Standard Specification for Design and Performance of a Light Sport Airplane[1]

1. Scope

1.1 This specification covers airworthiness requirements for the design of powered fixed wing light sport aircraft, an "airplane."

1.2 This specification is applicable to the design of a light sport aircraft/airplane as defined by regulations and limited to VFR flight.

1.3 *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 requirements prior to use.*

2. Referenced Documents

2.1 *ASTM Standards:*[2]

F 2316 Specification for Airframe Emergency Parachutes for Light Sport Aircraft

F 2339 Practice for Design and Manufacture of Reciprocating Spark Ignition Engines for Light Sport Aircraft

2.2 *Federal Aviation Regulations:* [3]

FAR-33 Airworthiness Standards: Aircraft Engines

2.3 *Joint Aviation Requirements:* [4]

JAR-E Engines

JAR-22 Sailplanes and Powered Sailplanes

3. Terminology

3.1 *Definitions:*

3.1.1 *flaps*—any movable high lift device.

3.1.2 *maximum empty weight,* W_E (N)—largest empty weight of the airplane, including all operational equipment that is installed in the airplane: weight of the airframe, powerplant, required equipment, optional and specific equipment, fixed ballast, full engine coolant and oil, hydraulic fluid, and the unusable fuel. Hence, the maximum empty weight equals maximum takeoff weight minus minimum useful load: $W_E = W - W_U$.

3.1.3 *minimum useful load,* W_U (N)—where $W_U = W - W_E$.

3.1.4 *night*—hours between the end of evening civil twilight and the beginning of morning civil twilight.

3.1.4.1 *Discussion*—Civil twilight ends in the evening when the center of the sun's disc is 6 degrees below the horizon, and begins in the morning when the center of the sun's disc is 6 degrees below the horizon.

3.2 *Abbreviations:*

3.2.1 *AR*—aspect ratio = b^2 / S

3.2.2 *b*—wing span, (m)

3.2.3 *c*—chord, (m)

3.2.4 *CAS*—calibrated air speed, (m/s, kts)

3.2.5 C_L—lift coefficient of the airplane

3.2.6 C_D—drag coefficient of the airplane

3.2.7 *CG*—center of gravity

3.2.8 C_m—moment coefficient (C_m is with respect to c/4 point, positive nose up)

3.2.9 C_{MO}—zero lift moment coefficient

3.2.10 C_n—normal coefficient

3.2.11 *g*—acceleration as a result of gravity = 9.81 m/s^2

3.2.12 *IAS*—indicated air speed (m/s, kts)

3.2.13 *ICAO*—International Civil Aviation Organization

3.2.14 *LSA*—light sport aircraft

3.2.15 *MAC*—mean aerodynamic chord (m)

3.2.16 *n*—load factor

3.2.17 n_1—airplane positive maneuvering limit load factor

3.2.18 n_2—airplane negative maneuvering limit load factor

3.2.19 n_3—load factor on wheels

3.2.20 *P*—power, (kW)

3.2.21 ρ—air density (kg/m^3) = 1.225 at sea level standard conditions

3.2.22 *POH*—Pilot Operating Handbook

3.2.23 *q*—dynamic pressure (N/m^2) = $1/2\rho V^2$

3.2.24 *RC*—climb rate (m/s)

3.2.25 *S*—wing area (m^2)

3.2.26 *V*—airspeed (m/s, kts)

3.2.27 V_A—design maneuvering speed

3.2.28 V_C—design cruising speed

3.2.29 V_D—design diving speed

3.2.30 V_{DF}—demonstrated flight diving speed ($V_{DF} \leq V_D$)

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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 Federal Aviation Administration (FAA), 800 Independence Ave., SW, Washington, DC 20591, http://www.faa.gov.

[4] Available from Global Engineering Documents, 15 Inverness Way, East Englewood, CO 80112-5704, http://global.ihs.com.

3.2.31 V_F—design flap speed

3.2.32 V_{FE}—maximum flap extended speed

3.2.33 V_H—maximum speed in level flight with maximum continuous power (corrected for sea level standard conditions)

3.2.34 V_{NE}—never exceed speed ($V_H \leq V_{NE} \leq 0.9V_{DF}$)

3.2.35 V_S—stalling speed or minimum steady flight speed at which the airplane is controllable (flaps retracted)

3.2.36 V_{S1}—stalling speed or minimum steady flight speed with the flaps in a specific configuration

3.2.37 V_{S0}—stalling speed or minimum steady flight speed at which the airplane is controllable in the landing configuration (flaps fully deployed)

3.2.38 V_{SP}—maximum spoiler/speed brake extended speed

3.2.39 V_R—ground gust speed

3.2.40 V_X—speed for best angle of climb

3.2.41 V_Y—speed for best rate of climb

3.2.42 W—maximum takeoff or maximum design weight (N)

3.2.43 W_E—maximum empty airplane weight (N)

3.2.44 W_U—minimum useful load (N)

3.2.45 w—average design surface load (N/m^2)

4. Flight

4.1 *Proof of Compliance*:

4.1.1 Each of the following requirements shall be met at the most critical weight and CG configuration. Unless otherwise specified, the speed range from stall to V_{DF} or the maximum allowable speed for the configuration being investigated shall be considered.

4.1.1.1 V_{DF} may be less than or equal to V_D.

4.1.1.2 If V_{DF} chosen is less than V_D, V_{NE} must be less than or equal to $0.9V_{DF}$ and greater than or equal to $1.1V_C$.

4.1.2 The following tolerances are acceptable during flight testing:

Weight	+5 %, −10 %
Weight, when critical	+5 %, −1 %
CG	±7 % of total travel

4.2 *Load Distribution Limits*:

4.2.1 *Minimum Useful Load Requirement*:

4.2.1.1 For a single-place airplane:

$$W_U = 845 + 3P, (N)$$

where:
P = rated engine power, kW.

4.2.1.2 For a two-place airplane:

$$W_U = 1690 + 3P, (N)$$

where:
P = rated engine power, kW.

4.2.2 Minimum flying weight shall be determined.

NOTE 1—For reference, standard occupant weight = 845 N (190 lb). For the minimum flying weight, standard occupant weight = 534 N (120 lb). Fuel density = 0.72 kg/L (7 N/L; 6 lb/U.S. gal).

4.2.3 Empty CG, most forward, and most rearward CG shall be determined.

4.2.4 Fixed or removable ballast, or both, may be used if properly installed and placarded.

4.3 *Propeller Speed and Pitch Limits*—Propeller configuration shall not allow the engine to exceed safe operating limits established by the engine manufacturer under normal conditions.

4.3.1 Maximum RPM shall not be exceeded with full throttle during takeoff, climb, or flight at $0.9V_H$, and 110 % maximum continuous RPM shall not be exceeded during a glide at V_{NE} with throttle closed.

4.4 *Performance, General*—All performance requirements apply in standard ICAO atmosphere in still air conditions and at sea level. Speeds shall be given in indicated (IAS) and calibrated (CAS) airspeeds.

4.4.1 *Stalling Speeds*—Wing level stalling speeds V_{SO} and V_S shall be determined by flight test at a rate of speed decrease of 1 kts/s or less, throttle closed, with maximum takeoff weight, and most unfavorable CG.

4.4.2 *Takeoff*—With the airplane at maximum takeoff weight, full throttle, the following shall be measured using normal takeoff procedures:

NOTE 2—The procedure used for normal takeoff, including flap position, shall be specified within the POH.

4.4.2.1 Ground roll distance to takeoff on a runway with minimal grade.

4.4.2.2 Distance to clear a 15-m (50-ft) obstacle at a climb speed of at least $1.3V_{S1}$.

4.4.3 *Climb*—At maximum takeoff weight, flaps in the position specified for climb within the POH, and full throttle:

4.4.3.1 Rate of climb at V_Y shall exceed 95 m/min (312 fpm).

4.4.3.2 Climb gradient at V_X shall exceed $\frac{1}{12}$.

4.4.4 *Landing*—For landing with throttle closed and flaps extended, the following shall be determined:

4.4.4.1 Landing distance from 15 m (50 ft) above ground when speed at 15 m (50 ft) is $1.3V_{SO}$.

4.4.4.2 Ground roll distance with reasonable braking if so equipped.

4.4.5 *Balked Landing*—The airplane shall demonstrate a full-throttle climb gradient at $1.3V_{SO}$ which shall exceed $\frac{1}{30}$ within 5 s of power application from aborted landing. If the flaps may be promptly and safely retracted without loss of altitude and without sudden changes in attitude, they may be retracted.

4.5 *Controllability and Maneuverability*:

4.5.1 *General*:

4.5.1.1 The airplane shall be safely controllable and maneuverable during takeoff, climb, level flight (cruise), dive to V_{DF} or the maximum allowable speed for the configuration being investigated, approach, and landing (power off and on, flaps retracted and extended) through the normal use of primary controls.

4.5.1.2 Smooth transition between all flight conditions shall be possible without exceeding pilot force as shown in Table 1.

4.5.1.3 Full control shall be maintained when retracting and extending flaps within their normal operating speed range (V_{SO} to V_{FE}).

4.5.1.4 Lateral, directional, and longitudinal control shall be possible down to V_{SO}.

4.5.2 *Longitudinal Control*:

TABLE 1 Pilot Force

Pilot force as applied to the controls	Pitch, N (lb)	Roll, N (lb)	Yaw, N (lb)
For temporary application (less than 2 min):			
Stick	200 (45)	100 (22.5)	…
Wheel (applied to rim)	200 (45)	100 (22.5)	…
Rudder pedal	…	…	400 (90)
For prolonged application:	23 (5.2)	23 (5.2)	110 (24.7)

4.5.2.1 With the airplane trimmed as closely as possible for steady flight at $1.3V_{S1}$, it must be possible at any speed between $1.1V_{S1}$ and $1.3V_{S1}$ to pitch the nose downward so that a speed not less than $1.3V_{S1}$ can be reached promptly. This must be shown with the airplane in all possible configurations, with simultaneous application of full power and nose down pitch control, and with power at idle.

4.5.2.2 Longitudinal control forces shall increase with increasing load factor.

4.5.3 *Directional and Lateral Control*:

4.5.3.1 It must be possible to reverse a steady 30° banked coordinated turn through an angle of 60°, from both directions: (*1*) within 5 s from initiation of roll reversal, with the airplane trimmed as closely as possible to $1.3V_{S1}$, flaps in the takeoff position, and maximum takeoff power; and (*2*) within 4 s from initiation of roll reversal, with the airplane trimmed as closely as possible to $1.3V_{SO}$, flaps fully extended, and engine at idle.

4.5.3.2 With and without flaps deployed, rapid entry into, or recovery from, a maximum cross-controlled slip shall not result in uncontrollable flight characteristics.

4.5.3.3 Lateral and directional control forces shall not reverse with increased deflection.

4.5.4 *Static Longitudinal Stability*:

4.5.4.1 The airplane shall demonstrate the ability to trim for steady flight at speeds appropriate to the climb, cruise, and landing approach configurations; at minimum and maximum weight; and forward and aft CG limits.

4.5.4.2 The airplane shall exhibit positive longitudinal stability characteristics at any speed above V_{S1}, up to the maximum allowable speed for the configuration being investigated, and at the most critical power setting and CG combination.

