attitude, or the maximum angle allowing ground clearance by each part of the airplane, whichever is less.

8.5.2 For airplanes with either tail or nose wheels, ground reactions are assumed to be vertical, with the wheels up to speed before the maximum vertical load is attained.

8.6 *One-Wheel Landing Conditions*—For the one-wheel landing condition, the airplane is assumed to be in the level attitude and to contact the ground on one side of the main landing gear. In this attitude, the ground reactions must be the same as those obtained on that side under 8.4.

8.7 *Side Load Conditions:*

8.7.1 For the side load condition, the airplane is assumed to be in a level attitude with only the main wheels contacting the ground and with the shock absorbers and tires in their static positions.

8.7.2 The limit vertical load factor must be 1.33, with the vertical ground reaction divided equally between the main wheels.

8.7.3 The limit side inertia factor must be 0.83, with the side ground reaction divided between the main wheels so that:

8.7.3.1 0.5 (W) is acting inboard on one side; and

8.7.3.2 0.33 (W) is acting outboard on the other side.

8.7.4 The side loads prescribed in 8.7.3 are assumed to be applied at the ground contact point and the drag loads may be assumed to be zero.

8.8 *Braked Roll Conditions*—Under braked roll conditions, with the shock absorbers and tires in their static positions, the following apply:

8.8.1 The limit vertical load factor must be 1.33.

8.8.2 The attitudes and ground contacts must be those described in 8.4 for level landings.

8.8.3 A drag reaction equal to the vertical reaction at the wheel multiplied by a coefficient of friction of 0.8 must be applied at the ground contact point of each wheel with brakes, except that the drag reaction need not exceed the maximum value based on limiting brake torque.

8.9 *Ground Loads—Supplementary Conditions for Tail Wheels*—In determining the ground loads on the tail wheel and affected supporting structures, the following apply:

8.9.1 For the obstruction load, the limit ground reaction obtained in the tail down landing condition is assumed to act up and aft through the axle at 45°. The shock absorber and tire may be assumed to be in their static positions.

8.9.2 For the 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, is assumed. In addition:

8.9.2.1 If a swivel is used, the tail wheel is assumed to be swiveled 90° to the airplane longitudinal axis with the resultant ground load passing through the axle;

8.9.2.2 If a lock, steering device, or shimmy damper is used, the tail wheel is also assumed to be in the trailing position with the side load acting at the ground contact point; and

8.9.2.3 The shock absorber and tire are assumed to be in their static positions.

8.9.3 If a tail wheel, bumper, or an energy absorption device is provided to show compliance with 14 CFR Part 23, Sec. 23.925 (b), the following applies:

8.9.3.1 Suitable design loads must be established for the tail wheel, bumper, or energy absorption device; and

8.9.3.2 The supporting structure of the tail wheel, bumper, or energy absorption device must be designed to withstand the loads established in 8.9.3.1.

8.10 *Supplementary Conditions for Nose Wheels*—In determining the ground loads on nose wheels and affected supporting structures, and assuming that the shock absorbers and tires are in their static positions, the following conditions must be met:

8.10.1 For aft loads, the limit force components at the axle must be:

8.10.1.1 A vertical component of 2.25 times the static load on the wheel; and

8.10.1.2 A drag component of 0.8 times the vertical load.

8.10.2 For forward loads, the limit force components at the axle must be:

8.10.2.1 A vertical component of 2.25 times the static load on the wheel; and

8.10.2.2 A forward component of 0.4 times the vertical load.

8.10.3 For side loads, the limit force components at ground contact must be:

8.10.3.1 A vertical component of 2.25 times the static load on the wheel; and

8.10.3.2 A side component of 0.7 times the vertical load.

8.10.4 For airplanes with a steerable nose wheel that is controlled by hydraulic or other power, at design takeoff weight with the nose wheel in any steerable position, the application of 1.33 times the full steering torque combined with a vertical reaction equal to 1.33 times the maximum static reaction on the nose gear must be assumed. However, if a torque limiting device is installed, the steering torque can be reduced to the maximum value allowed by that device.

8.10.5 For airplanes with a steerable nose wheel that has a direct mechanical connection to the rudder pedals, the mechanism must be designed to withstand the steering torque for the maximum pilot forces specified in 7.4.2.

8.11 *Supplementary Conditions for Skiplanes*—In determining ground loads for skiplanes, and assuming that the airplane is resting on the ground with one main ski frozen at rest and the other skis free to slide, a limit side force equal to 0.036 times the design maximum weight must be applied near the tail assembly, with a factor of safety of 1.

8.12 *Jacking Loads:*

8.12.1 The airplane must be designed for the loads developed when it is supported on jacks at the design maximum weight assuming the following load factors for landing gear jacking points at a three-point attitude and for primary flight structure jacking points in the level attitude:

8.12.1.1 Vertical-load factor of 1.35 times the static reactions.

8.12.1.2 Fore, aft, and lateral load factors of 0.4 times the vertical static reactions.

8.12.2 The horizontal loads at the jack points must be reacted by inertia forces so as to result in no change in the direction of the resultant loads at the jack points.

8.12.3 The horizontal loads must be considered in all combinations with the vertical load.

8.13 *Towing Loads*—The towing loads of this section must be applied to the design of tow fittings and their immediate attaching structure.

8.13.1 The towing loads specified in 8.13.4 must be considered separately. These loads must be applied at the towing fittings and must act parallel to the ground. In addition:

8.13.1.1 A vertical load factor equal to 1.0 must be considered acting at the center of gravity; and

8.13.1.2 The shock struts and tires must be in their static positions.

8.13.2 For towing points not on the landing gear but near the plane of symmetry of the airplane, the drag and side tow load components specified for the auxiliary gear apply. For towing points located outboard of the main gear, the drag and side tow load components specified for the main gear apply. Where the specified angle of swivel cannot be reached, the maximum obtainable angle must be used.

8.13.3 The towing loads specified in 8.13.4 must be reacted as follows:

8.13.3.1 The side component of the towing load at the main gear must be reacted by a side force at the static ground line of the wheel to which the load is applied.

8.13.3.2 The towing loads at the auxiliary gear and the drag components of the towing loads at the main gear must be reacted as follows:

(1) A reaction with a maximum value equal to the vertical reaction must be applied at the axle of the wheel to which the load is applied. Enough airplane inertia to achieve equilibrium must be applied.

(2) The loads must be reacted by airplane inertia.

8.13.4 The prescribed towing loads are as follows, where W is the design maximum weight:

Tow Point	Position	Load		
		Magnitude	No.	Direction
Main Gear		0.225W per main gear unit	1.	Forward, parallel to drag axis
			2.	Forward, at 30° to drag axis
			3.	Aft, parallel to drag axis
			4.	Aft, at 30° to drag axis
Auxiliary Gear	Swiveled forward	0.3W	5.	Forward
			6.	Aft
	Swiveled aft	0.3W	7.	Forward
			8.	Aft
	Swiveled 45° from fwd	0.15W	9.	Forward, in plane of wheel
			10.	Aft, in plane of wheel
	Swiveled 45° from aft	0.15W	11.	Forward, in plane of wheel
			12.	Aft, in plane of wheel

8.14 *Ground Loads*—Ground load; unsymmetrical loads on multiple-wheel:

8.14.1 *Pivoting Loads*—The airplane is assumed to pivot about on side on the main gear with:

8.14.1.1 The brakes on the pivoting unit locked; and

8.14.1.2 Loads corresponding to a limit vertical load factor of 1, and coefficient of friction of 0.8, applied to the main gear and its supporting structure.

8.14.2 *Unequal Tire Loads*—The loads established under 8.1 through 8.6 must be applied in turn, in a 60/40 % distribution, to the dual wheels and tires in each dual wheel landing gear unit.

8.14.3 *Deflated Tire Loads*—For the deflated tire condition:

8.14.3.1 60 % of the loads established under 8.1 through 8.6 must be applied in turn to each wheel in a landing gear unit; and

8.14.3.2 60 % of the limit drag and side loads, and 100 % of the limit vertical load established under 8.7 and 8.8 or lesser vertical load obtained under 8.14.3.1, must be applied in turn to each wheel in the dual wheel landing gear unit.

9. Water Loads

9.1 *Water Load Conditions:*

9.1.1 The structure of seaplanes and amphibians must be designed for water loads developed during takeoff 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.

9.1.2 Unless sufficient satisfactory service experience is available, a rational analysis of the water loads, or the methods specified in Appendix X7 may be used.

9.1.3 Each seaplane main float must meet the requirements of this section.

APPENDIXES

(Nonmandatory Information)

X1. SIMPLIFIED DESIGN LOAD CRITERIA

X1.1 Limitations

X1.1.1 The methods provided in this appendix provide one possible means (but not the only possible means) of compliance and can only be applied to level 1 and level 2 airplanes.

X1.1.2 These methods may be applied to airplanes meeting the following limitations without further justification:

X1.1.2.1 A single engine excluding turbine powerplants.

X1.1.2.2 A main wing located closer to the airplane's center of gravity than to the aft, fuselage-mounted, empennage.

X1.1.2.3 A main wing that contains a quarter-chord sweep angle of not more than 15° fore or aft.

X1.1.2.4 A main wing that is equipped with trailing-edge controls (ailerons or flaps, or both).

X1.1.2.5 A main wing aspect ratio not greater than 7.0.

X1.1.2.6 A main wing that does not have winglets, outboard fins, or other wingtip devices.

X1.1.2.7 A horizontal tail aspect ratio not greater than 4.0.

X1.1.2.8 A horizontal tail volume coefficient not less than 0.34.

X1.1.2.9 A vertical tail aspect ratio not greater than 2.0.

X1.1.2.10 A vertical tail planform area not greater than 10 % of the wing planform area.

X1.1.2.11 Horizontal and vertical tail airfoil sections must both be symmetrical.

X1.1.3 This appendix may be used outside of the limitations in X1.1.2 when evidence can be provided that the method provides safe and reliable results.

X1.1.4 Airplanes with any of the following design features shall not use this appendix:

X1.1.4.1 Canard, tandem-wing, close-coupled, or tailless arrangements of the lifting surfaces.

X1.1.4.2 Biplane or multiplane wing arrangements.

X1.1.4.3 V-tail or any arrangement where the horizontal stabilizer is supported by the vertical stabilizer (T-tail, cruciform-tail (+), etc.).

X1.1.4.4 Wings with slatted lifting surfaces.

X1.1.4.5 Full-flying stabilizing surfaces (horizontal and vertical).

X1.2 Abbreviations

n_1 = Airplane Positive Maneuvering Limit Load Factor
n_2 = Airplane Negative Maneuvering Limit Load Factor
n_3 = Airplane Positive Gust Limit Load Factor at V_C
n_4 = Airplane Negative 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 = $11.0\sqrt{n_1 W/S}$ knots
$V_{A\ min}$ = Minimum Design Maneuvering Speed = $15.0\sqrt{n_1 W/S}$ knots (however this need not exceed VC used in design)
$V_{C\ min}$ = Minimum Design Cruising Speed = $17.0\sqrt{n_1 W/S}$ knots (however this need not exceed 0.9VH, see X1.3.5(b))
$V_{D\ min}$ = Minimum Design Dive Speed = $24.0\sqrt{n_1 W/S}$ knots (however this need not exceed $1.4V_{C\ \min}\sqrt{n_1/3.8}$)

X1.3 Flight Loads

X1.3.1 Each flight load may be considered independent of altitude and, except for the local supporting structure for dead weight items, only the maximum design weight conditions must be investigated.