4.5.4.3 Stability shall be shown by a tendency for the airplane to return toward trimmed steady flight after: (*1*) a "push" from trimmed flight that results in a speed increase, followed by a non-abrupt release of the pitch control; and (*2*) a "pull" from trimmed flight that results in a speed decrease, followed by a non-abrupt release of the pitch control.

4.5.4.4 The airplane shall demonstrate compliance with this section while in trimmed steady flight for each flap and power setting appropriate to the following configurations: (*1*) climb (flaps set as appropriate and maximum continuous power); (*2*) cruise (flaps retracted and 75 % maximum continuous power); and (*3*) approach to landing (flaps fully extended and engine at idle).

4.5.4.5 While returning toward trimmed steady flight, the airplane shall: (*1*) not decelerate below stalling speed V_{S1}; (*2*) not exceed V_{NE} or the maximum allowable speed for the configuration being investigated; and (*3*) exhibit decreasing amplitude for any long-period oscillations.

4.5.5 *Static Directional and Lateral Stability*:

4.5.5.1 The airplane must maintain a trimmed condition around the roll and yaw axis with respective controls fixed.

4.5.5.2 The airplane shall exhibit positive directional and lateral stability characteristics at any speed above V_{S1}, up to the maximum allowable speed for the configuration being investigated, and at the most critical power setting and CG combination.

4.5.5.3 Directional stability shall be shown by a tendency for the airplane to recover from a skid condition after release of the yaw control.

4.5.5.4 Lateral stability shall be shown by a tendency for the airplane to return toward a level-wing attitude after release of the roll control from a slip condition.

4.5.5.5 The airplane shall demonstrate compliance with this section while in trimmed steady flight for each flap and power setting appropriate to the following configurations: (*1*) climb (flaps as appropriate and maximum continuous power); (*2*) cruise (flaps retracted and 75 % maximum continuous power); and (*3*) approach to landing (flaps fully extended and engine at idle).

4.5.6 *Dynamic Stability*—Any oscillations shall exhibit decreasing amplitude within the appropriate speed range (V_{SO} to V_{FE} flaps extended and V_S to V_{DF} flaps retracted).

4.5.7 *Wings Level Stall*—It shall be possible to prevent more than 20° of roll or yaw by normal use of the controls during the stall and the recovery at all weight and CG combinations.

4.5.8 *Turning Flight and Accelerated Stalls*:

4.5.8.1 Turning flight and accelerated stalls shall be performed in both directions as follows: after establishing a 30° coordinated turn, the turn shall be tightened until the stall. After the turning stall, level flight shall be regained without exceeding 60° of additional roll in either direction. No excessive loss of altitude, nor tendency to spin, nor speed buildup shall be associated with the recovery. The rate of speed reduction must be constant, and may not exceed 1 kts/s for a turning flight stall, and be 3 to 5 kts/s with steadily increasing load factor for an accelerated stall.

4.5.8.2 Both turning flight and accelerated stalls shall be performed: (*1*) with flaps retracted, at 75 % maximum continuous power and at idle; and (*2*) with flaps extended, at 75 % maximum continuous power and at idle (speed not to exceed V_{FE}).

4.5.9 *Spinning*:

4.5.9.1 For airplanes placarded "no intentional spins," the airplane must be able to recover from a one-turn spin or a 3-s spin, whichever takes longer, in not more than one additional turn, with the controls used in the manner normally used for recovery.

4.5.9.2 For airplanes in which intentional spinning is allowed, the airplane must be able to recover from a three-turn spin in not more than one and one-half additional turn.

4.5.9.3 In addition, for either 4.5.9.1 or 4.5.9.2:

(*1*) For both the flaps-retracted and flaps-extended conditions, the applicable airspeed limit and limit maneuvering load factor may not be exceeded.

(2) There may be no excessive control forces during the spin or recovery.

(3) It must be impossible to obtain uncontrollable spins with any use of the controls.

(4) For the flaps-extended condition, the flaps may be retracted during recovery.

4.5.9.4 For those airplanes of which the design is inherently spin resistant, such resistance must be proven by test and documented. If proven spin resistant, the airplane must be placarded "no intentional spins" but need not comply with 4.5.9.1-4.5.9.3.

4.6 *Vibrations*—Flight testing shall not reveal, by pilot observation, heavy buffeting (except as associated with a stall), excessive airframe or control vibrations, flutter (with proper attempts to induce it), or control divergence, at any speed from V_{SO} to V_{DF}.

4.7 *Ground Control and Stability*:

4.7.1 It must be possible to taxi, takeoff, and land while maintaining control of the airplane, up to the maximum crosswind component specified within the POH.

4.7.2 Wheel brakes must operate so as not to cause unpredictable airplane response or control difficulties.

5. Structure

5.1 *General*:

5.1.1 *Loads*:

5.1.1.1 Strength requirements are specified in terms of limit loads (the maximum loads to be expected in service) and ultimate loads (limit loads multiplied by prescribed factors of safety). Unless otherwise provided, prescribed loads are limit loads.

5.1.1.2 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.

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

5.1.1.4 The simplified structural design criteria given in Appendix X1 may be used for airplanes with conventional configurations. If Appendix X1 is used, the entire appendix must be substituted for the corresponding paragraphs of this subpart, that is, 5.2.1 to 5.7.3. Appendix X2 contains acceptable methods of analysis that may be used for compliance with the loading requirements for the wings and fuselage.

5.1.2 *Factor of Safety*:

5.1.2.1 Unless otherwise provided in 5.1.2.2, an ultimate load factor of safety of 1.5 must be used.

5.1.2.2 Special ultimate load factors of safety shall be applied to the following:

2.0 × 1.5 = 3.0	on castings
1.2 × 1.5 = 1.8	on fittings
2.0 × 1.5 = 3.0	on bearings at bolted or pinned joints subject to rotation
4.45 × 1.5 = 6.67	on control surface hinge-bearing loads except ball and roller bearing hinges
2.2 × 1.5 = 3.3	on push-pull control system joints
1.33 × 1.5 = 2	on cable control system joints, seat belt/harness fittings (including the seat if belt/harness is attached to it)

5.1.3 *Strength and Deformation*:

5.1.3.1 The structure must be able to support limit loads without permanent deformation. At any load up to limit loads, the deformation may not interfere with safe operation.

5.1.3.2 The structure must be able to support ultimate loads without failure for at least 3 s. However, when proof of strength is shown by dynamic tests simulating actual load conditions, the 3-s limit does not apply.

5.1.4 *Proof of Structure*—Each design requirement must be verified by means of conservative analysis or test (static, component, or flight), or both.

5.1.4.1 Compliance with the strength and deformation requirements of 5.1.3 must be shown for each critical load condition. Structural analysis may be used only if the structure conforms to those for which experience has shown this method to be reliable. In other cases, substantiating load tests must be made. Dynamic tests, including structural flight tests, are acceptable if the design load conditions have been simulated. Substantiating load tests should normally be taken to ultimate design load.

5.1.4.2 Certain parts of the structure must be tested as specified in 6.9.

5.2 *Flight Loads*:

5.2.1 *General*:

5.2.1.1 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.

5.2.1.2 Compliance with the flight load requirements of this section must be shown at each practicable combination of weight and disposable load within the operating limitations specified in the POH.

5.2.2 *Symmetrical Flight Conditions*:

5.2.2.1 The appropriate balancing horizontal tail loads 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 5.2.2 to 5.2.6.

5.2.2.2 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.

5.2.2.3 In computing the loads arising in the conditions prescribed above, the angle of attack is assumed to be changed suddenly without loss of air speed until the prescribed load factor is attained. Angular accelerations may be disregarded.

5.2.2.4 The aerodynamic data required for establishing the loading conditions must be verified by tests, calculations, or by conservative estimation. In the absence of better information, the maximum negative lift coefficient for rigid lifting surfaces may be assumed to be equal to –0.80. If the pitching moment coefficient, C_{mo}, is less than ±0.025, a coefficient of at least ±0.025 must be used.

5.2.3 *Flight Envelope*—Compliance shall be shown at any combination of airspeed and load factor on the boundaries of the flight envelope. The flight envelope represents the envelope of the flight loading conditions specified by the criteria of 5.2.4 and 5.2.5 (see Fig. 1).

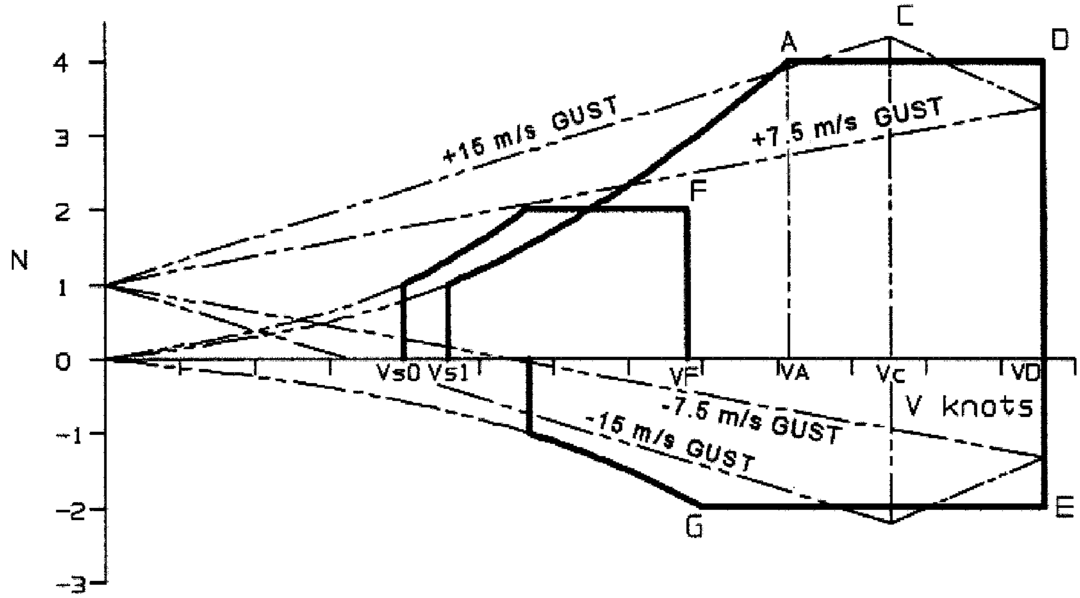


FIG. 1 Flight Envelope

5.2.3.1 *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 Fig. 1 that represents the envelope of the flight loading conditions specified by the maneuvering and gust criteria of 5.2.5 and 5.2.6 respectively.

5.2.3.2 *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: (*1*) the positive maneuvering load factor specified in 5.2.5.1 at speeds up to V_D; and (*2*) the negative maneuvering load factor specified in 5.2.5.2 at speeds up to V_D.

5.2.3.3 *Gust Envelope*—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 m/s (49.2 ft/s) at V_C; and (*2*) positive and negative gusts of 7.5 m/s (24.6 ft/s) at V_D (see Fig. 1).