X1.3.2 Table X1.1 must be used to determine values of n_1, n_2, n_3, and n_4, corresponding to the maximum design weights. Fig. X1.1 presents a generalized flight envelope.

X1.3.3 Figs. X1.2 and X1.3 must be used to determine values of n_3 and n_4, corresponding to the minimum flying weights, and, if these load factors are greater than the load factors at the design weight, the supporting structure for dead weight items must be substantiated for the resulting higher load factors.

X1.3.4 Each specified wing and tail loading is independent of the center of gravity range. However, a c.g. range must be selected for the airplane and the basic fuselage structure must be investigated for the most adverse dead weight loading conditions for the c.g. range selected.

X1.3.5 The following loads and loading conditions are the minimums for which strength must be provided in the structure:

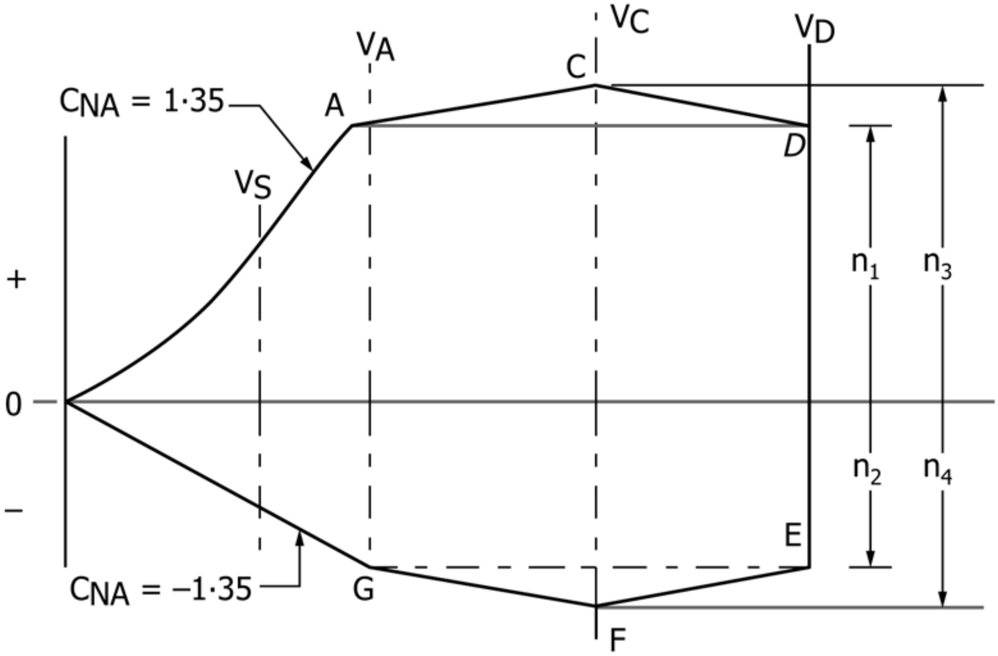
X1.3.5.1 *Airplane Equilibrium*—The aerodynamic wing loads may be considered to act normal to the relative wind, and to have a magnitude of 1.05 times the airplane normal loads (as determined from X1.4.2 and X1.4.3) for the positive flight conditions and a magnitude equal to the airplane normal loads for the negative conditions. Each chordwise and normal component of this wing load must be considered.

X1.3.5.2 *Minimum Design Airspeeds*—The minimum design airspeeds may not be less than the minimum speeds found in X1.2. In addition, $V_{C\ min}$ need not exceed values of $0.9V_H$ actually obtained at sea level for the lowest design weight for

TABLE X1.1 Minimum Design Limit Flight Load Factors

Flight Load Factors		Not Approved for Aerobatics	Approved for Aerobatics
Flaps Up	n_1	3.8	6.0
	n_2	$-0.5n_1$	
	n_3	Find n_3 from Fig. X1.2	
	n_4	Find n_4 from Fig. X1.3	
Flaps Down	n_{flap}	$0.5n_1$	
	n_{flap}	Zero[A]	

[A] Vertical wing load may be assumed equal to zero and only the flap part of the wing need be checked for this condition.



NOTE 1—Conditions "C" or "F" need only be investigated when n_3W/S or n_4W/S is greater than n_1W/S or n_2W/S respectively.

NOTE 2—Condition "G" need not be investigated when the supplementary condition specified in 4.14 is investigated.

FIG. X1.1 Generalized Flight Envelope

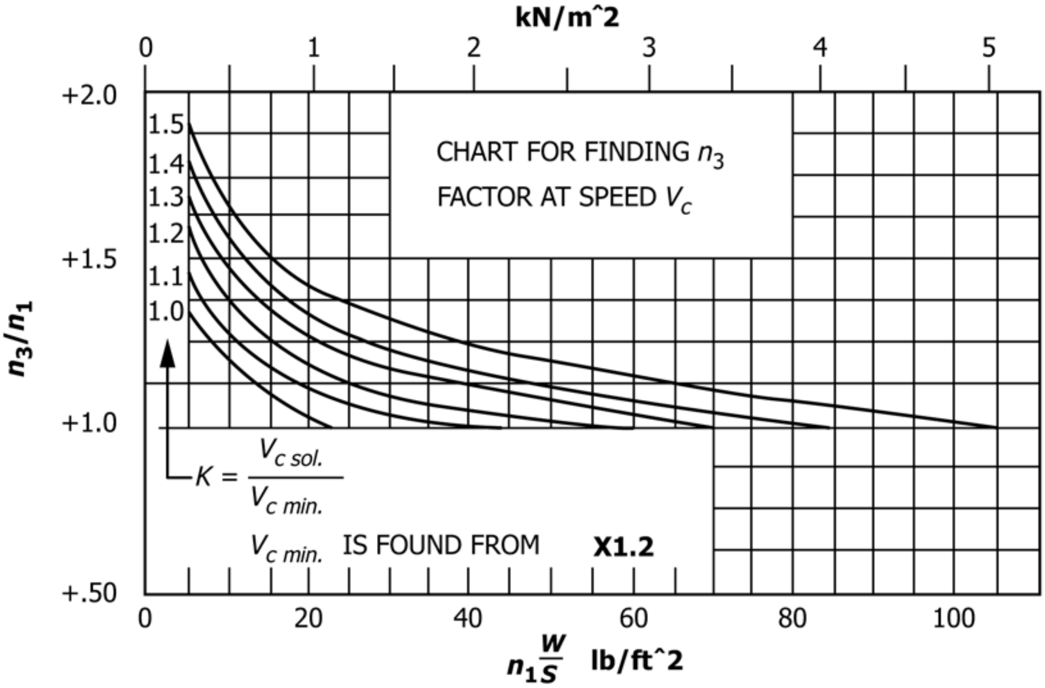


FIG. X1.2 Chart for Finding n_3 Factor at Speed V_C

which certification is desired. In computing these minimum design airspeeds, n_1 may not be less than 3.8.

X1.3.5.3 *Flight Load Factor*—The limit flight load factors specified in Table X1.1 represent the ratio of the aerodynamic force component (acting normal to the assumed longitudinal axis of the airplane) to the weight of the airplane. A positive flight load factor is an aerodynamic force acting upward, with respect to the airplane.

X1.4 Flight Conditions

X1.4.1 *General*—Each design condition in X1.4.2 and X1.4.4 must be used to assure sufficient strength for each

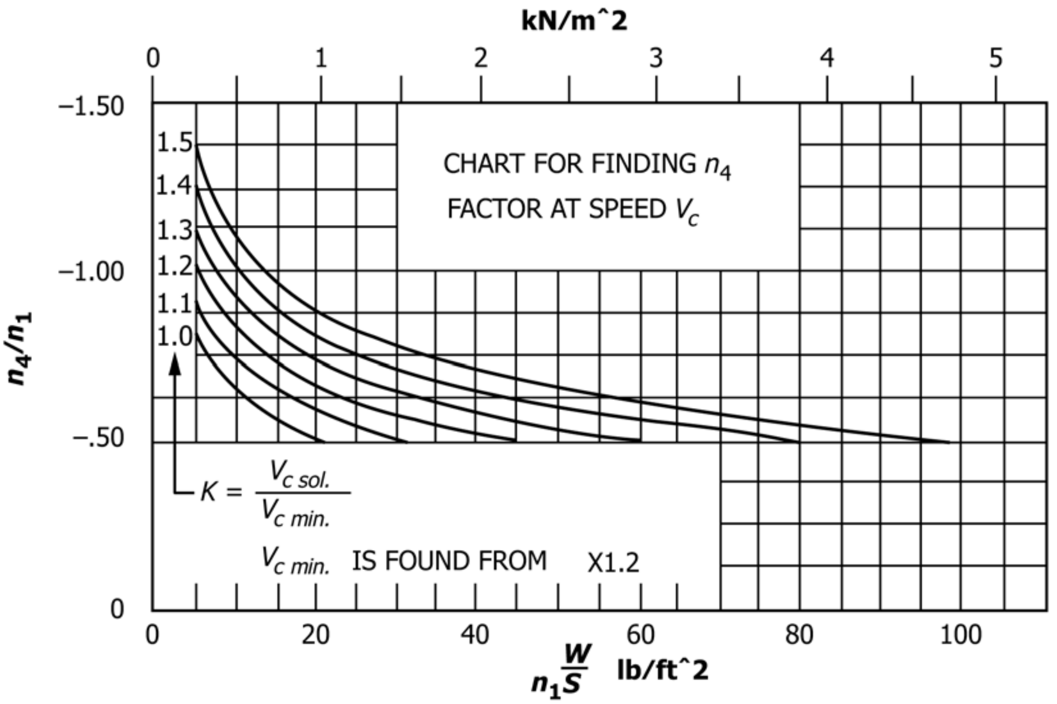


FIG. X1.3 Chart for Finding n_4 Factor at Speed V_C

condition of speed and load factor on or within the boundary of a V-n diagram for the airplane similar to the diagram in Fig. X1.1. This diagram must also be used to determine the airplane structural operating limitations as specified in 14 CFR Part 23, Sec. 23.1501 (c) through 23.1513 and 23.1519.

X1.4.2 *Symmetrical Flight Conditions*—The airplane must be designed for symmetrical flight conditions as follows:

X1.4.2.1 The airplane must be designed for at least the four basic flight conditions, "A", "D", "E", and "G" as noted on the flight envelope of Fig. X1.1. In addition, the following requirements apply:

(1) The design limit flight load factors corresponding to conditions "D" and "E" of Fig. X1.1 must be at least as great as those specified in Table X1.1 and Fig. X1.1, and the design speed for these conditions must be at least equal to the value of $V_{D\ min}$ from X1.2.

(2) For conditions "A" and "G" of Fig. X1.1, the load factors must correspond to those specified in Table X1.1, and the design speeds must be computed using these load factors with the maximum static lift coefficient C_{NA} determined by the applicant. However, in the absence of more precise computations, these latter conditions may be based on a value of $C_{NA} = \pm 1.35$ and the design speed for condition "A" may be less than $V_{A\ min}$.

(3) Conditions "C" and "F" of Fig. X1.1 need only be investigated when n_3W/S or n_4W/S are greater than n_1W/S or n_2W/S, respectively.

X1.4.2.2 If flaps or other high lift devices intended for use at the relatively low airspeed of approach, landing, and takeoff, are installed, the airplane must be designed for the two flight conditions corresponding to the values of limit flap-down factors specified in Table X1.1 with the flaps fully extended at not less than the design flap speed $V_{F\ min}$ from X1.2.

X1.4.3 *Unsymmetrical Flight Conditions*—Each affected structure must be designed for unsymmetrical loadings as follows:

X1.4.3.1 The aft fuselage-to-wing attachment must be designed for the critical vertical surface load determined in accordance with X2.2.3.1 and X2.2.3.2.

X1.4.3.2 The wing and wing carry-through structures must be designed for 100 % of condition "A" loading on one side of the plane of symmetry and 70 % on the opposite side, or 60 % on the opposite side for airplanes approved for aerobatics.

X1.4.3.3 The wing and wing carry-through structures must 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 and 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, C_{mo}, modified over the aileron portion of the span, must be computed as follows:

(1) $C_m = C_{mo} + 0.01\ \delta_u$ (up aileron side) wing basic airfoil.

(2) $C_m = C_{mo} - 0.01\ \delta_d$ (down aileron side) wing basic airfoil, where δ_u is the up aileron deflection and δ_d is the down aileron deflection.

X1.4.3.4 $\Delta_{critical}$, which is the sum of $\delta_u + \delta_d$, must be computed as follows:

(1) Compute Δ_a and Δ_b from the formulas:

$$\Delta_a = \frac{V_A}{V_C} \times \Delta_p \qquad (X1.1)$$

$$\Delta_b = 0.5 \frac{V_A}{V_D} \times \Delta_p \qquad (X1.2)$$

where:

Δ_p = the maximum total deflection (sum of both aileron deflections) at V_A with V_A, V_C, and V_D described in X1.3.5.2.

(2) Compute *K* from the formula:

$$K = \frac{(C_{mo} - 0.01\ \delta_b)V_D^2}{(C_{mo} - 0.01\ \delta_a)V_C^2} \qquad (X1.3)$$

where:

δ_a = the down aileron deflection corresponding to Δ_a, and

δ_b = the down aileron deflection corresponding to Δ_b as computed in X1.4.3.4*(1)*.

(3) If *K* is less than 1.0, Δ_a is $\Delta_{critical}$ and must be used to determine δ_u and δ_d. In this case, V_C is the critical speed which must be used in computing the wing torsion loads over the aileron span.

(4) If *K* is equal to or greater than 1.0, Δ_b is $\Delta_{critical}$ and must be used to determine δ_u and δ_d. In this case, V_D is the critical speed which must be used in computing the wing torsion loads over the aileron span.

X1.4.4 *Supplementary Conditions; Rear Lift Truss; Engine Torque; Side Load on Engine Mount*—Each of the following supplementary conditions must be investigated:

X1.4.4.1 In designing the rear lift truss, the special condition specified in 4.14 may be investigated instead of condition "G" of Fig. X1.1.

X1.4.4.2 Each engine mount and its supporting structures must be designed for:

(1) The maximum limit torque corresponding to maximum take-off power (MTO Power) and propeller speed acting simultaneously with 75% of the limit loads resulting from the maximum positive maneuvering flight load factor n_1.

(2) The maximum limit torque corresponding to MCP (maximum continuous power) and propeller speed acting simultaneously with the limit loads resulting from the maximum positive maneuvering flight load factor n_1; and

(3) The maximum limit torque must be obtained by multiplying the mean torque by a factor of 1.33 for engines with five or more cylinders. For 4, 3, and 2 cylinder engines, the factor must be 2, 3, and 4, respectively.

X1.4.4.3 Each engine mount and its supporting structure must be designed for the loads resulting from a lateral limit load factor of not less than 1.47, or 2.0 for airplanes approved for aerobatics.

X2. ACCEPTABLE METHODS FOR CONTROL SURFACE LOADS CALCULATIONS

X2.1 Limitations

X2.1.1 The methods provided in this appendix provide one possible means (but not the only possible means) of compliance and can only be applied to level 1 and level 2 airplanes.

X2.1.2 These methods may be applied to airplanes meeting the following limitations without further justification:

X2.1.2.1 A leading edge sweep angle (of the control surface) of not more than 15° fore or aft.

X2.1.2.2 Horizontal and vertical tail airfoil sections must both be symmetrical.

X2.1.2.3 For ailerons and flaps, a main wing that does not have winglets, outboard fins, or other wingtip devices.

X2.1.3 This appendix may be used outside of the limitations in X2.1.2 when evidence can be provided that the method provides safe and reliable results.

X2.1.4 Airplanes with any of the following design features shall not use this appendix:

X2.1.4.1 For flaps and ailerons, biplane or multiplane wing arrangements.

X2.1.4.2 Stabilizers and control surfaces on V-tail arrangements.

X2.1.4.3 For vertical stabilizer, any tail arrangement where the horizontal stabilizer is supported by the vertical stabilizer (T-tail, cruciform-tail (+), etc.).

X2.1.4.4 For flaps and ailerons, wings with delta planforms.

X2.1.4.5 On surfaces and their associated control surface which employ slatted lifting devices.

X2.1.4.6 Full-flying stabilizing surfaces (horizontal and vertical).

X2.2 Control Surface Loads

X2.2.1 *General*—Each control surface load must be determined using the criteria of X2.2.2 and must lie within the simplified loadings of X2.2.3.

X2.2.2 *Limit Pilot Forces*—In each control surface loading condition described in X2.2.3 through X2.2.5, the air loads on the movable surfaces and the corresponding deflections need not exceed those which could be obtained in flight by employing the maximum limit pilot forces specified in the table in 7.4.2. If the surface loads are limited by these maximum limit pilot forces, the tabs must either be considered to be deflected to their maximum travel in the direction which would assist the pilot or the deflection must correspond to the maximum degree of "out of trim" expected at the speed for the condition under consideration. The tab load, however, need not exceed the value specified in Table X2.1.

X2.2.3 *Surface Loading Conditions*—Each surface loading condition must be investigated as follows:

X2.2.3.1 Simplified limit surface loadings and distributions for the horizontal tail, vertical tail, aileron, wing flaps, and trim tabs are specified in Table X2.1 and Figs. X2.1 and X2.2. If more than one distribution is given, each distribution must be investigated. Fig. X2.1 is limited to use with vertical tails with aspect ratios less than 2.5 and horizontal tails with aspect ratios less than 5 and tail volumes greater than 0.4.

TABLE X2.1 Average Limit Control Surface Loading

Surface	Direction of Loading	Magnitude of Loading	Chordwise Distribution
Horizontal Tail I	(a) Up and Down (b) Unsymmetrical loading (Up and Down)	Fig. X2.1 Curve (2) 100% w̄ on one side, 65% w̄ on other side of airplane center line. For airplanes approved for aerobatics, see X2.2.3.	Fig. X2.3
Vertical Tail II	Right and Left	Fig. X2.1 Curve (1)	Same as above
Aileron III	(a) Up and Down	Fig. X2.2 Curve (5)	
Wing Flap IV	(a) Up (b) Down	Fig. X2.2 Curve (4) 0.25 × Up Load (a)	
Trim Tab V	(a) Up and Down	Fig. X2.2 Curve (3)	Same as (D) above

NOTE 1—The surface loading of I, II, III, and V above are based on speeds $V_{A\ min}$ and $V_{C\ min}$. The loading of IV is based on $V_{F\ min}$. If values of speed greater than these minimums are selected for design, the appropriate surface loadings must be multiplied by the ratio $\left[\frac{V_{selected}}{V_{minimum}}\right]^2$. For conditions I, II, III, and V the multiplying factor used must be the higher of $\left[\frac{V_{A\ selected}}{V_{A\ minimum}}\right]^2$ or $\left[\frac{V_{C\ selected}}{V_{C\ minimum}}\right]^2$.

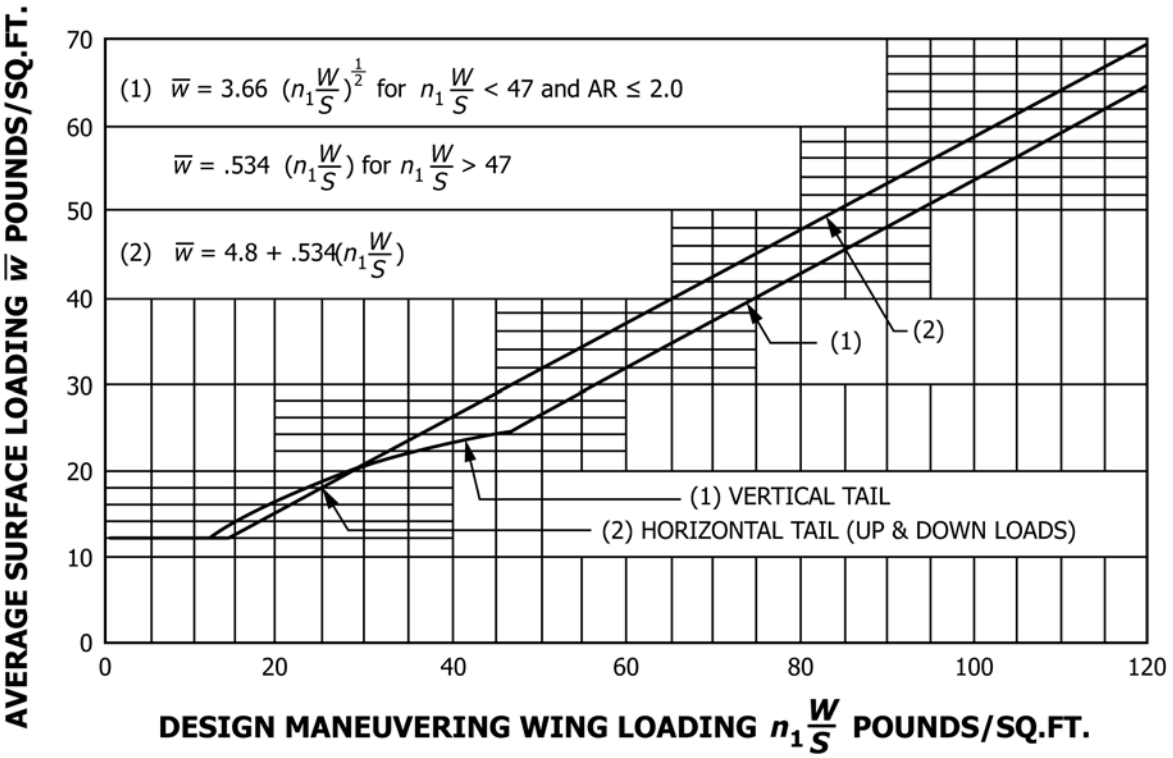


FIG. X2.1 Average Limit Control Surface Loading

(1) The distribution of load along the span of the surface, irrespective of the chordwise load distribution, must be assumed proportional to the total chord, except on horn balanced surfaces.

(2) The load on the stabilizer and elevator, and the load on fin and rudder, must be distributed chordwise as shown in Fig. X2.3.

(3) In order to ensure adequate torsional strength and also to cover maneuvers and gusts, the most severe loads must be considered in association with every center of pressure position between the leading edge and the half chord of the mean chord of the surface (stabilizer and elevator, or fin and rudder).

(4) To ensure adequate strength under high leading edge loads, the most severe stabilizer and fin loads must be further considered as being increased by 50 % over the leading 10 % of the chord with the loads aft of this appropriately decreased to retain the same total load.

(5) The most severe elevator and rudder loads should be further considered as being distributed parabolically from three times the mean loading of the surface (stabilizer and elevator, or fin and rudder) at the leading edge of the elevator and rudder, respectively, to zero at the trailing edge according to the following equation (see Fig. X2.4).