5.2.4 *Design Airspeeds*:

5.2.4.1 *Design Maneuvering Speed, V_A*:

$$V_A = V_S \cdot \sqrt{n1}$$

$$V_S = \sqrt{\frac{W}{\frac{1}{2}\rho C_{LMAX} S}}, (m/s) = 2.484\sqrt{\frac{W}{C_{LMAX} S}}\,(kts)$$

where:

V_S = computed stalling speed at the design maximum weight with the flaps retracted, and

$n1$ = positive limit maneuvering load factor used in design.

5.2.4.2 *Design Flap Speed, V_F*—For each landing setting, V_F must not be less than the greater of: (*1*) 1.4 V_S, where V_S is the computed stalling speed with the wing flaps retracted at the maximum weight; and (*2*) 2.0 V_{SO}, where V_{SO} is the computed stalling speed with wing flaps fully extended at the maximum weight.

5.2.4.3 *Design Cruising Speed, V_C*—(*1*) V_C in knots may not be less than $4.77\sqrt{W/S}$; and (*2*) V_C need not be greater than 0.9 V_H at sea level.

5.2.4.4 *Design Dive Speed, V_D*:

$$V_D = 1.4 \times V_{C\,min}$$

where:

$V_{C\,min}$ = required minimum cruising speed.

5.2.5 *Limit Maneuvering Load Factors*:

5.2.5.1 The positive limit maneuvering load factor n_1 may not be less than 4.0.

5.2.5.2 The negative limit maneuvering load factor n_2 may not be greater than −2.0.

5.2.5.3 Loads with wing flaps extended: (*1*) if flaps or other similar high lift devices are used, the airplane must be designed for $n_1 = 2.0$ with the flaps in any position up to V_F; and (*2*) $n_2 = 0$.

5.2.5.4 Loads with speed control devices: (*1*) if speed control devices such as speed brakes or spoilers are used, the airplane must be designed for a positive limit load factor of 3.0 with the devices extended in any position up to the placard device extended speed; and (*2*) maneuvering load factors lower than those specified in 5.2.5 may be used if the airplane has design features that make it impossible to exceed these in flight.

5.2.6 *Gust Load Factors*—The airplane must be designed for the loads resulting from:

5.2.6.1 The gust velocities specified in 5.2.3.3 with flaps retracted, and

5.2.6.2 Positive and negative gusts of 7.5 m/s (24.6 ft/s) nominal intensity at V_F with the flaps fully extended.

NOTE 3—In the absence of a more rational analysis, the gust load factors may be computed by the method of Appendix X3.

5.2.7 *Unsymmetrical Flight Conditions*—The airplane is assumed to be subjected to the unsymmetrical flight conditions of 5.2.7.1 and 5.2.7.2. Unbalanced aerodynamic moments about the center of gravity must be reacted in a rational or conservative manner considering the principle masses furnishing the reacting inertia forces.

5.2.7.1 *Rolling Conditions*—The airplane shall be designed for the loads resulting from the roll control deflections and speeds specified in 5.7.1 in combination with a load factor of at least two thirds of the positive maneuvering load factor prescribed in 5.2.5.1. The rolling accelerations may be obtained by the methods given in X2.3. The effect of the roll control displacement on the wing torsion may be accounted for by the method of X2.3.2 and X2.3.3.

5.2.7.2 *Yawing Conditions*—The airplane must be designed for the yawing loads resulting from the vertical surface loads specified in 5.5.

5.2.8 *Special Conditions for Rear Lift Truss*:

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

$$V = 1.26\sqrt{\frac{W}{S}} + 8.7, (kts)$$

where:

W/S = wing loading, N/m^2.

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

5.2.9 *Engine Torque*—The engine mount and its supporting structure must be designed for the effects of:

5.2.9.1 The limit torque corresponding to takeoff power and propeller speed acting simultaneously with 75 % of the limit loads from flight condition of 5.2.5.1.

5.2.9.2 The limit torque corresponding to maximum continuous power and propeller speed acting simultaneously with the limit loads from flight condition of 5.2.5.1.

5.2.9.3 For conventional reciprocating engines with positive drive to the propeller, the limit torque to be accounted for in 5.2.9.1 and 5.2.9.2 is obtained by multiplying the mean torque by one of the following factors:

For four-stroke engines:

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

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

For two-stroke engines:

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

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

5.2.10 *Side Load on Engine Mount*:

5.2.10.1 The 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 1.5.

5.2.10.2 The side load prescribed in 5.2.10.1 may be assumed to be independent other flight conditions.

5.2.10.3 If applicable, the nose wheel loads of 5.8.1.7 must also be considered.

5.3 *Control Surface and System Loads*:

5.3.1 *Control Surface Loads*—The control surface loads specified in 5.3.3 through 5.7.3 are assumed to occur in the conditions described in 5.2.2 through 5.2.6.

5.3.2 *Control System Loads*—Each part of the primary control system situated between the stops and the control surfaces must be designed for the loads corresponding to at least 125 % of the of the computed hinge moments of the movable control surfaces resulting from the loads in the conditions prescribed in 5.3.1 through 5.7.3. In computing the hinge moments, reliable aerodynamic data must be used. In no case may the load in any part of the system be less than those resulting from the application of 60 % of the pilot forces described in 5.3.3. In addition, the system limit loads need not exceed the loads that can be produced by the pilot. Pilot forces used for design need not exceed the maximum pilot forces prescribed in 5.3.3.

5.3.3 *Loads Resulting from Limit Pilot Forces*—The main control systems for the direct control of the airplane about its longitudinal, lateral, or yaw axis, including the supporting points and stops, must be designed for the limit loads resulting from the limit pilot forces as follows:

5.3.3.1 *Pitch*—445 N (100 lb) at the grips of the stick or wheel.

5.3.3.2 *Roll*—180 N (40.5 lb) at the grip(s) of the stick or wheel.

5.3.3.3 *Yaw*—580 N (130 lb) acting forward on one rudder pedal.

5.3.3.4 The rudder control system must be designed to a load of 580 N (130 lb) per pedal acting simultaneously on both pedals in the forward direction.

5.3.4 *Dual-Control Systems*—Dual-control systems must be designed for the loads resulting from each pilot applying 0.75 times the load specified in 5.3.3 with the pilots acting in opposition.

5.3.5 *Secondary Control Systems*—Secondary control systems, such as those for flaps and trim control must be designed for the maximum forces that a pilot is likely to apply.

5.3.6 *Control System Stiffness and Stretch*—The amount of control surface or tab movement available to the pilot shall not be dangerously reduced by elastic stretch or shortening of the system in any condition.

5.3.7 *Ground Gust Conditions*—The control system from the control surfaces to the stops or control locks, when installed, must be designed for limit loads due to gusts corresponding to the following hinge moments:

$$M_S = k \cdot C_S \cdot S_S \cdot q \quad (1)$$

where:

M_S = limit hinge moment,
C_S = mean chord of the control surface aft of the hinge line,
S_S = area of the control surface aft of the hinge line,
q = dynamic pressure corresponding to an airspeed of 38 kts, and
k = limit hinge moment coefficient due to ground gust = 0.75.

5.3.8 *Control Surface Mass Balance Weights*—If applicable shall be designed for:

5.3.8.1 The $n = 16$ limit load normal to the surface, and

5.3.8.2 The $n = 8$ limit load fore and aft and parallel to the hinge line.

5.3.9 The motion of wing flaps on opposite sides of the plane of symmetry must be synchronized by a mechanical interconnection unless the airplane has safe flight characteristics with the wing flaps retracted on one side and extended on the other.

5.3.10 All primary controls shall have stops within the system to withstand the greater of pilot force, 125 % of surface loads, or ground gust loads (see 5.3.7).

5.4 *Horizontal Stabilizing and Balancing Surfaces*:

5.4.1 *Balancing Loads*:

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

5.4.1.2 Horizontal stabilizing surfaces must be designed for the balancing loads occurring at any point on the limit maneuvering envelope and in the air-brake and wing-flap positions specified in 5.2.5.3.

5.4.2 *Maneuvering Loads*—Horizontal stabilizing surfaces must be designed for pilot-induced pitching maneuvers imposed by the following conditions:

5.4.2.1 At speed V_A, maximum upward deflection of pitch control surface,

5.4.2.2 At speed V_A, maximum downward deflection of pitch control surface,

5.4.2.3 At speed V_D, one-third maximum upward deflection of pitch control surface, and

5.4.2.4 At speed V_D, one-third maximum downward deflection of pitch control surface.

NOTE 4—In 5.4.2, the following assumptions should be made: the airplane is initially in level flight, and its altitude and airspeed do not change. The loads are balanced by inertia forces.

5.4.3 *Gust Loads*—The horizontal stabilizing surfaces must be designed for the loads resulting from:

5.4.3.1 The gust velocities specified in 5.2.3.3 with flaps retracted, and

5.4.3.2 Positive and negative gusts of 7.5 m/s (24.6 ft/s) nominal intensity at V_F with the flaps fully extended.

NOTE 5—In the absence of a more rational analysis, the horizontal surfaces gust loads may be computed by the method of Appendix X4.

5.5 *Vertical Stabilizing Surfaces*:

5.5.1 *Maneuvering Loads*—The vertical stabilizing surfaces must be designed for maneuvering loads imposed by the following conditions:

5.5.1.1 At speed V_A, full deflection of the yaw control in both directions.

5.5.1.2 At speed V_D, one-third full deflection of the yaw control in both directions.

5.5.2 *Gust Loads*:

5.5.2.1 The vertical stabilizing surfaces must be designed to withstand lateral gusts of the values prescribed in 5.2.3.3.

NOTE 6—In the absence of a more rational analysis, the vertical surfaces gust loads may be computed by the method in Appendix X4.2.

5.5.3 *Outboard Fins or Winglets*:

5.5.3.1 If outboard fins or winglets are 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.

5.5.3.2 If outboard fins or winglets extend above and below the horizontal surface, the critical vertical surface loading (the load per unit area determined in accordance with 5.5.1 and 5.5.2) must be applied to:

(1) The part of the vertical surface above the horizontal surface with 80 % of that loading applied to the part below the horizontal surface or wing, and

(2) The part of the vertical surface below the horizontal surface or wing with 80 % of that loading applied to the part above the horizontal surface or wing.

5.5.3.3 The end plate effects of outboard fins or winglets must be taken into account in applying the yawing conditions of 5.5.1 and 5.5.2 to the vertical surfaces in 5.5.3.2.

5.5.3.4 When rational methods are used for computing loads, the maneuvering loads of 5.5.1 on the vertical surfaces and the $n = 1$ horizontal surface or wing load, including induced loads on the horizontal surface or wing and moments or forces exerted on the horizontal surfaces or wing, must be applied simultaneously for the structural loading condition.

5.6 *Supplementary Conditions for Stabilizing Surfaces*:

5.6.1 *Combined Loads on Stabilizing Surfaces*:

5.6.1.1 With the airplane in a loading condition corresponding to A or D in Fig. 1 (whichever condition leads to the higher balance load) the loads on the horizontal surface must be combined with those on the vertical surface as specified in 5.5.1. It must be assumed that 75 % of the loads according to 5.4.2 for the horizontal stabilizing surface and 5.5.1 for the vertical stabilizing surface are acting simultaneously.