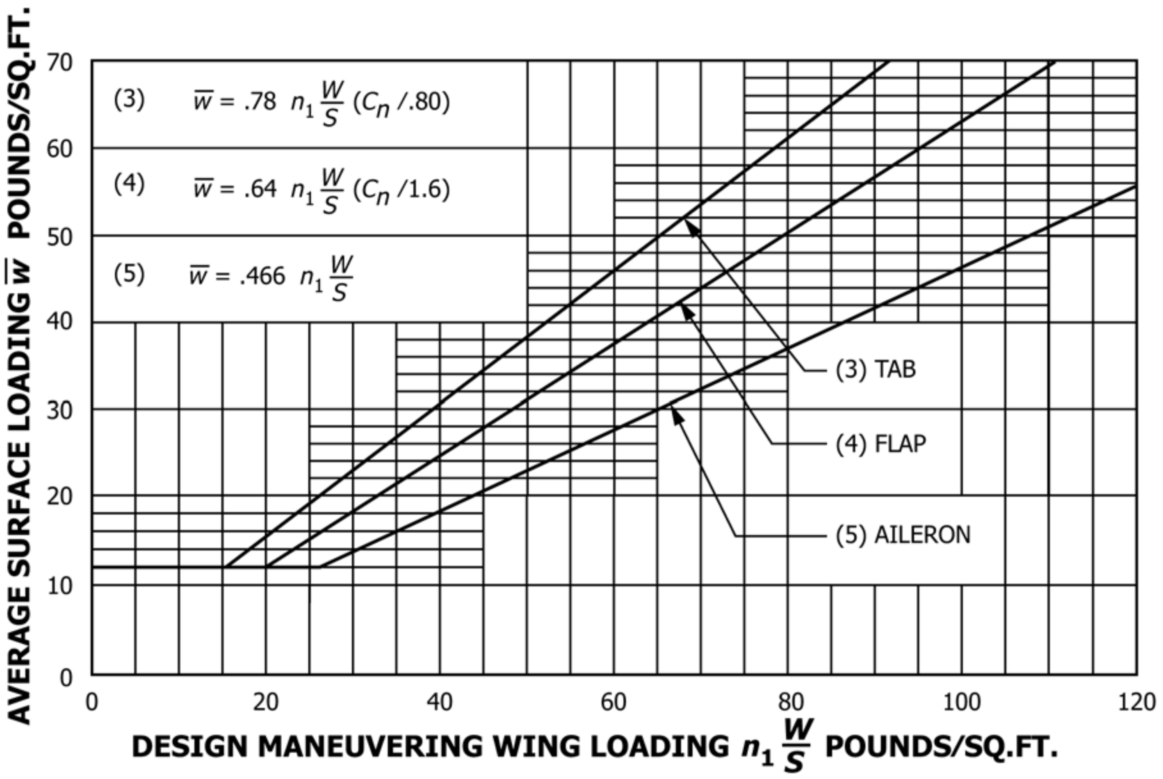


FIG. X2.2 Average Limit Control Surface Loading

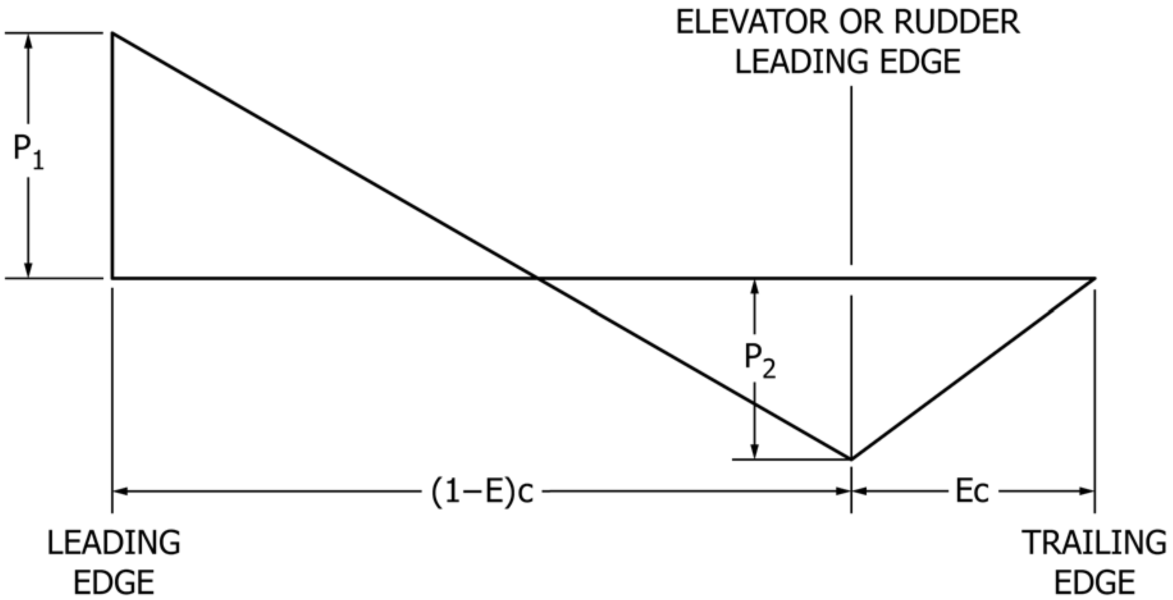


FIG. X2.3 Chordwise Load Distribution for Stabilizer and Elevator or Fin and Rudder

$$P(x) = 3(\bar{w}) \frac{(c - x)^2}{c_f^2} \qquad \text{(X2.1)}$$

where:

$P(x)$ = local pressure at the chordwise stations x,
c = chord length of the tail surface,
c_f = chord length of the elevator and rudder respectively, and
$\bar{w}$ = average surface loading as specified in Fig. X2.1.

(6) The chordwise loading distribution for ailerons, wing flaps, and trim tabs are specified in Table X2.1.

X2.2.3.2 For airplanes approved for aerobatics, the horizontal tail must be investigated for an unsymmetrical load of 100 % w̄ on one side of the airplane centerline and 50 % on the other side of the airplane centerline.

X2.2.4 *Outboard Fins*—Outboard fins must meet the requirements of 4.22.

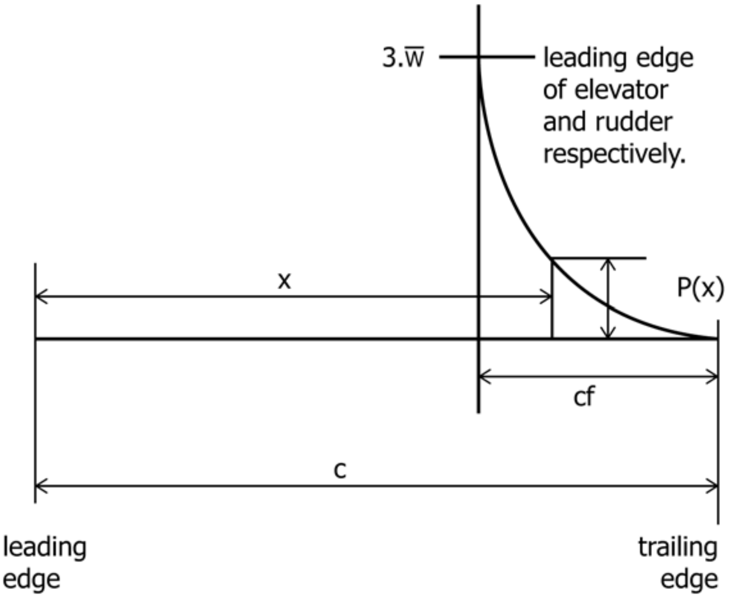


FIG. X2.4

X2.2.5 *T- and V-Tails*—T- and V-tails must meet the requirements of 4.19.

X2.2.6 *Special Devices*—Special devices must meet the requirements of 4.26.

X3. ACCEPTABLE METHODS FOR PRIMARY CONTROL SYSTEM LOADS CALCULATIONS

X3.1 Limitations

X3.1.1 The methods provided in this appendix provide one possible means (but not the only possible means) of compliance and can only be applied to level 1 and level 2 airplanes.

X3.1.2 These requirements apply only to the primary control systems used for the direct control of the airplane about its longitudinal, lateral, or yaw axis.

X3.2 Control System Loads

X3.2.1 *Primary Flight Controls and Systems*—Each primary flight control and system must be designed as follows:

X3.2.1.1 The flight control system and its supporting structure must be designed for loads corresponding to 125 % of the computed hinge moments of the movable control surface in the conditions prescribed in X2.2. In addition:

(1) The system limit loads need not exceed those that could be produced by the pilot and automatic devices operating the controls; and

(2) The design must provide a rugged system for service use, including jamming, ground gusts, taxiing downwind, control inertia, and friction.

X3.2.1.2 Acceptable maximum and minimum limit pilot forces for elevator, aileron, and rudder controls are shown in the table in 7.4.2. These pilot forces must be assumed to act at the appropriate control grips or pads as they would under flight conditions, and to be reacted at the attachments of the control system to the control surface horn.

X3.2.2 *Dual Controls*—If there are dual controls, the systems must be designed for pilots operating in opposition, using individual pilot loads equal to 75 % of those obtained in accordance with X3.2.1, except that individual pilot loads may not be less than the minimum limit pilot forces shown in the table in 7.4.2.

X3.2.3 *Ground Gust Conditions*—Ground gust conditions must meet the requirements of 7.9.

X3.2.4 *Secondary Controls and Systems*—Secondary controls and systems must meet the requirements of 7.6.

X4. CONTROL SURFACE LOADING

X4.1 Limitations

X4.1.1 The methods provided in this appendix provide one possible means (but not the only possible means) of compliance and can only be applied to level 1 airplanes.

X4.1.2 These methods may be applied to airplanes meeting the limitations in X1.1.2 without further justification.

X4.1.3 This appendix may be used outside of the limitations in X1.1.2 when evidence can be provided that the method provides safe and reliable results.

X4.1.4 Airplanes with any of the design features listed in X1.1.4 shall not use this appendix.

X4.2 General

X4.2.1 If allowed by the specific requirements in Sections 4 and 7, the values of control surface loading in this appendix may be used to determine the detailed rational requirements of 4.16 through 4.26 and 7.4 through 7.9 unless it is determined that these values result in unrealistic loads.

X4.2.2 In the control surface loading conditions of X4.3, the airloads on the movable surfaces need not exceed those that could be obtained in flight by using the maximum limit pilot forces prescribed in 7.4.2. If the surface loads are limited by these maximum limit pilot forces, the tabs must be deflected:

X4.2.2.1 To their maximum travel in the direction that would assist the pilot; or

X4.2.2.2 In an amount corresponding to the greatest degree of out-of-trim expected at the speed for the condition being considered.

X4.2.3 For a seaplane version of a landplane the landplane wing loadings may be used to determine the limit maneuvering control surface loadings (in accordance with X4.3 and Fig. X4.1) if:

X4.2.3.1 The power of the seaplane engine does not exceed the power of the landplane engine;

X4.2.3.2 The placard maneuver speed of the seaplane does not exceed the placard maneuver speed of the landplane;

X4.2.3.3 The maximum weight of the seaplane does not exceed the maximum weight of the landplane by more than 10%;

X4.2.3.4 The landplane service experience does not show any serious control-surface load problem; and

X4.2.3.5 The landplane service experience is of sufficient scope to ascertain with reasonable accuracy that no serious control surface load problem will develop on the seaplane.