5.6.1.2 The stabilizing surfaces and fuselage must be designed for asymmetric loads on the stabilizing surfaces which would result from application of the highest symmetric maneuver loads of 5.5.1 so that 100 % of the horizontal stabilizer surface loading is applied to one side of the plane symmetry and 70 % on the opposite side.

5.6.2 *Additional Loads Applying to V-Tails*—An airplane with a V-tail must be designed for a gust acting perpendicular to one of the surfaces at speed V_C. This condition is supplemental to the equivalent horizontal and vertical cases previously specified.

5.7 *Ailerons, Wing Flaps, and Special Devices*:

5.7.1 *Ailerons*—The ailerons must be designed for control loads corresponding to the following conditions:

5.7.1.1 At speed V_A, the full deflection of the roll control.

5.7.1.2 At speed V_D, one-third of the full deflection of the roll control.

5.7.2 *Flaps*—Wing flaps, their operating mechanisms, and supporting structure must be designed for the critical loads occurring in the flaps-extended operating range with the flaps in any position. The effects of propeller slipstream, corresponding to takeoff power, must be taken into account at a airspeed of not less than 1.4 V_S, where V_S is the computed stalling speed with flaps fully retracted at the design weight. For investigating the slipstream effects, the load factor may be assumed to be 1.0.

5.7.3 *Special Devices*—The loadings for special devices using aerodynamic surfaces, such as slots and spoilers, must be determined from test data or reliable aerodynamic data that allows close estimates.

5.8 *Ground Load Conditions*:

5.8.1 *Basic Landing Conditions*—The requirements for the basic landing conditions are given in 5.8.1.1 to 5.8.1.3, Table 2, and Fig. 2.

5.8.1.1 The load factor on the wheels, n_j, may be computed as follows:

$$n_j = \frac{h + \frac{d}{3}}{ef \times d}$$

where:

h	=	drop height, (cm) = $1.32\sqrt{W/S}$ with W/S in N/m^2, but h larger than 23 cm (9.1 in.),
d	=	total shock absorber travel, cm = $d_{tire} + d_{shock}$,
ef	=	shock efficiency, and
$ef \times d$	=	$0.5 \times d$ for tire and rubber or spring shocks, or
	=	$0.5 \times d_{tire} + 0.65 \times d_{shock}$ for hydraulic shock absorbers.

5.8.1.2 If n_j is larger than 3.33, all concentrated masses (engine, fuel tanks, occupant seats, ballast, etc.) must be substantiated for a limit landing load factor of $n_j + 0.67 = n$ which is greater than 4.

5.8.1.3 The usual ultimate factor of safety of 1.5 applies to these conditions, unless a drop test from the reserve energy height, $hr = 1.44h$, shows that a lower factor may be used. If the shock absorber is of a fast energy absorbing type, the ultimate loads are the limit load multiplied by the conservative reserve energy factor of 1.2.

5.8.1.4 *Side Load Conditions*—The requirements for the side load conditions on the main wheels in a level attitude are given in Fig. 3.

5.8.1.5 *Braked Roll Conditions*—The requirements for the braked roll conditions on the main wheels in a level attitude are given in Fig. 4.

5.8.1.6 *Supplementary Conditions for Tail Wheel*—The requirements for the tail wheel conditions in a tail down attitude are given in Fig. 5.

5.8.1.7 *Supplementary Conditions for Nose Wheel*—The requirements for supplementary conditions for nose wheels are given in Fig. 6 (the static load is at the combination of weight and CG that gives the maximum loads).

5.8.1.8 For the conditions in 5.8.1.4 to 5.8.1.7, the shock absorbers and tires are assumed to be in their static position.

5.9 *Water Load Conditions*:

5.9.1 The structure of seaplanes and amphibians must be designed for water loads developed during takeoff and landing with the airplane in any attitude likely to occur in normal operations at appropriate forward and sinking velocities under the most severe sea conditions likely to be encountered. Unless sufficient satisfactory service experience is available, a rational analysis of the water loads, or the methods specified in Appendix X5 may be used.

5.10 *Emergency Landing Conditions*:

5.10.1 The structure must be designed to protect each occupant during emergency landing conditions when occupants (through seat belts or harnesses, or both) as well as any concentrated weight located behind or above the occupant (such as engine, baggage, fuel, ballast, and so forth), experience the static inertia loads corresponding to the following ultimate load factors (these are three independent conditions):

5.10.1.1 $n = 3$ up,

5.10.1.2 $n = 9$ ($n = 10$ for engines) forward, and

5.10.1.3 $n = 1.5$ lateral.

5.11 *Other Loads*:

5.11.1 *Tie-Down Points*—Tie-down points shall be designed for the maximum wind at which the airplane may be tied down in the open. V_R = 38 kts minimum as in accordance with 5.3.7 may be used.

5.11.2 *Parachute System Loads*—If the aircraft is to be equipped with an emergency parachute system (Ballistic Recovery System), the attachment point(s) to the airframe must be designed in accordance with Specification F 2316.

TABLE 2 Basic Landing Conditions

NOTE 1—$K = 0.25$
$L = 2/3$ = ratio of the assumed wing lift to the airplane weight
$n = n_j + 2/3$ = load factor
n_j = load factor on wheels in accordance with 5.8.1

NOTE 2—See Fig. 2 for the airplane 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
Vertical component at CG	nW	nW	nW	nW	nW
Fore and aft component at CG	KnW	0	KnW	KnW	0
Lateral component in either direction at CG	0	0	0	0	0
Shock absorber deflection (rubber or spring shock absorber), %	100 %	100 %	100 %	100 %	100 %
Tire deflection	Static	Static	Static	Static	Static
Main wheel loads (V_r)	(n-L)W	(n-L)Wb/d	(n–L)Wa'/d'	(n-L)W	(n-L)W
(both wheels) (D_r)	KnW	0	KnWa'/d'	KnW	0
Tail (nose) wheels (V_r)	0	(n-L)Wa/d	(n-L)W'/d'	0	0
Loads (D_r)	0	0	KnWb'/d'	0	0

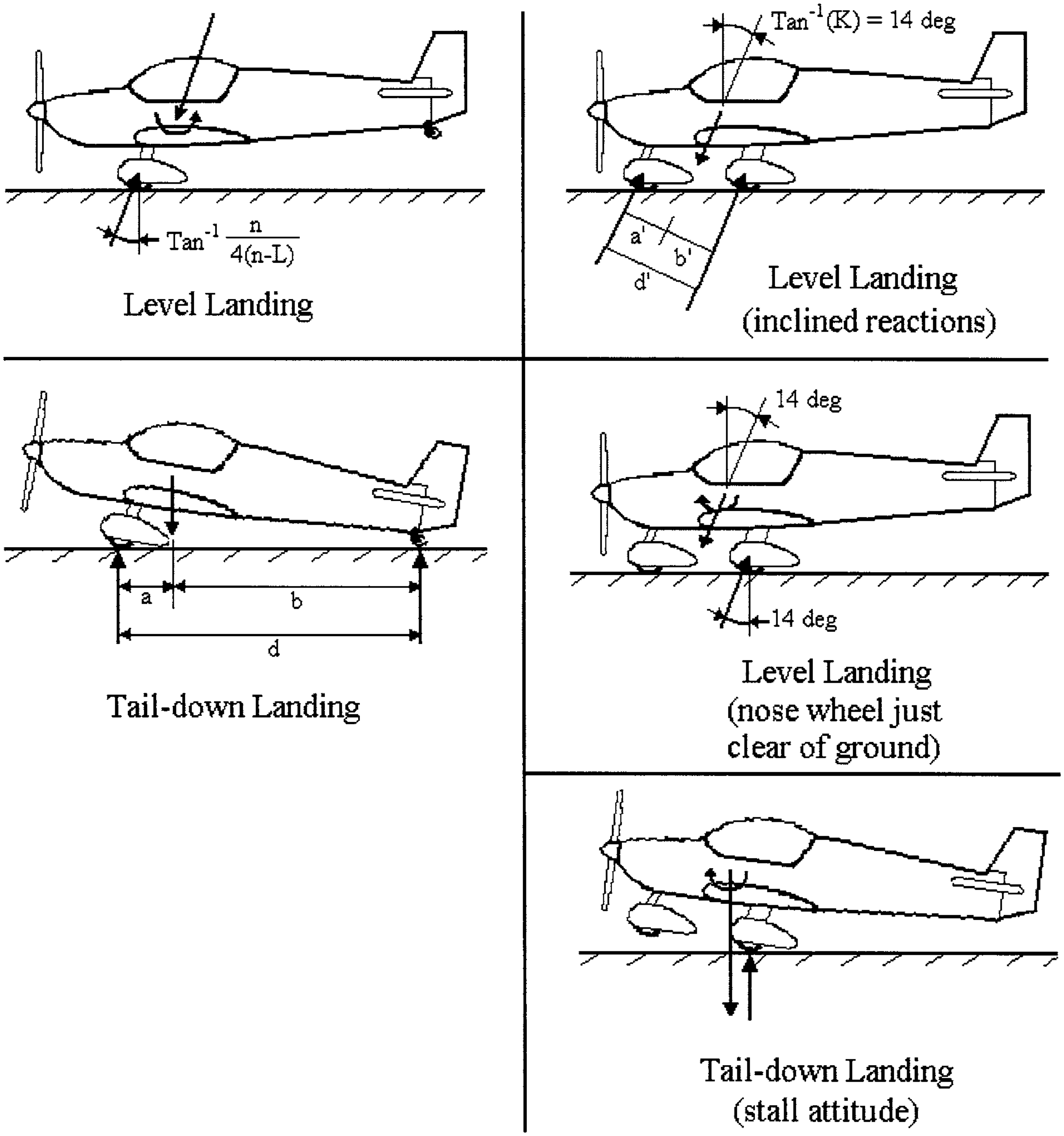


FIG. 2 Basic Landing Conditions

5.11.3 *Loads from Single Masses*—The attachment means for all single masses which are part of the equipment for the airplane must be designed to withstand loads corresponding to the maximum design load factors to be expected from the established flight and ground loads, including the emergency landing conditions of 5.10.

6. Design and Construction

6.1 *General*—The integrity of any novel or unusual design feature having an important bearing on safety shall be established by test.

6.2 *Materials*—Materials shall be suitable and durable for the intended use. Design values (strength) must be chosen so that no structural part is under strength as a result of material variations or load concentration, or both.

6.3 *Fabrication Methods*—Manufactured parts, assemblies, and completed airplanes shall be produced in accordance with the manufacturer's quality assurance and production acceptance test procedures.

6.4 *Self-Locking Nuts*—No self-locking nut shall be used on any bolt subject to rotation in operation unless a nonfriction locking device is used in addition to the self-locking device.

6.5 *Protection of Structure*—Protection of the structure against weathering, corrosion, and wear, as well as suitable ventilation and drainage, shall be provided as required.