X4.3 Control Surface Loads

X4.3.1 Acceptable values of limit average maneuvering control surface loadings may be obtained from Fig. X4.1 in accordance with the following:

X4.3.2 For horizontal tail surfaces:

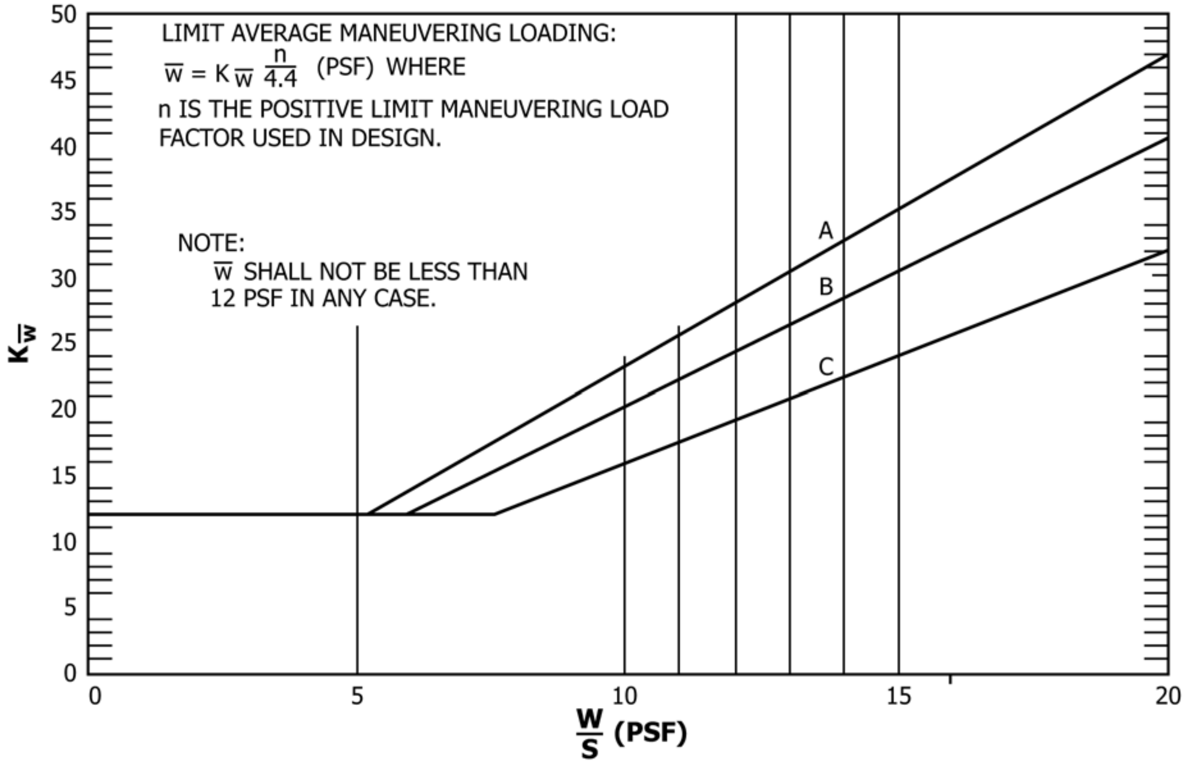


FIG. X4.1 Limit Average Maneuvering Control Surface Loading

X4.3.2.1 With the conditions in 4.17.1.1, obtain $\bar{w}$ as a function of *W/S* and surface deflection, using:

(1) Curve C of Fig. X4.1 for a deflection of 10° or less;
(2) Curve B of Fig. X4.1 for a deflection of 20°;
(3) Curve A of Fig. X4.1 for a deflection of 30° or more;
(4) Interpolation for all other deflections; and
(5) The distribution of Fig. X4.6; and

X4.3.2.2 With the conditions in 4.17.1.2, obtain $\bar{w}$ from curve B of Fig. X4.1 using the distribution of Fig. X4.6.

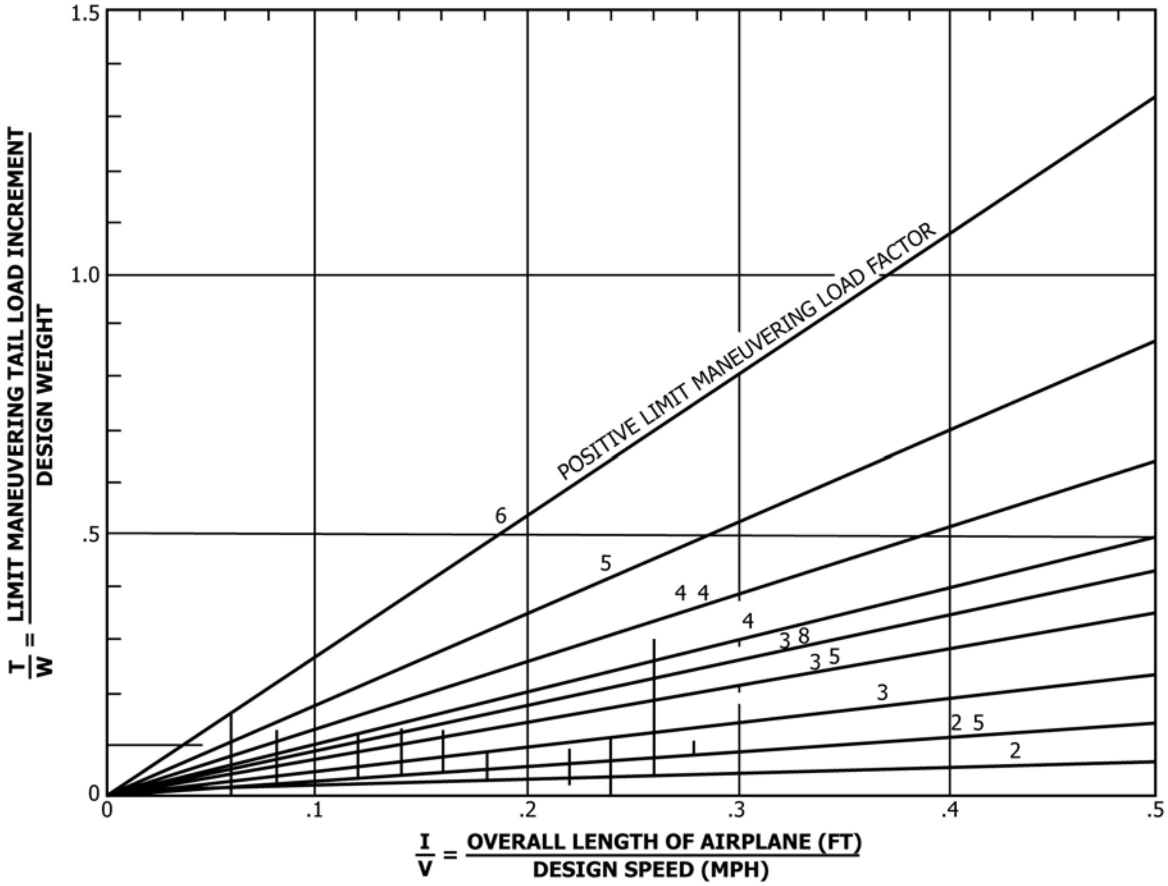
X4.3.3 For vertical tail surfaces:

X4.3.3.1 With the conditions in 4.20.1.1, obtain $\bar{w}$ as a function of *W/S* and surface deflection using the same requirements as used in X4.3.2.1;

X4.3.3.2 With the conditions in 4.20.1.2, obtain $\bar{w}$ from Curve C, using the distribution of Fig. X4.5; and

X4.3.3.3 With the conditions in 4.20.1.3, obtain $\bar{w}$ from Curve A, using the distribution of Fig. X4.7.

X4.3.4 For ailerons, obtain $\bar{w}$ from Curve B, acting in both the up and down directions, using the distribution of Fig. X4.8.



NOTE 1—As an alternative to Fig. X4.2, the following may be used:

$$\frac{T}{W} = \frac{k^2}{g l_t V} \times 20.1\, n_1 (n_1 - 1.5)$$

where:

k = the radius of gyration of the airplane in pitch,
l_t = the distance between the airplane center of gravity and the center of the lift of the horizontal tail, and
V = the airplane speed in m/s.

FIG. X4.2 Maneuvering Tail Load Increment (Up Or Down)