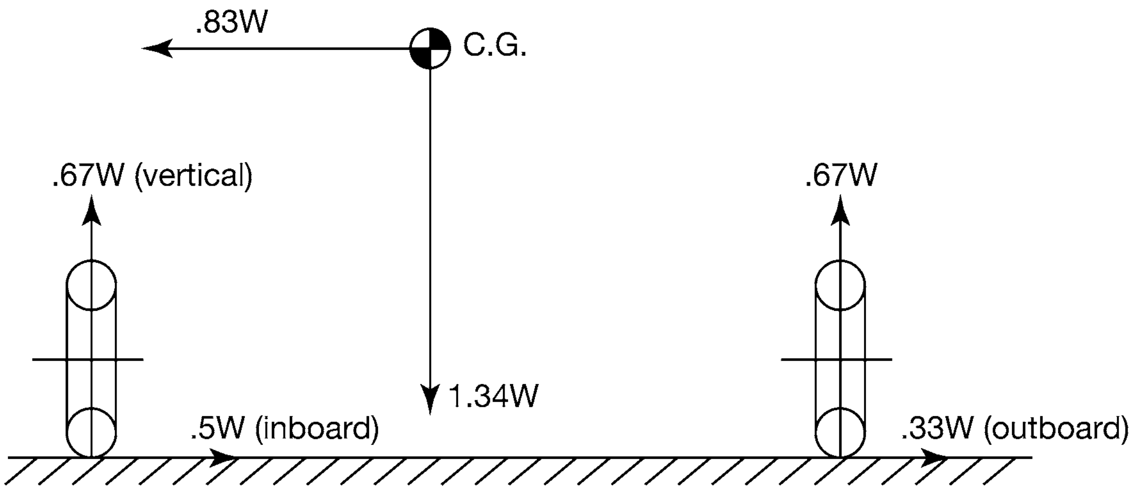


FIG. 3 Side Load Conditions

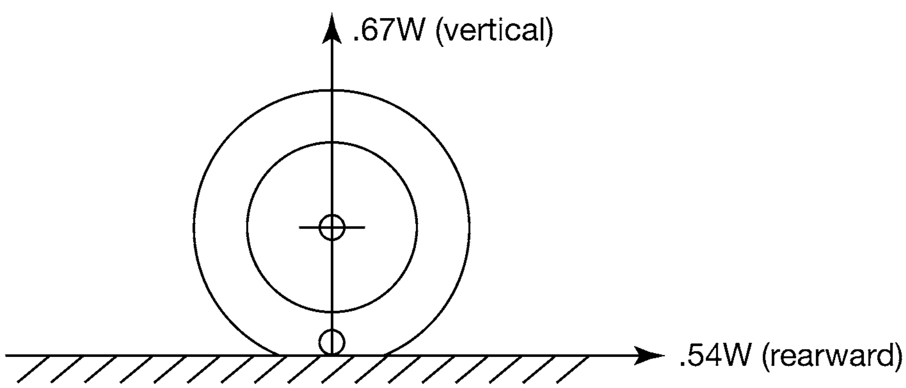


FIG. 4 Braked Roll Conditions

6.6 *Accessibility*—Accessibility for critical structural elements and control system inspection, adjustment, maintenance, and repair shall be provided.

6.7 *Rigging*—Unless specified otherwise, rigging and derigging must be able to be performed by persons having no more than average skill. It must be possible to inspect the airplane easily for correct rigging and safe-tying.

6.8 *Proof of Design*—Fulfillment of the design requirements for the airplane shall be determined by conservative analysis, or tests, or a combination of both. Structural analysis alone may be used for validation of the structural requirements only if the structure conforms to those for which experience has shown this method to be reliable. Flight tests to limit load factors at maximum takeoff weight and at speeds from V_A to the maximum allowable speed for the configuration being investigated are an acceptable proof (see 5.1.3 and 5.1.4).

6.9 *Control System-Operation Test*—It must be shown by functional tests that the control system installed on the airplane is free from interference, jamming, excessive friction, and excessive deflection when the control system design loads (see 5.3) are applied to the controls and the surfaces. The control system stops must withstand those loads.

6.10 *Pilot Compartment*:

6.10.1 Pilot comfort, appropriate visibility (instruments, placards, and outside), accessibility, ability to conduct an emergency escape, and ability to reach all controls for smooth and positive operation shall be provided.

6.10.2 Occupant seat belts, harnesses, and baggage restraints, and their attachments to the structure shall be designed for the maximum load factors corresponding to the specified ground and flight conditions including the emergency landing conditions prescribed in 5.10.

7. Powerplant

7.1 *Installation*—The powerplant installation shall be easily accessible for inspection and maintenance. The powerplant attachment to the airframe is part of the structure and shall withstand the applicable load factors.

7.2 *Engines*—Installed engines shall meet Practice F 2339, LSA engine design and production standards, or shall be type and production certified under FAR-33, JAR-E, or JAR-22 Subpart H, design and production standards.

NOTE 7—Type certified engines may be subject to additional regulatory

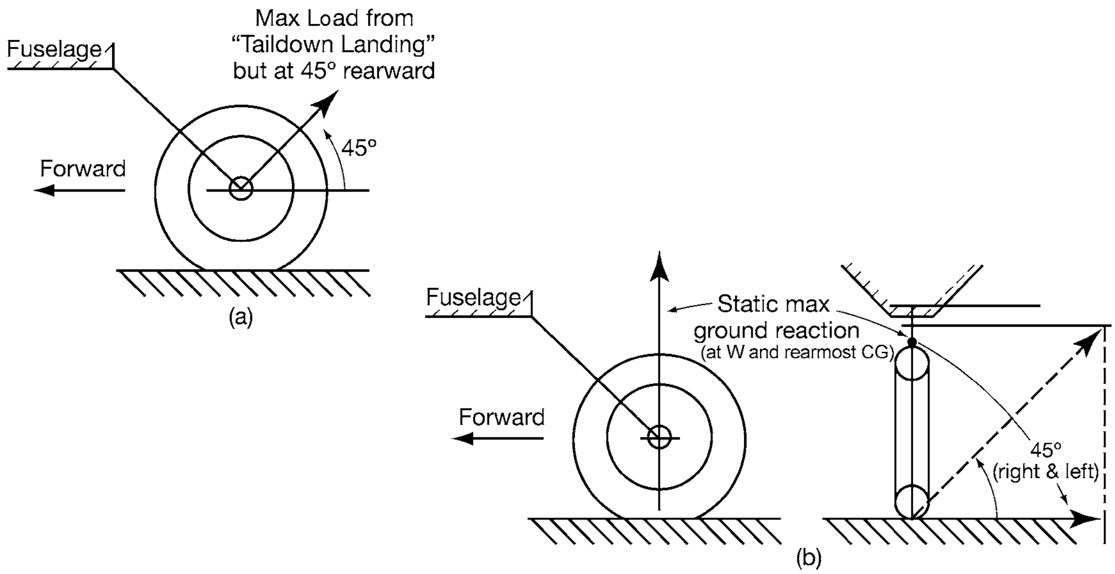


FIG. 5 Supplementary Conditions for Tail Wheel

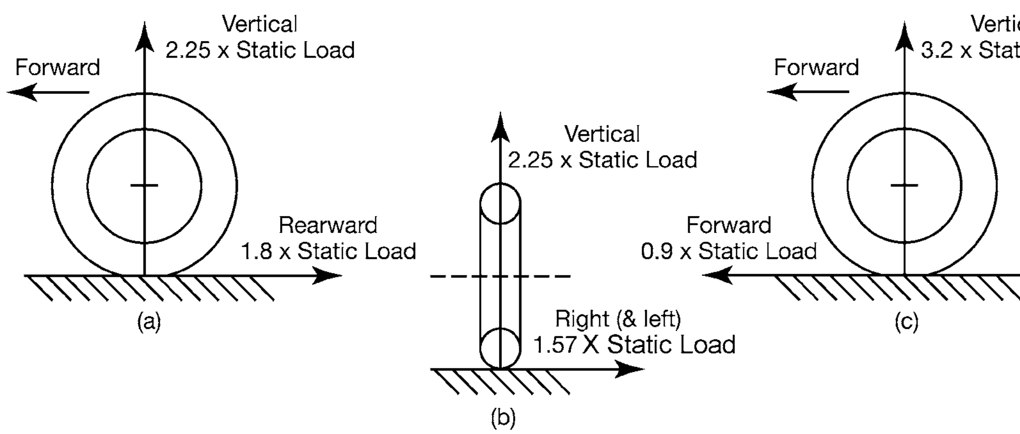


FIG. 6 Supplementary Conditions for Nose Wheel

maintenance requirements.

7.3 *Fuel System*:

7.3.1 The unusable fuel quantity for each tank must be established by tests and shall not be less than the quantity at which the first evidence of engine fuel starvation occurs under each intended flight operation and maneuver.

7.3.2 Tanks must be protected against wear from vibrations and their installation shall be able to withstand the applicable inertia loads.

7.3.3 Fuel tanks shall be designed to withstand a positive pressure of 24.5 kPa (3.55 psi) (2.5-m (8.2-ft) water column).

7.3.4 The filler must be located outside the passenger compartment and spilled fuel must be prevented from entering or accumulating in any enclosed part of the airplane.

7.3.5 Each tank must be vented. The vent must discharge clear of the airplane.

7.3.6 There must be at least one drain to allow safe drainage. A drainable sediment bowl located at the lowest point in the fuel system may be used instead of the drainable sump in the fuel tank.

7.3.7 A fuel strainer or filter accessible for cleaning and replacement must be included in the system.

7.3.8 The fuel lines must be properly supported and protected from vibrations and wear.

7.3.9 Fuel lines located in an area subject to high heat (engine compartment) must be fire resistant or protected with a fire-resistant covering.

7.3.10 There must be a fuel shutoff valve accessible to the pilot while wearing a seat belt or harness.

7.4 *Oil System*—If an engine is provided with an oil system, it must be:

7.4.1 Capable of supplying the engine with an adequate quantity of oil at a temperature not exceeding the maximum established by the engine manufacturer, and

7.4.2 The oil tank or radiator, or both, must be installed to withstand the applicable inertia loads and vibrations, and the oil breather (vent) must be resistant to blockage caused by icing. Oil foam from the breather shall not constitute a hazard.

7.5 *Induction System*—The engine air induction system shall be designed to minimize the potential of carburetor icing.

7.6 *Fire Prevention*—The engine, if enclosed, must be isolated from the rest of the airplane by a firewall or shroud. It must be constructed as far as practical to prevent liquid, gas, or flames, or a combination thereof, from entering the airplane. The use of any one of the following materials shall be acceptable without further testing:

7.6.1 Stainless steel, not less than 0.46 mm (0.018 in.) thick,

7.6.2 Mild steel (corrosion-protected), not less than 0.46 mm (0.018 in.) thick, or

7.6.3 Alternative materials that provide protection equivalent to 7.6.1 or 7.6.2.

8. Required Equipment

8.1 The aircraft shall be designed with the following minimum instrumentation and equipment:

8.2 *Flight and Navigation Instruments*:

8.2.1 Airspeed indicator, and

8.2.2 Altimeter.

8.3 *Powerplant Instruments*:

8.3.1 Fuel quantity indicator,

8.3.2 Tachometer (RPM),

8.3.3 Engine "kill" switch, and

8.3.4 Engine instruments as required by the engine manufacturer.

8.4 *Miscellaneous Equipment*:

8.4.1 If installed, an electrical system shall include a master switch and overload protection devices (fuses or circuit breakers).

8.4.2 The electric wiring shall be sized according to the load of each circuit.

8.4.3 The battery installation shall withstand all applicable inertia loads.

8.4.4 Battery containers shall be vented outside of the airplane (see 6.5).

8.5 *Safety Belts and Harnesses*—There must be a seat belt and harness for each occupant and adequate means to restrain the baggage.

9. Pilot Operating Handbook

9.1 Each airplane shall include a Pilot Operating Handbook (POH). The POH shall contain at least the following section headings and related information when applicable to a specific airplane and shall be listed in the order shown as follows. All flight speeds shall be presented as calibrated airspeeds (CAS) and all specifications and limitations shall be those determined from the preceding relative design criteria.

9.2 *General Information*:

9.3 *Airplane and Systems Descriptions*:

9.3.1 Engine,

9.3.2 Propeller,

9.3.3 Fuel and fuel capacity,

9.3.4 Oil, and

9.3.5 Operating weights and loading (occupants, baggage, fuel, ballast).

9.4 *Operating Limitations*:

9.4.1 Stalling speeds at maximum takeoff weight (V_S and V_{S0}),

9.4.2 Flap extended speed range (V_{S0} to V_{FE}),

9.4.3 Maximum maneuvering speed (V_A),

9.4.4 Never exceed speed (V_{NE}),

9.4.5 Crosswind and wind limitations,

9.4.6 Service ceiling,

9.4.7 Load factors, and

9.4.8 Prohibited maneuvers.

9.5 *Weight And Balance Information*:

9.5.1 Installed equipment list, and

9.5.2 Center of gravity (CG) range and determination.

9.6 *Performance*:

9.6.1 Takeoff and landing distances,

9.6.2 Rate of climb,

9.6.3 Cruise speeds,

9.6.4 RPM, and

9.6.5 Fuel consumption.

9.7 *Emergency Procedures*.

9.8 *Normal Procedures*—The following operating procedures and handling information shall be provided:

9.8.1 Preflight check,

9.8.2 Engine starting,

9.8.3 Taxiing,

9.8.4 Normal takeoff,

9.8.5 Best angle of climb speed (V_X),

9.8.6 Best rate of climb speed (V_Y),

9.8.7 Cruise,

9.8.8 Approach,

9.8.9 Normal landing,

9.8.10 Short field takeoff and landing procedures, if any,

9.8.11 Balked landing procedures, and

9.8.12 Information on stalls, spins, and any other useful pilot information.

9.9 *Aircraft Ground Handling and Servicing*:

9.9.1 Servicing fuel, oil, coolant, and

9.9.2 Towing and tie-down instructions.

9.10 *Required Placards and Markings*:

9.10.1 Airspeed indicator range markings,

9.10.2 Operating limitations on instrument panel, if applicable,

9.10.3 Passenger Warning: "This aircraft was manufactured in accordance with Light Sport Aircraft airworthiness standards and does not conform to standard category airworthiness requirements,"

9.10.4 "NO INTENTIONAL SPINS," if applicable, and

9.10.5 Miscellaneous placards and markings.

9.11 *Supplementary Information*:

9.11.1 Familiarization flight procedures, and

9.11.2 Pilot operating advisories, if any.

10. Keywords

10.1 fixed-wing aircraft; light sport airplane

ANNEXES

(Mandatory Information)

A1. ADDITIONAL REQUIREMENTS FOR LIGHT SPORT AIRPLANES USED TO TOW GLIDERS

A1.1 Applicability

A1.1.1 This annex is applicable to light sport airplanes that are to be used to tow gliders.

A1.2 Minimum Climb Performance While Towing

A1.2.1 The aircraft must be capable of achieving a gradient of climb while towing of at least 1⁄18 , while not exceeding the maximum placarded towing speed of the towing aircraft, or the maximum safe towing speed of the aircraft being towed.

A1.2.2 The aircraft must be capable of achieving a rate of climb while towing of at least 46 m/min (150 ft/min), while not exceeding the maximum placarded towing speed of the towing aircraft, or the maximum safe towing speed of the aircraft being towed.

NOTE A1.1—Compliance with this section must take into account the performance and control capabilities of both the towing aircraft and the aircraft being towed. In order to account for varying performance and control capabilities on the part of the towed aircraft, the manufacturer of the towing aircraft may specify a maximum weight and maximum drag for the towed aircraft at each speed for which the towing aircraft is approved for tow operations, such that the required climb performances can be achieved. Compliance with this section is then shown when the towed aircraft is safely controllable under tow at a speed for which its drag and weight are within these prescribed maximum weight and drag limits.

A1.3 Controllability and Maneuverability

A1.3.1 The tow aircraft shall be safely controllable and maneuverable during all ground and flight operations applicable to normal towing operations, including both deliberate and inadvertent release of the glider being towed.

A1.4 Stability

A1.4.1 It shall be possible to conduct normal towing operations, including both deliberate and inadvertent release of the glider being towed, without incurring any dangerous reduction in the stability of the aircraft.

A1.5 Structure and Strength Requirements

A1.5.1 Strength requirements for the aircraft structure shall take into account the effects of loads arising from towing equipment that is installed on the aircraft in accordance with A1.6.

A1.6 Design and Construction

A1.6.1 *Glider Towing Equipment Installations*:

A1.6.1.1 The maximum all up takeoff weight of the glider to be towed, including pilot and all equipment, shall be selected by the manufacturer.

A1.6.1.2 The maximum glider towing speed (V_T), shall be selected by the manufacturer. The V_T shall be at least $1.3V_S$, where V_S is the computed stalling speed of the aircraft in the cruise configuration without a glider in tow.

A1.6.1.3 Tow equipment attach points on the airframe shall have limit and ultimate factors of safety of not less than 1.0 and 1.5 respectively, when loads equal to 1.2 of the nominal strength of the weak link (see A1.6.1.5) are applied through the towing hook installation for the following conditions, simultaneously with the loads arising from the most critical normal accelerations (as defined in the normally applicable requirements for structure and strength) at the speed V_T. The applicable conditions are as follows:

(1) The speed is assumed to be at the maximum glider towing speed V_T, and

(2) The load at the towing hook installation is assumed to be acting in each of the following directions, relative to the longitudinal centerline of the aircraft: horizontally backwards; backwards and upwards at 40° to the horizontal; backwards and downwards at 20° to the horizontal; and horizontally backwards and 25° sideways in both directions.

A1.6.1.4 The towing hook shall be of a quick release type. It shall be established by test that when the release control is operated simultaneously with loads equal to 10 % and 180 % of the nominal strength of the weak link (see A1.6.1.5) applied to the towing hook in each of the directions prescribed in A1.6.1.3(*2*): (*1*) the tow cable will be released; (*2*) the released cable will be unlikely to cause damage to or become entangled with any part of the aircraft; and (*3*) the pilot effort required shall not be less than 20 N (4.5 lb) nor greater than 100 N (22.5 lb).

A1.6.1.5 The release control shall be located so that the pilot can operate it without having to release any other primary flight control.

A1.6.1.6 The maximum strength of any weak link that may be interposed in the towing cable shall be established. For the determination of loads to be applied for the purpose of this section, the strength of the weak link shall not be less than 900 N (202.3 lb).

A1.7 Operating Limitations

A1.7.1 Operating limitations applicable to towing operations must be established and included in the Pilot's Operating Handbook, to include at a minimum:

A1.7.1.1 The maximum permissible towing speed (V_T).

A1.7.1.2 The maximum weak link strength (may be specified in terms of the weight of the glider to be towed).

A1.7.1.3 The maximum permissible all up weight of the glider to be towed.

A2. LIGHT SPORT AIRCRAFT TO BE FLOWN AT NIGHT

A2.1 Applicability

A2.1.1 This annex is applicable to light sport airplanes that are to be flown at night.

A2.2 Flight

A2.2.1 No additional requirements for night operations.

A2.3 Structure

A2.3.1 No additional requirements for night operations.

A2.4 Design and Construction

A2.4.1 No additional requirements for night operations.

A2.5 Powerplant

A2.5.1 A powerplant that has been specifically approved for night operations and complies with Section 7.

A2.6 Required Equipment

A2.6.1 Instrument lights as specified in A2.7.1;

A2.6.2 Position lights as specified in A2.7.2;

A2.6.3 An aviation red or aviation white anti-collision light system specified in A2.7.3;

A2.6.4 If the aircraft is operated for hire, one electric landing light specified in A2.7.4;

A2.6.5 An adequate source of electrical energy for all installed electrical and radio equipment specified in A2.9.2;

A2.6.6 One spare set of fuses, or three spare fuses of each kind required, that are accessible to the pilot in flight if fuses are installed;

A2.6.7 One switch for each: position lights, anti-collision light system, and if installed, landing light, taxi light, and cabin light as specified in A2.9.1; and

A2.6.8 One attitude indicator.

A2.7 Lighting Requirements

A2.7.1 *Instrument Lights*—The instrument lights must:

A2.7.1.1 Make each instrument and control easily readable and discernible;

A2.7.1.2 Be installed so that their direct rays and rays reflected from the windshield or other surface are shielded from the pilot's eyes;

A2.7.1.3 Have dimmer(s) capable of decreasing the intensity of all instrument, radio, and control lighting; and

A2.7.1.4 Have enough distance or insulating material between current carrying parts and the housing so that vibration in flight will not cause shorting.

A2.7.1.5 A cabin dome light is not an instrument light.

A2.7.2 *Position Lights*:

A2.7.2.1 *General*—Each part of each position light system must meet the applicable requirements of this specification and each system as a whole must meet the requirements of A2.7.2.6-A2.7.2.11.

A2.7.2.2 *Left and Right Position Lights*—Left and right position lights must consist of a red and a green light spaced laterally as far apart as practicable and installed on the airplane such that, with the airplane in the normal flying position, the red light is on the left side and the green light is on the right side.

A2.7.2.3 *Rear Position Light*—The rear position light must be a white light mounted as far aft as practicable on the tail or on each wing tip.

A2.7.2.4 *Light Covers and Color Filters*—Each light cover or color filter must be at least flame-resistant and may not change color or shape or lose any appreciable light transmission during normal use.

A2.7.2.5 —Position light system dihedral angles.

A2.7.2.6 *Position Light System Dihedral Angles*:

(1) Except as provided in (5) of this section, each position light must, as installed, show unbroken light within the dihedral angles described in this section.

(2) Dihedral angle L (left) is formed by two intersecting vertical planes, the first parallel to the longitudinal axis of the airplane, and the other at 110° to the left of the first, as viewed when looking forward along the longitudinal axis.

(3) Dihedral angle R (right) is formed by two intersecting vertical planes, the first parallel to the longitudinal axis of the airplane, and the other at 110° to the right of the first, as viewed when looking forward along the longitudinal axis.

(4) Dihedral angle A (aft) is formed by two intersecting vertical planes making angles of 70° to the right and to the left, respectively, to a vertical plane passing through the longitudinal axis, as viewed when looking aft along the longitudinal axis.