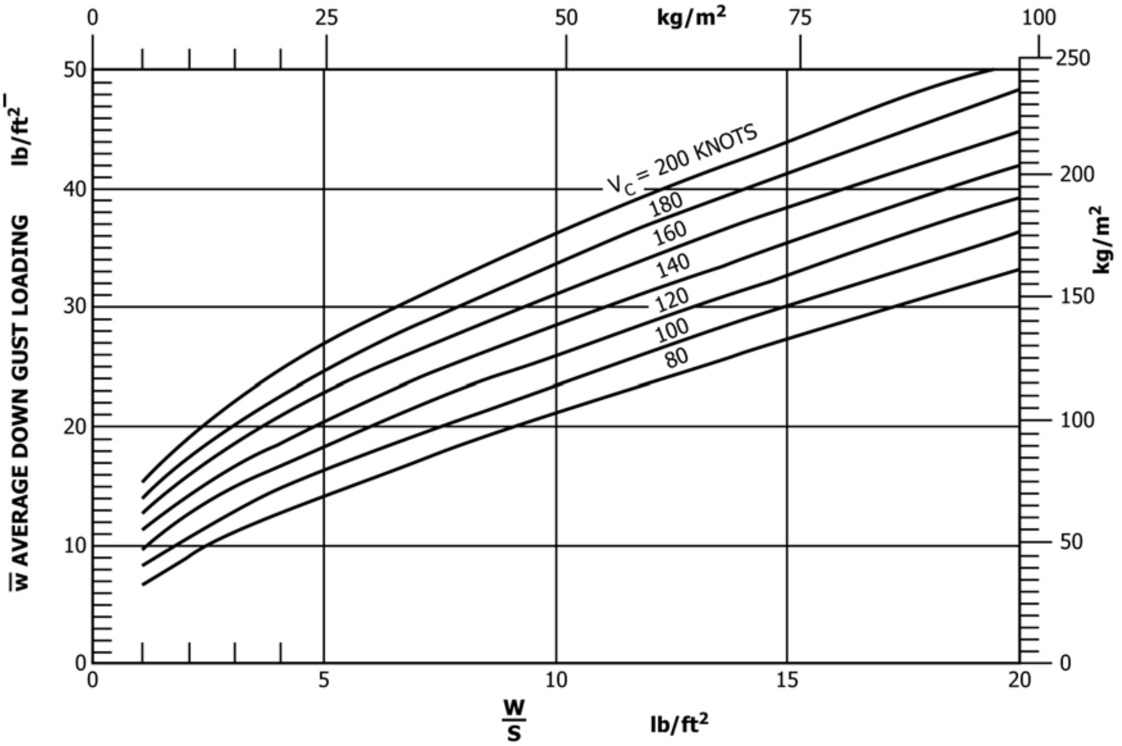


FIG. X4.3 Up and Down Gust Loading on Horizontal Tail Surface

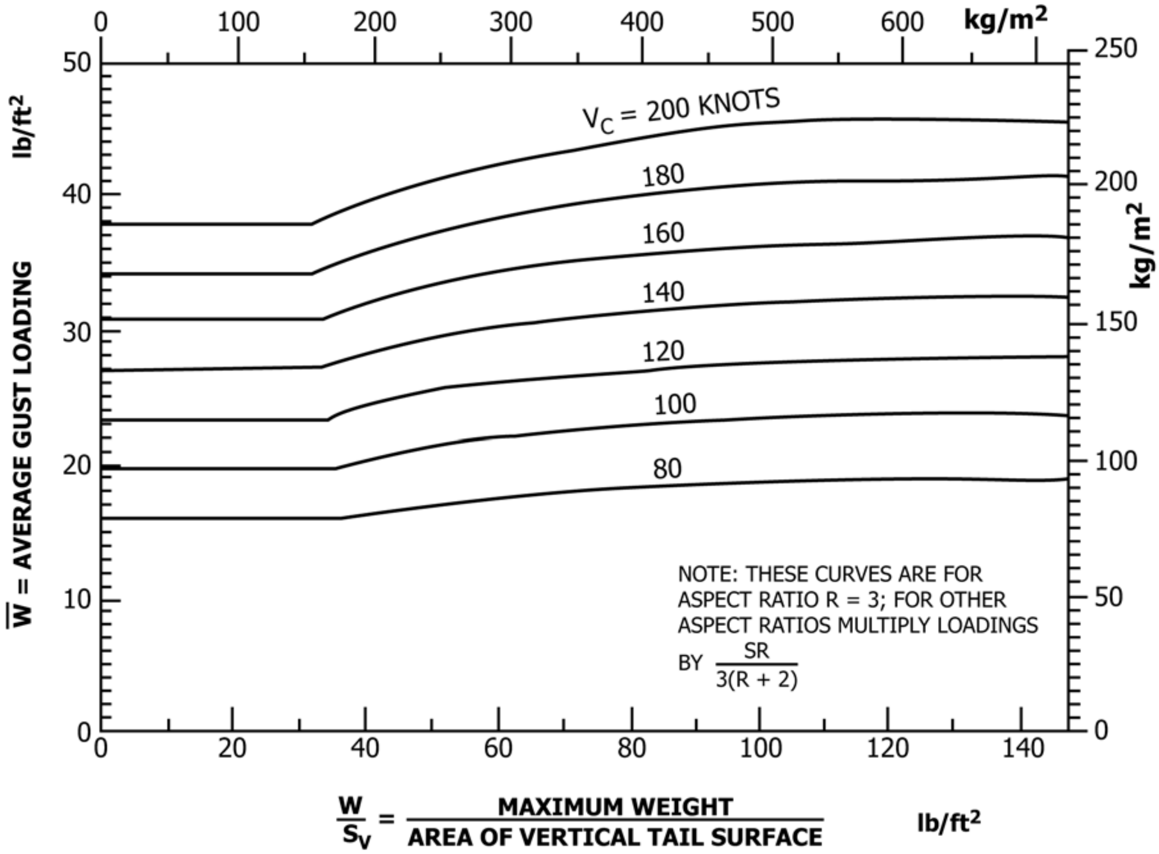
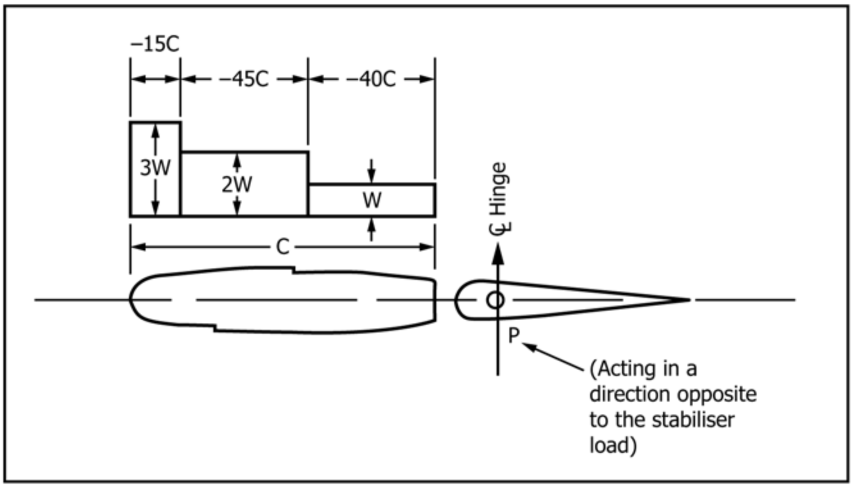


FIG. X4.4 Gust Loading on Vertical Tail Surface



Notes:
(a) In balancing conditions in 4.16, P = 40% of net balancing load (flaps retracted); and P = 0 (flaps deflected).
(b) In the condition in 4.20.1.2, P = 20% of net tail load.
(c) The load on the fixed surface must be:
  (1) 140% of the net balancing load for the flaps retracted case of note (a);
  (2) 100% of the net balancing load for the flaps deflected case of note (a); and
  (3) 120% of the net balancing load for the case in note (b).

FIG. X4.5 Tail Surface Load Distribution

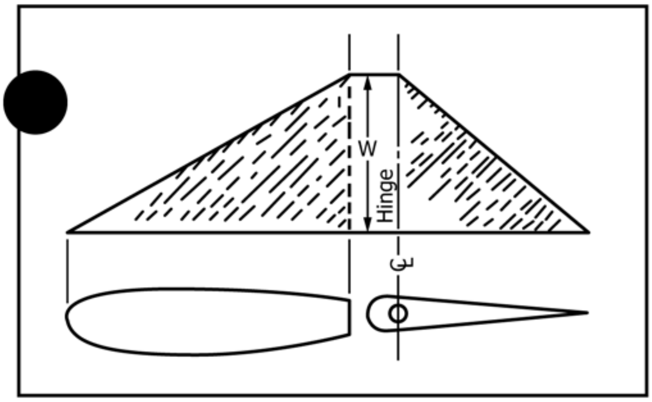


FIG. X4.6 Tail Surface Load Distribution

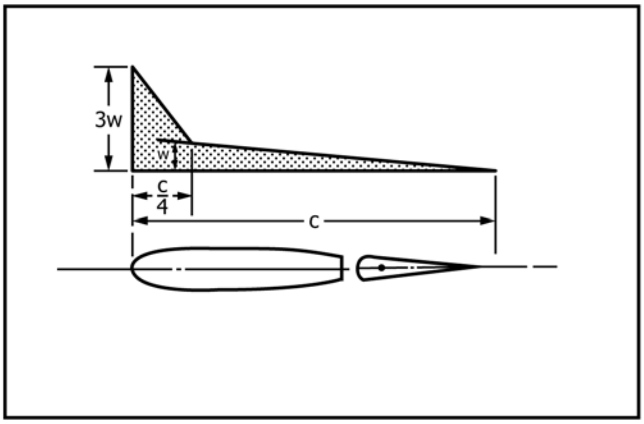


FIG. X4.7 Tail Surface Load Distribution

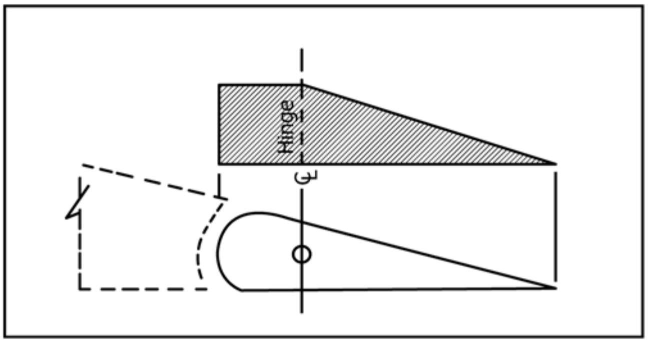


FIG. X4.8 Aileron Load Distribution

X5. BASIC LANDING CONDITIONS

X5.1 Basic Landing Conditions

X5.1.1 Table X5.1 and Fig. X5.1 present the basic landing conditions that can be used when showing compliance with the requirements in Section 8.

TABLE X5.1 Basic Landing Conditions

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	8.4.1.1	8.5.1.1	8.4.1.2*(1)*	8.4.1.2*(2)*	8.5.1.2 and 8.5.2
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 (Vr)	(n-L)W	(n-L)Wb/d	(n-L)Wa’/d’	(n-L)W	(n-L)W
(both wheels) (Dr)	KnW	0	KnWa’/d’	KnW	0
Tail (nose) wheel (Vf)	0	(n-L)Wa/d	(n-L)Wb’/d’	0	0
loads (Df)	0	0	KnWb’/d’	0	0
Notes	(1), (3), and (4)	(4)	(1)	(1), (3), and (4)	(3) and (4)

NOTE 1—*K* may be determined as follows: *K* = 0.25 for *W* = 1361 kg [3000 lb] or less; *K* = 0.33 for *W* = 2722 kg [6000 lb] or greater, with linear variation of *K* between these weights.

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

NOTE 3—Unbalanced moments must be balanced by a rational or conservative method.

NOTE 4—*L* is defined in 14 CFR Part 23, Sec. 23.725 (b).

NOTE 5—*n* is the limit inertia load factor, at the c.g. of the airplane, selected under 8.2.4, 8.2.6, and 8.2.7.

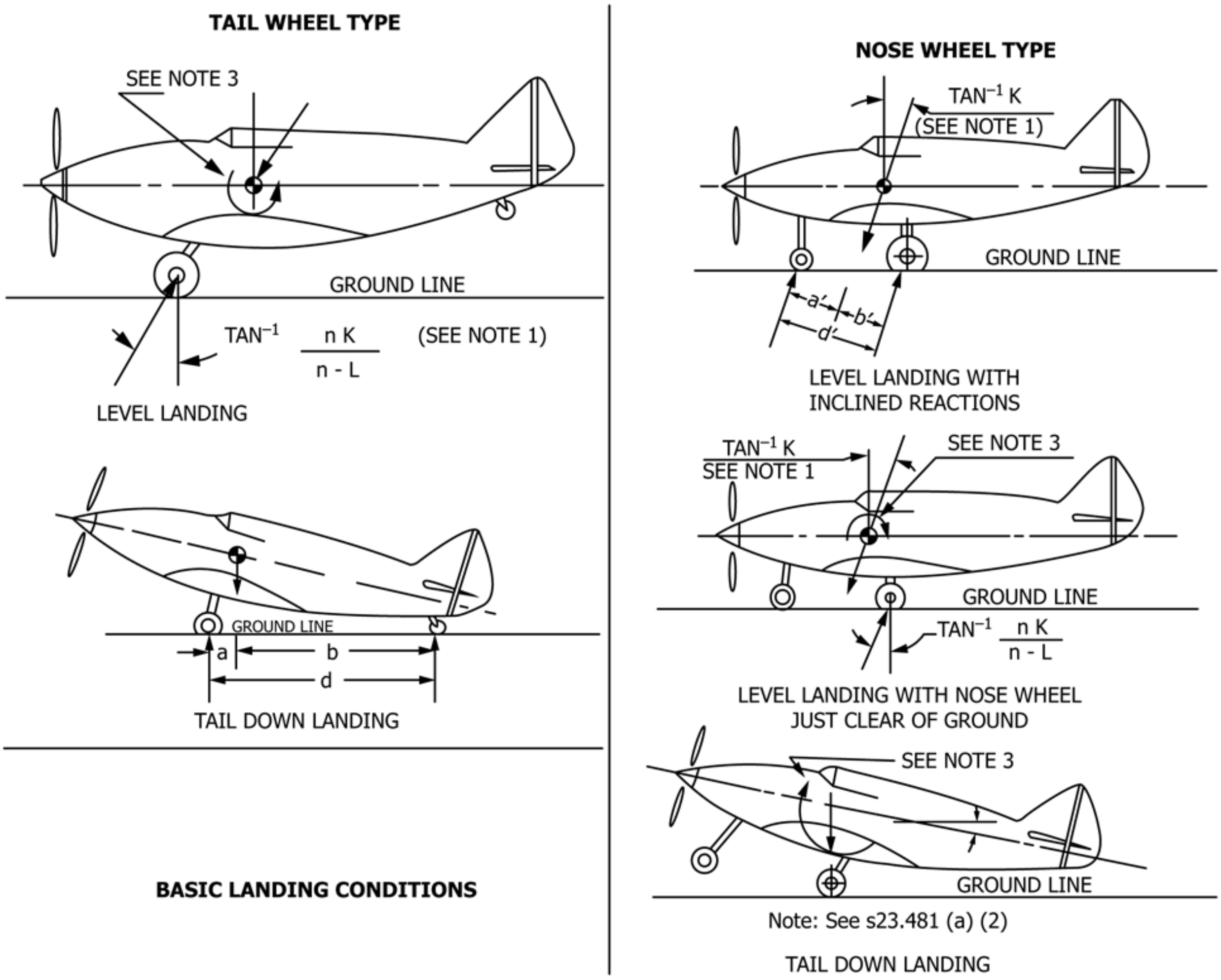


FIG. X5.1 Basic Landing Conditions

X6. WHEEL SPIN-UP AND SPRING-BACK LOADS

X6.1 Wheel Spin-Up Loads

X6.1.1 The following method for determining wheel spin-up loads for landing conditions is based on NACA T.N. 863. However, the drag component used for design may not be less than the drag load prescribed in 8.4.2.