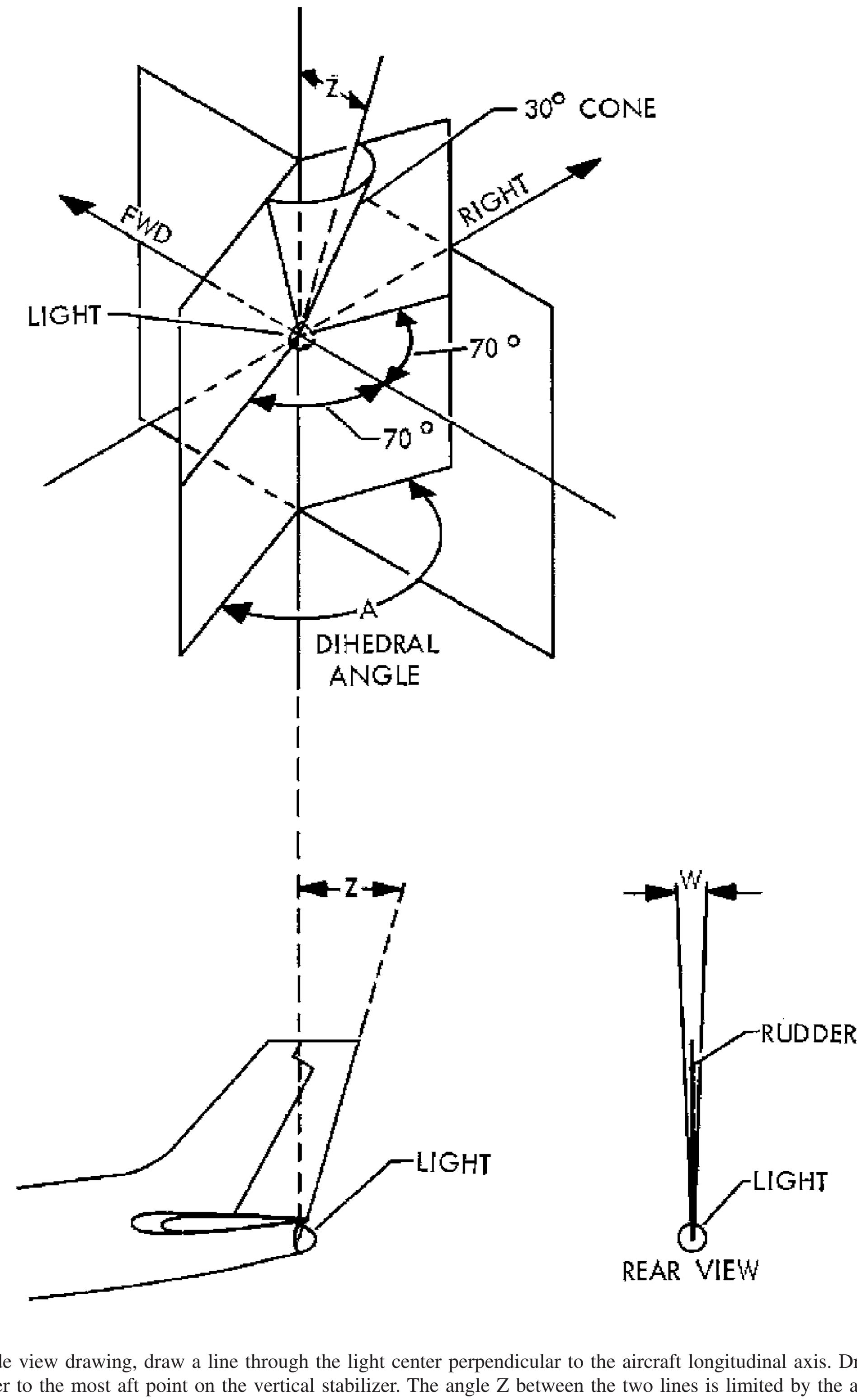
(5) If the rear position light, when mounted as far aft as practicable in accordance with Sec. 2.7.2(c), cannot show unbroken light within dihedral angle A (as defined in (4) of this section), a solid angle or angles of obstructed visibility totaling not more than 0.04 steradians is allowable within that dihedral angle, if such solid angle is within a cone whose apex is at the rear position light and whose elements make an angle of 30° with a vertical line passing through the rear position light (see Fig. A2.1).

A2.7.2.7 *Position Light Distribution and Intensities*:

(1) General—The intensities prescribed in this section must be provided by new equipment with each light cover and color filter in place. Intensities must be determined with the light source operating at a steady value equal to the average luminous output of the source at the normal operating voltage of the airplane. The light distribution and intensity of each position light must meet the requirements of the section on position lights.

(2) Position Lights—The light distribution and intensities of position lights must be expressed in terms of minimum intensities in any vertical plane, and maximum intensities in overlapping beams, with dihedral angles L, R, and A, and must meet the following requirements:

(*a*) *Intensities in the horizontal plane*—Each intensity in the horizontal plane (the plane containing the longitudinal axis of the airplane and perpendicular to the plane of symmetry of the airplane) must equal or exceed the values in A2.7.2.8.



NOTE 1—On the side view drawing, draw a line through the light center perpendicular to the aircraft longitudinal axis. Draw a second line upward through the light center to the most aft point on the vertical stabilizer. The angle Z between the two lines is limited by the airworthiness rules to 30°. Fig. A2.1 shows an example of angle Z.

NOTE 2—On the rear view drawing, draw angle W, which is formed by two lines drawn upward from the light center to the maximum right and left obstructions with angle Z. When a protrusion causes a very small zone of obstruction, it may be discounted, unless total obstructions are near the regulatory limit. When a rear view drawing is not available, a combination of other drawings or measurements on the actual aircraft can be used to determine angle W.

NOTE 3—Multiple angle Z degrees by angle W degrees to obtain the amount of obstruction in square degrees. The method is conservative, as obstructions as wide as angle W may not exist throughout angle Z. Convert the measurement to steradians by dividing the square degree value by 3284. The number 3284 is a conversion factor to obtain steradians from square degrees.

FIG. A2.1 Rear Position Light Obstructions

(*b*) *Intensities in any vertical plane*—Each intensity in any vertical plane (the plane perpendicular to the horizontal plane) must equal or exceed the appropriate value in A2.7.2.9, where I is the minimum intensity prescribed in A2.7.2.8 for the corresponding angles in the horizontal plane.

(*c*) *Intensities in overlaps between adjacent signals*—No intensity in any overlap between adjacent signals may exceed the values in A2.7.2.10, except that higher intensities in overlaps may be used with main beam intensities substantially greater than the minima specified in A2.7.2.8 and A2.7.2.9, if the overlap intensities in relation to the main beam intensities do not adversely affect signal clarity. When the peak intensity of the left and right position lights is more than 100 candles, the maximum overlap intensities between them may exceed the values in A2.7.2.10 if the overlap intensity in Area A is not more than 10 % of peak position light intensity and the overlap intensity in Area B is not more than 2.5 % of peak position light intensity.

(3) Rear position light installation—A single rear position light may be installed in a position displaced laterally from the plane of symmetry of an airplane if: (*1*) the axis of the maximum cone of illumination is parallel to the flight path in level flight; and (*2*) there is no obstruction aft of the light and between planes 70° to the right and left of the axis of maximum illumination.

A2.7.2.8 *Minimum Intensities in the Horizontal Plane of Position Lights*—Each position light intensity must equal or exceed the applicable values in the following table:

Dihedral Angle (Light Included)	Angle from Right or Left of Longitudinal Axis, Meausured from Dead Ahead	Intensity (Candles)
L and *R* (red and green)	0° to 10°	40
	10° to 20°	30
	20° to 110°	5
A (rear white)	110° to 180°	20

A2.7.2.9 *Minimum Intensities in any Vertical Plane of Position Lights*— Each position light intensity must equal or exceed the applicable values in the following table:

Angle above or below the horizontal plane	Intensity
0°	1.00 *I.*
0° to 5°	0.90 *I.*
5° to 10°	0.80 *I.*
10° to 15°	0.70 *I.*
15° to 20°	0.50 *I.*
20° to 30°	0.30 *I.*
30° to 40°	0.10 *I.*
40° to 90°	0.05 *I.*

A2.7.2.10 *Maximum Intensities in Overlapping Beams of Position Lights*—No position light intensity may exceed the applicable values in the following table, except as provided in A2.7.2.7 (2)(c):

Overlaps	Maximum Intensity Area A (candles)	Area B (candles)
Green in dihedral angle *L*	10	1
Red in dihedral angle *R*	10	1
Green in dihedral angle *A*	5	1
Red in dihedral angle *A*	5	1
Rear white in dihedral angle *L*	5	1
Rear white in dihedral angle *R*	5	1

where:

Area A = all directions in the adjacent dihedral angle that pass through the light source and intersect the common boundary plant at more than 10° but less than 20°, and

Area B = all directions in the adjacent dihedral angle that pass through the light source and intersect the common boundary plane at more than 20°.

A2.7.2.11 *Color Specifications*—Each position light color must have the applicable International Commission on Illumination chromaticity coordinates as follows:

(1) Aviation Red—*y* is not greater than 0.335; and *z* is not greater than 0.002.

(2) Aviation Green—*x* is not greater than 0.440 – 0.320 y; *x* is not greater than y – 0.170; and *y* is not less than 0.390 – 0.170 x.

(3) Aviation White—*x* is not less than 0.300 and not greater than 0.540; *y* is not less than ″x – 0.040″ or ″y_0 – 0.010″, whichever is the smaller; and *y* is not greater than ″x + 0.020″ nor ″0.636 – 0.400x″; where y_0 is the *y* coordinate of the Planckian radiator for the value of *x* considered.

A2.7.3 *Anticollision Light System*:

A2.7.3.1 *General*—The airplane must have an anti-collision light system that: (*1*) consists of one or more anti-collision lights located so that their light will not impair the flight crewmembers' vision or detract from the conspicuity of the position lights; and (*2*) meets the requirements of A2.7.3.2 through A2.7.3.6.

A2.7.3.2 *Field of Coverage*—The system must consist of enough lights to illuminate the vital areas around the airplane, considering the physical configuration and flight characteristics of the airplane. The field of coverage must extend in each direction within at least 75° above and 75° below the horizontal plane of the airplane, except that there may be solid angles of obstructed visibility totaling not more than 0.5 steradians.

A2.7.3.3 *Flashing Characteristics*—The arrangement of the system, that is, the number of light sources, beam width, speed of rotation, and other characteristics, must give an effective flash frequency of not less than 40, nor more than 100, cycles per minute. The effective flash frequency is the frequency at which the airplane's complete anti-collision light system is observed from a distance, and applies to each sector of light, including any overlaps that exist when the system consists of more than one light source. In overlaps, flash frequencies may exceed 100, but not 180, cycles per minute.

A2.7.3.4 *Color*—Each anti-collision light must be either aviation red or aviation white and must meet the applicable requirements of A2.7.2.11.

A2.7.3.5 *Light Intensity*—The minimum light intensities in any vertical plane, measured with the red filter (if used) and expressed in terms of "effective" intensities, must meet the requirements of A2.7.3.6. The following relation must be assumed:

$$I_e = \frac{\int_{t_1}^{t_2} I(t)\,dt}{0.2 + (t_2 - t_1)} \tag{A2.1}$$

where:

I_e = effective intensity (candles),
$I(t)$ = instantaneous intensity as a function of time, and
$t_2 - t_1$ = flash time interval.