$$F_{Hmax} = \frac{1}{r_e}\sqrt{\frac{2I_w(V_H - V_C)nF_{Vmax}}{t_z}} \qquad \text{(X6.1)}$$

where:

F_{Hmax} = maximum rearward horizontal force acting on the wheel (in pounds);

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

I_w = rotational mass moment of inertia of rolling assembly (in slug feet);

V_H = linear velocity of airplane parallel to ground at instant of contact (assumed to be $1.2V_{S0}$, in feet per second);

V_C = peripheral speed of tire, if pre-rotation is used (in feet per second) (there must be a positive means of pre-rotation before pre-rotation may be considered);

n = effective coefficient of friction (0.80 may be used);

F_{Vmax} = maximum vertical force on wheel (pounds) = n_jW_e, where W_e and n_j are defined in 14 CFR Part 23, Sec. 23.725; and

t_z = time interval between ground contact and attainment of maximum vertical force on wheel (seconds). (However, if the value of F_{Vmax}, from the above equation exceeds 0.8 F_{Vmax}, the latter value must be used for F_{Hmax}).

X6.1.2 This equation assumes a linear variation of load factor with time until the peak load is reached and under this assumption, the equation 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. Therefore, rational or conservative allowances must 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. For exceptionally large wheels, a wheel peripheral velocity equal to the ground speed may not have been attained at the time of maximum vertical gear load. However, as stated above, the drag spin-up load need not exceed 0.8 of the maximum vertical loads.

X6.1.3 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 must be determined, in the level landing condition, by assuming that the wheel spin-up loads calculated by the methods of this appendix are reversed. Dynamic spring-back is likely to become critical for landing gear units having wheels of large mass or high landing speeds.

X7. ACCEPTABLE MEANS FOR CALCULATION OF WATER LOADS

X7.1 Design Weights and Center of Gravity Positions

X7.1.1 *Design Weights*—The water load requirements must be met at each operating weight up to the design landing weight except that, for the takeoff condition prescribed in X7.5, the design water takeoff weight (the maximum weight for water taxi and takeoff run) must be used.

X7.1.2 *Center of Gravity Positions*—The critical centers of gravity within the limits for which certification is requested must be considered to reach maximum design loads for each part of the seaplane structure.

X7.2 Application of Loads

X7.2.1 Unless otherwise prescribed, the seaplane as a whole is assumed to be subjected to the loads corresponding to the load factors specified in X7.3.

X7.2.2 In applying the loads resulting from the load factors prescribed in X7.3, the loads may be distributed over the hull or main float bottom (in order to avoid excessive local shear loads and bending moments at the location of water load application) using pressures not less than those prescribed in X7.6.3.

X7.2.3 For twin float seaplanes, each float must be treated as an equivalent hull on a fictitious seaplane with a weight equal to one-half the weight of the twin float seaplane.

X7.2.4 Except in the takeoff condition of X7.5, the aerodynamic lift on the seaplane during the impact is assumed to be ⅔ of the weight of the seaplane.

X7.3 Water Loads—Hull and Main Float Load Factors

X7.3.1 Water reaction load factors nw must be computed in the following manner:

X7.3.1.1 For the step landing case:

$$n_w = \frac{C_1 V_{S0}^{\ 2}}{(\tan^{2/3} \beta) W^{1/3}} \tag{X7.1}$$

X7.3.1.2 For the bow and stern landing cases:

$$n_w = \frac{C_1 V_{S0}^{\ 2}}{(\tan^{2/3} \beta) W^{1/3}} \times \frac{K_1}{(1 + r_x^{\ 2})^{2/3}} \tag{X7.2}$$

X7.3.2 The following values are used:

X7.3.2.1 n_w = water reaction load factor (that is, the water reaction divided by seaplane weight).

X7.3.2.2 C_1 = empirical seaplane operations factor equal to 0.012 (except that this factor may not be less than that necessary to obtain the minimum value of step load factor of 2.33).

X7.3.2.3 V_{S0} = seaplane stalling speed in knots with flaps extended in the appropriate landing position and with no slipstream effect.

X7.3.2.4 β = angle of dead rise at the longitudinal station at which the load factor is being determined in accordance with Fig. X7.1.

X7.3.2.5 W = seaplane design landing weight in pounds.

X7.3.2.6 K_1 = empirical hull station weighing factor, in accordance with Fig. X7.2.

X7.3.2.7 r_x = ratio of distance, measured parallel to hull reference axis, from the center of gravity of the seaplane to the hull longitudinal station at which the load factor is being computed to the radius of gyration in pitch of the seaplane, the hull reference axis being a straight line, in the plane of symmetry, tangential to the keel at the main step.

X7.3.3 For a twin float seaplane, because of the effect of flexibility of the attachment of the floats to the seaplane, the factor K_1 may be reduced at the bow and stern to 0.8 of the value shown in Fig. X7.2. This reduction applies only to the design of the carry through and seaplane structure.

X7.4 Hull and Main Float Landing Conditions

X7.4.1 *Symmetrical Step, Bow, and Stern Landing*—For symmetrical step, bow, and stern landings, the limit water reaction load factors are those computed under X7.3. In addition:

X7.4.1.1 For symmetrical step landings, the resultant water load must be applied at the keel, through the center of gravity, and must be directed perpendicularly to the keel line;

X7.4.1.2 For symmetrical bow landings, the resultant water load must be applied at the keel, one-fifth of the longitudinal distance from the bow to the step, and must be directed perpendicularly to the keel line; and

X7.4.1.3 For symmetrical stern landings, the resultant water load must be applied at the keel, at a point 85 % of the longitudinal distance from the step to the stern post, and must be directed perpendicularly to the keel line.

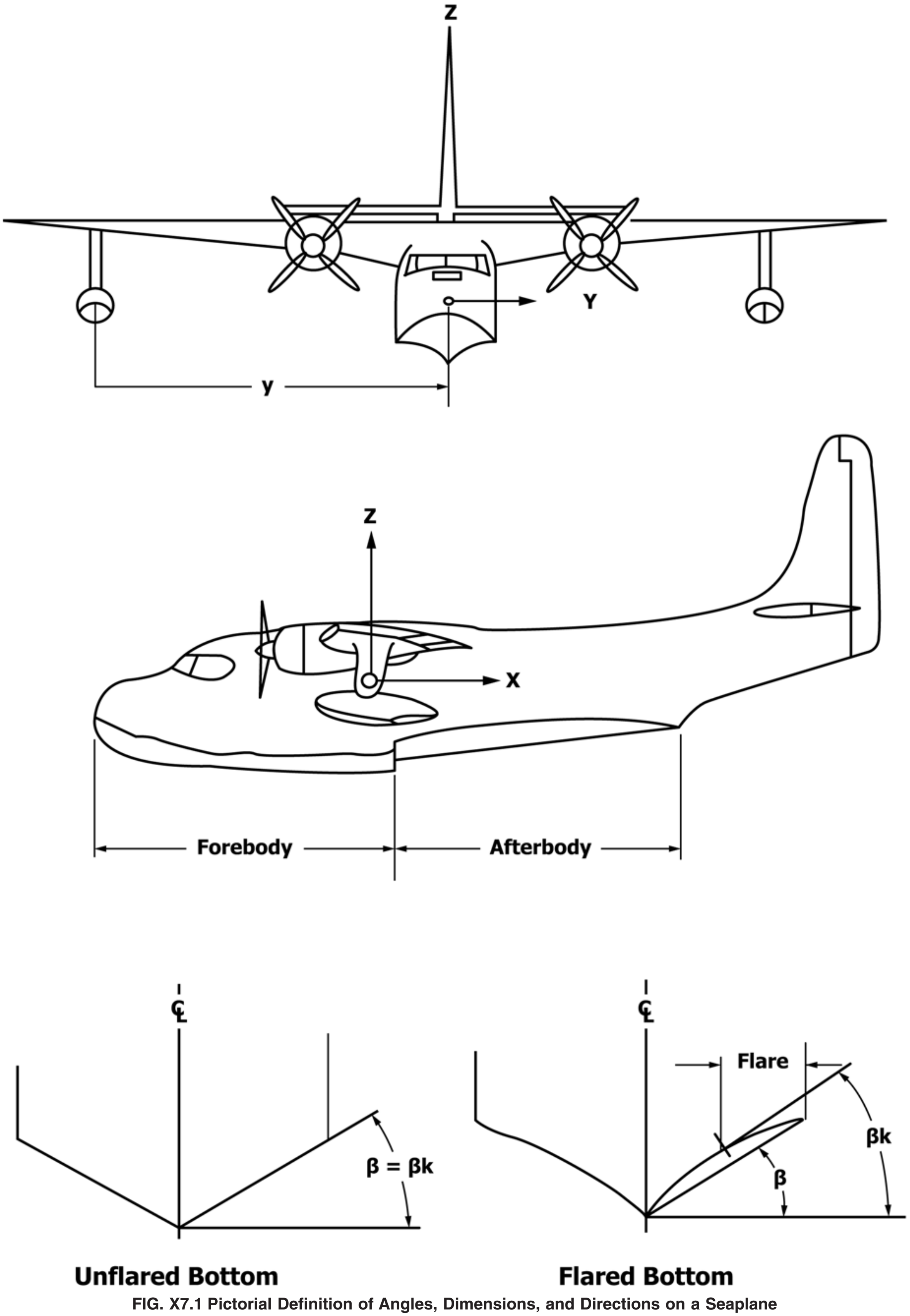
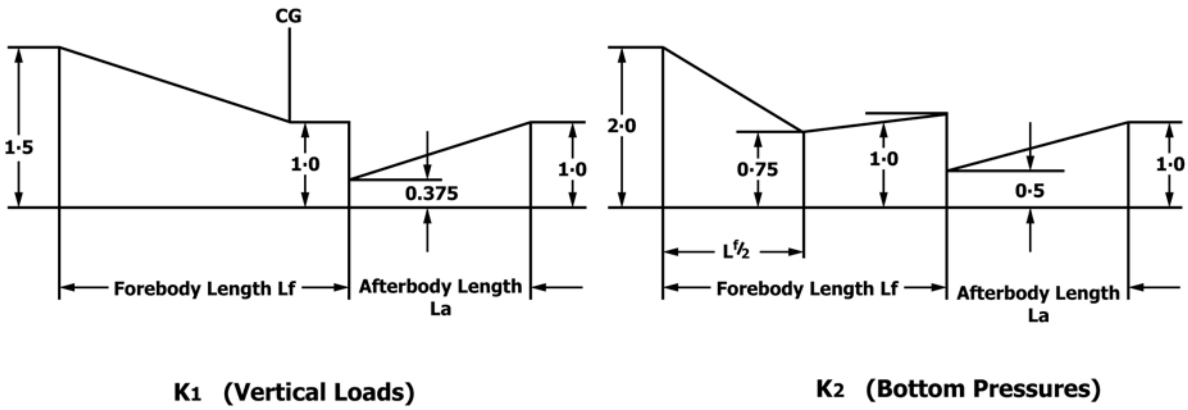


FIG. X7.1 Pictorial Definition of Angles, Dimensions, and Directions on a Seaplane



K1 (Vertical Loads) **K2 (Bottom Pressures)**

FIG. X7.2 Hull Station Weighing Factor

X7.4.2 *Unsymmetrical Landing for Hull And Single Float Seaplanes*—Unsymmetrical step, bow, and stern landing conditions must be investigated. In addition:

X7.4.2.1 The loading for each condition consists of an upward component and a side component equal, respectively, to 0.75 and 0.25 tan β times the resultant load in the corresponding symmetrical landing condition; and

X7.4.2.2 The point of application and direction of the upward component of the load is the same as that in the symmetrical condition, and the point of application of the side component is at the same longitudinal station as the upward component but is directed inward perpendicularly to the plane of symmetry at a point midway between the keel and the chine lines.

X7.4.3 *Unsymmetrical Landing; Twin Float Seaplanes*—The unsymmetrical loading consists of an upward load at the step of each float of 0.75 and a side load of 0.25 tan β at one float times the step landing load reached under X7.3. The side load is directed inboard, perpendicularly to the plane of symmetry midway between the keel and chine lines of the float, at the same longitudinal station as the upward load.

X7.5 Water Loads—Hull and Main Float Takeoff Condition

X7.5.1 For the wing and its attachment to the hull or main float, the aerodynamic wing lift is assumed to be zero; and

X7.5.2 A downward inertia load, corresponding to a load factor computed from the following formula, must be applied:

$$n = \frac{C_{TO} V_{S1}^{\ 2}}{(\tan^{2/3} \beta) W^{1/3}} \qquad (X7.3)$$

where:

n = inertia load factor;
C_{TO} = empirical seaplane operations factor equal to 0.004;
V_{S1} = seaplane stalling speed (knots) at the design takeoff weight with the flaps extended in the appropriate takeoff position;
β = angle of dead rise at the main step (degrees); and
W = design water takeoff weight in pounds.

X7.6 Hull and Main Float Bottom Pressures

X7.6.1 *General*—The hull and main float structure, including frames and bulkheads, stringers, and bottom plating, must be designed under this section.

X7.6.2 *Local Pressures*—For the design of the bottom plating and stringers and their attachments to the supporting structure, the following pressure distributions must be applied:

X7.6.2.1 For an unflared bottom, the pressure at the chine is 0.75 times the pressure at the keel, and the pressures between the keel and chine vary linearly, in accordance with Fig. X7.3. The pressure at the keel (p.s.i.) is computed as follows:

$$p_K = \frac{C_2 K_2 V_{S1}^{\ 2}}{\tan \beta_k} \qquad (X7.4)$$

where:

P_K = pressure (p.s.i.) at the keel;
C_2 = 0.00213;
K_2 = hull station weighing factor, in accordance with Fig. X7.2;
V_{S1} = seaplane stalling speed (knots) at the design water takeoff weight with flaps extended in the appropriate takeoff position; and
β_k = angle of dead rise at keel, in accordance with Fig. X7.1.

X7.6.2.2 For a flared bottom, the pressure at the beginning of the flare is the same as that for an unflared bottom, and the pressure between the chine and the beginning of the flare varies linearly, in accordance with Fig. X7.3. The pressure distribution is the same at that prescribed in X7.6.2.1 for an unflared bottom except that the pressure at the chine is computed as follows:

$$P_{ch} = \frac{C_3 K_2 V_{S1}^{\ 2}}{\tan \beta} \qquad (X7.5)$$

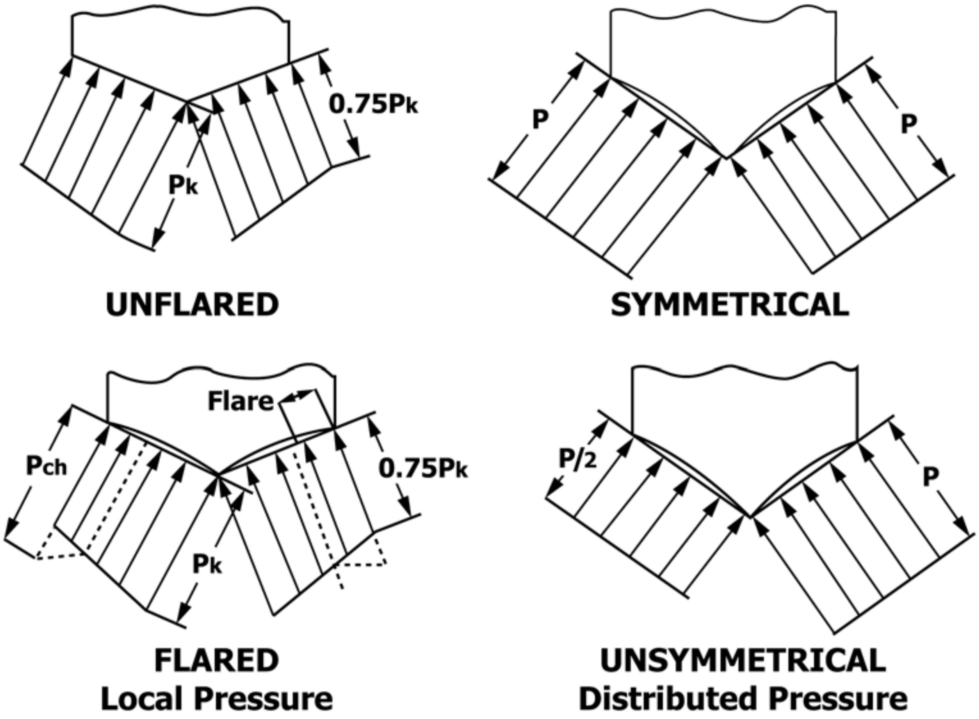


FIG. X7.3 Transverse Pressure Distributions

where:

P_{ch} = pressure (p.s.i) at the chine;
C_3 = 0.0016;
K_2 = hull station weighing factor, in accordance with Fig. X7.2;
V_{S1} = seaplane stalling speed (knots) at the design water takeoff weight with flaps extended in the appropriate takeoff position; and
β = angle of dead rise at appropriate station.

X7.6.2.3 The area over which these pressures are applied must simulate pressures occurring during high localized impacts on the hull or float, but need not extend over an area that would induce critical stresses in the frames or in the overall structure.

X7.6.3 *Distributed Pressures*—For the design of the frames, keel, and chine structure, the following pressure distributions apply:

X7.6.3.1 Symmetrical pressures as computed as follows:

$$P = \frac{C_4 K_2 V_{SO}^2}{\tan\beta} \qquad (X7.6)$$

where:

P = pressure (p.s.i.);
C_4 = 0.078 C_1 (with C_1 computed under X7.3);
K_2 = hull station weighing factor, determined in accordance with Fig. X7.2;
V_{SO} = seaplane stalling speed (knots) with landing flaps extended in the appropriate position and with no slipstream effect; and
β = angle of dead rise at appropriate station.

X7.6.3.2 The unsymmetrical pressure distribution consists of the pressures prescribed in X7.6.3.1 on one side of the hull or main float centerline and one-half of that pressure on the other side of the hull or main float centerline in accordance with Fig. X7.3.

X7.6.3.3 These pressures are uniform and must be applied simultaneously over the entire hull or main float bottom. The loads obtained must be carried into the sidewall structure of the hull proper, but need not be transmitted in a fore and aft direction as shear and bending loads.

X7.7 Auxiliary Float Loads

X7.7.1 *General*—Auxiliary floats and their attachments and supporting structures must be designed for the conditions prescribed in this section. In the cases specified in X7.7.2 through X7.7.5, the prescribed water loads may be distributed over the float bottom to avoid excessive local loads, using bottom pressures not less than those prescribed in X7.7.7.

X7.7.2 *Step Loading*—The resultant water load must be applied in the plane of symmetry of the float at a point three-fourths of the distance from the bow to the step and must be perpendicular to the keel. The resultant limit load is computed as follows, except that the value of *L* need not exceed three times the weight of the displaced water when the float is completely submerged:

$$L = \frac{C_5 V_{SO}^2 W^{2/3}}{\tan^{2/3}\beta_S \left(1 + r_y^2\right)^{2/3}} \qquad (X7.7)$$

where:

L = limit load (lb);
C_5 = 0.0053;
V_{SO} = seaplane stalling speed (knots) with landing flaps extended in the appropriate position and with no slipstream effect;

W = seaplane design landing weight in pounds;
β_S = angle of dead rise at a station ¾ of the distance from the bow to the step, but need not be less than 15°; and
r_y = ratio of the lateral distance between the center of gravity and the plane of symmetry of the float to the radius of gyration in roll.

X7.7.3 *Bow Loading*—The resultant limit load must be applied in the plane of symmetry of the float at a point one-fourth of the distance from the bow to the step and must be perpendicular to the tangent to the keel line at that point. The magnitude of the resultant load is that specified in X7.7.2.

X7.7.4 *Unsymmetrical Step Loading*—The resultant water load consists of a component equal to 0.75 times the load specified in X7.7.1 and a side component equal to 0.25 tan β times the load specified in X7.7.2. The side load must be applied perpendicularly to the plane of symmetry of the float at a point midway between the keel and the chine.

X7.7.5 *Unsymmetrical Bow Loading*—The resultant water load consists of a component equal to 0.75 times the load specified in X7.7.2 and a side component equal to 0.25 tan β times the load specified in X7.7.3. The side load must be applied perpendicularly to the plane of symmetry at a point midway between the keel and the chine.

X7.7.6 *Immersed Float Condition*—The resultant load must be applied at the centroid of the cross section of the float at a point one-third of the distance from the bow to the step. The limit load components are as follows:

$$\begin{aligned} vertical &= \rho g V \\ aft &= \frac{C_x \rho V^{2/3}(K\ V_{SO})^2}{2} \\ side &= \frac{C_Y \rho V^{2/3}(K\ V_{SO})^2}{2} \end{aligned} \tag{X7.8}$$

where:

ρ = mass density of water (slugs/ft^3);
V = volume of float (ft^3);
C_X = coefficient of drag force, equal to 0.133;
C_Y = coefficient of side force, equal to 0.106;
K = 0.8, except that lower values may be used if it is shown that the floats are incapable of submerging at a speed of 0.8 V_{SO} in normal operations;
V_{SO} = seaplane stalling speed (knots) with landing flaps extended in the appropriate position and with no slipstream effect; and
g = acceleration due to gravity (ft/s^2).

X7.7.7 *Float Bottom Pressures*—The float bottom pressures must be established under X7.6, except that the value of K_2 in the formulae may be taken as 1.0. The angle of dead rise to be used in determining the float bottom pressures is set forth in X7.7.2.

X7.8 Seawing Loads

X7.8.1 Seawing design loads must be based on applicable test data.