Normally, the maximum value of effective intensity is obtained when t_2 and t_1 are chosen so that the effective intensity is equal to the instantaneous intensity at t_2 and t_1.

A2.7.3.6 *Minimum Effective Intensities for Anti-collision Lights*—Each anti-collision light effective intensity must equal or exceed the applicable values in the following table.

Angle above or below the horizontal plane	Effective intensity (candles)
0° to 5°	400
5° to 10°	240
10° to 20°	80
20° to 30°	40
30° to 75°	20

A2.7.4 *Taxi and Landing Lights*— Each taxi and landing light must be designed and installed so that:

A2.7.4.1 No dangerous glare is visible to the pilots,

A2.7.4.2 The pilot is not seriously affected by halation,

A2.7.4.3 It provides enough light for night operations, and

A2.7.4.4 It does not cause a fire hazard in any configuration.

A2.8 Avionics—Must be illuminated in accordance with A2.7.1.

A2.9 Electrical Requirements

A2.9.1 *Switches*—Each switch must be:

A2.9.1.1 Rated by the switch manufacturer to carry its circuit's current;

A2.9.1.2 For circuits containing incandescent lamps, have a minimum in-rush rating of 15 times the lamp's continuous current;

A2.9.1.3 Constructed with enough distance or insulating material between current carrying parts and the housing so that vibration in flight will not cause shorting;

A2.9.1.4 Accessible to the pilot;

A2.9.1.5 Labeled as to operation and the circuit controlled; and

A2.9.1.6 Illuminated in accordance with A2.7.1.

A2.9.2 *Circuit Protection Requirements*—Circuit overload protection (fuses or circuit breakers) must:

A2.9.2.1 Be installed on each circuit containing wiring, equipment, or other components rated for less than the maximum output of the battery and alternator or generator;

A2.9.2.2 Be appropriately rated for each component installed on the protected circuit;

A2.9.2.3 Be accessible to and in clear view of the pilot;

A2.9.2.4 Open before the conductor emits smoke; and

A2.9.2.5 Automatic re-set circuit breakers may not be used.

A2.9.3 *Electrical Energy Requirements*—The total continuous electrical load may not exceed 80 % of the total rated generator or alternator output capacity.

A2.9.4 *Conductor Requirements*—Any wire or other material intended to conduct electricity must be:

A2.9.4.1 Rated to carry its circuits current;

A2.9.4.2 For wiring rated to 150°C, 600 V minimum;

A2.9.4.3 Constructed with enough distance or insulating material between current carrying conductors so that vibration in flight will not cause shorting; and

A2.9.4.4 Where used, insulating material must have, at a minimum, the equivalent or better properties of either PTFE -polytetrafluoroethylene (commonly known by the trade name, TEFLON) or ETFE-(Frequently referred to by the trade name, TEFZEL) a copolymer of PTFE and of polyethylene including:

(1) Temperature,
(2) Abrasion resistance,
(3) Cut-through resistance,
(4) Chemical resistance,
(5) Flammability,
(6) Smoke generation,
(7) Flexibility,
(8) Creep (at temperature), and
(9) Arc propagation resistance.

A2.10 Operating Instructions (AOI)

A2.10.1 Electrical system description must be included for night.

A2.11 Learning Documents

A2.11.1 *FAA AC 20-30B* —Aircraft position light and anti-collision light instillations.

A2.11.2 *A2.10.2 FAA AC 65-15A Chapter 11*—Aircraft Electrical Systems.

APPENDIXES

(Nonmandatory Information)

X1. SIMPLIFIED DESIGN LOAD CRITERIA FOR LIGHT SPORT AIRPLANES

X1.1 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 = $1.59\sqrt{n_1 W/S}$ kts
$V_{A\ min}$	=	minimum design maneuvering speed = $2.17\sqrt{n_1 W/S}$ kts but need not exceed V_C used in design
$V_{C\ min}$	=	minimum design cruising speed = $2.46\sqrt{n_1 W/S}$ kts but need not exceed $0.9\ V_H$
$V_{D\ min}$	=	minimum design dive speed = $3.47\sqrt{n_1 W/S}$ 7 kts but need not exceed $1.4\sqrt{V_{C\ min}\, n_1/3.8}$ (see X1.2.5.2)

X1.2 Flight Loads

X1.2.1 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.

X1.2.2 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.

X1.2.3 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.

X1.2.4 Each specified wing and tail loading is independent of the center of gravity range. The applicant, however, must select a CG range, and the basic fuselage structure must be investigated for the most adverse dead weight loading conditions for the CG range selected.

X1.2.5 The following loads and loading conditions are the minimums for which strength must be provided in the structure:

X1.2.5.1 *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 X1.3.2 and X1.3.3) for the positive flight conditions and magnitude equal to the airplane normal loads for the negative conditions. Each chord-wise and normal component of this wing load must be considered.

TABLE X1.1 Minimum Design Limit Flight Load Factors

Flaps Up	n_1= 4.0
	n_2= -0.5n_1
	n_3 from Fig. X1.2
	n_4 from Fig. X1.3
Flaps Down	n_f = 0.5n_1
	n_f = 0[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.

X1.2.5.2 *Minimum Design Airspeeds*—The minimum design airspeeds may be chosen by the applicant except that they may not be less than the minimum speeds found in X1.1. In addition, $V_{C\ min}$ need not exceed values of 0.9 V_H actually obtained at sea level for the lowest design weight category for which certification is desired. In computing these minimum design airspeeds, n_1 may not be less than 4.0.

X1.2.5.3 *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.

X1.3 Flight Conditions

X1.3.1 *General*—Each design condition in X1.3.2-X1.3.4 must be used to assure sufficient strength for each condition of speed and load factor on or within the boundary of a flight envelope 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.

X1.3.2 *Symmetrical Flight Conditions*—The airplane must be designed for symmetrical flight conditions as follows:

X1.3.2.1 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 the design speed for these conditions must be at least equal to the value of $V_{D\ min}$ from X1.1.

(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} = ±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 $n_3 W/S$ or $n_4 W/S$ of Appendix X1, is greater than $n_1 W/S$ and $n_2 W/S$, respectively.

X1.3.2.2 If the 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 X1.1.

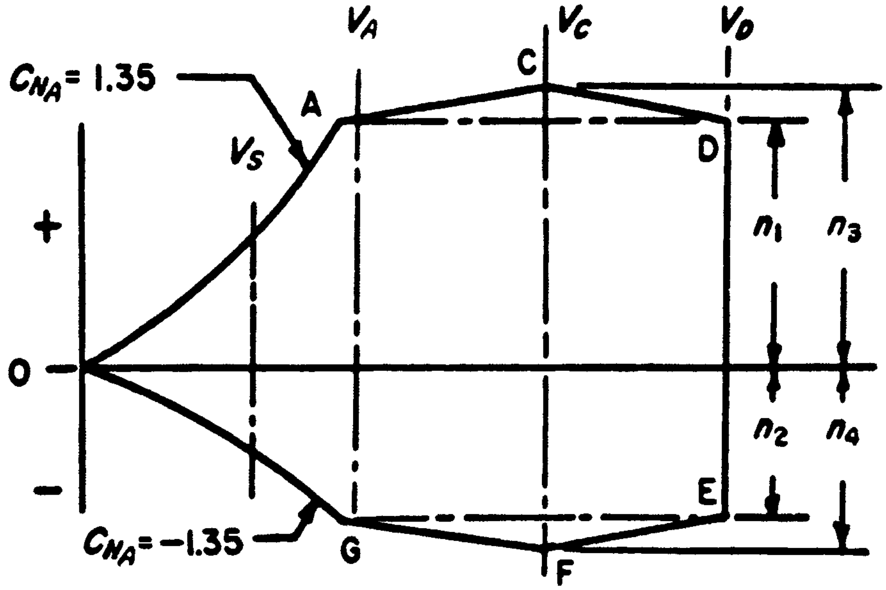


FIG. X1.1 Generalized Flight Envelope

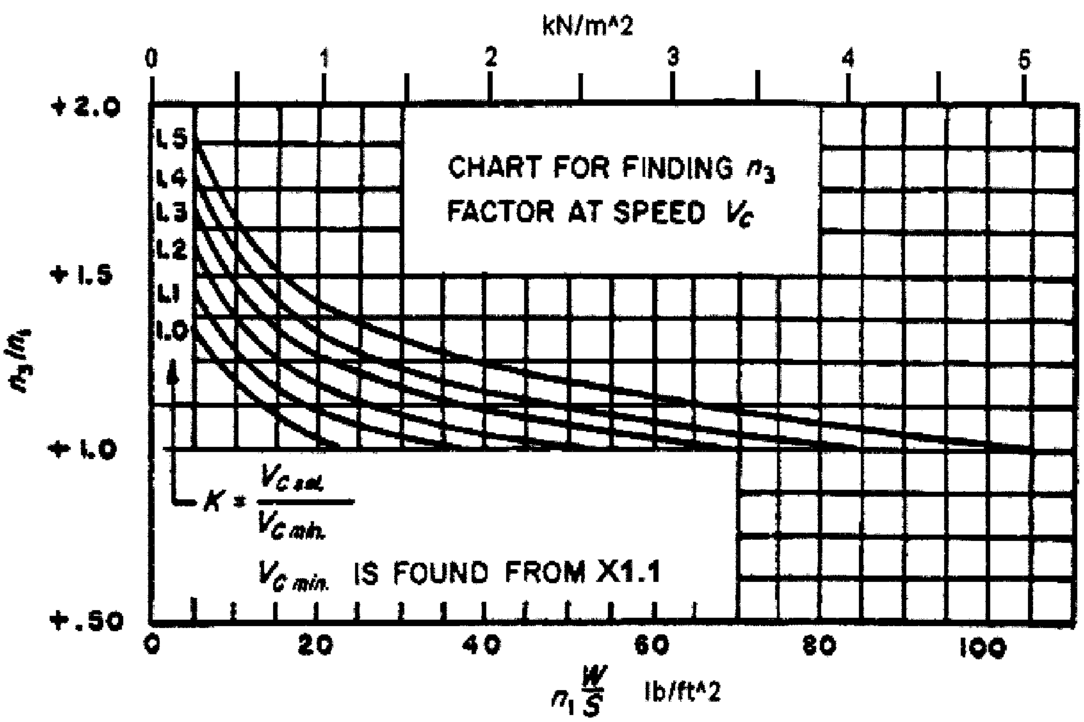


FIG. X1.2 Chart for Finding n_3 Factor at Speed V_c

X1.3.3 *Unsymmetrical Flight Conditions*—Each affected structure must be designed for unsymmetrical loadings as follows:

X1.3.3.1 The aft fuselage-to-wing attachment must be designed for the critical vertical surface load determined in accordance with X1.4.3.

X1.3.3.2 The wing and wing carry-through structures must be designed for 100 % of Condition “A” loading on one side of the airplanes plane of symmetry and 70 % on the opposite side.

X1.3.3.3 The wing and wing carry-through structures must be designed for the loads resulting from a combination 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 modified over the aileron portion of the span, must be computed as follows:

(1) $C_m = C_m + 0.01\ \delta_u$ (up aileron side) wing basic airfoil

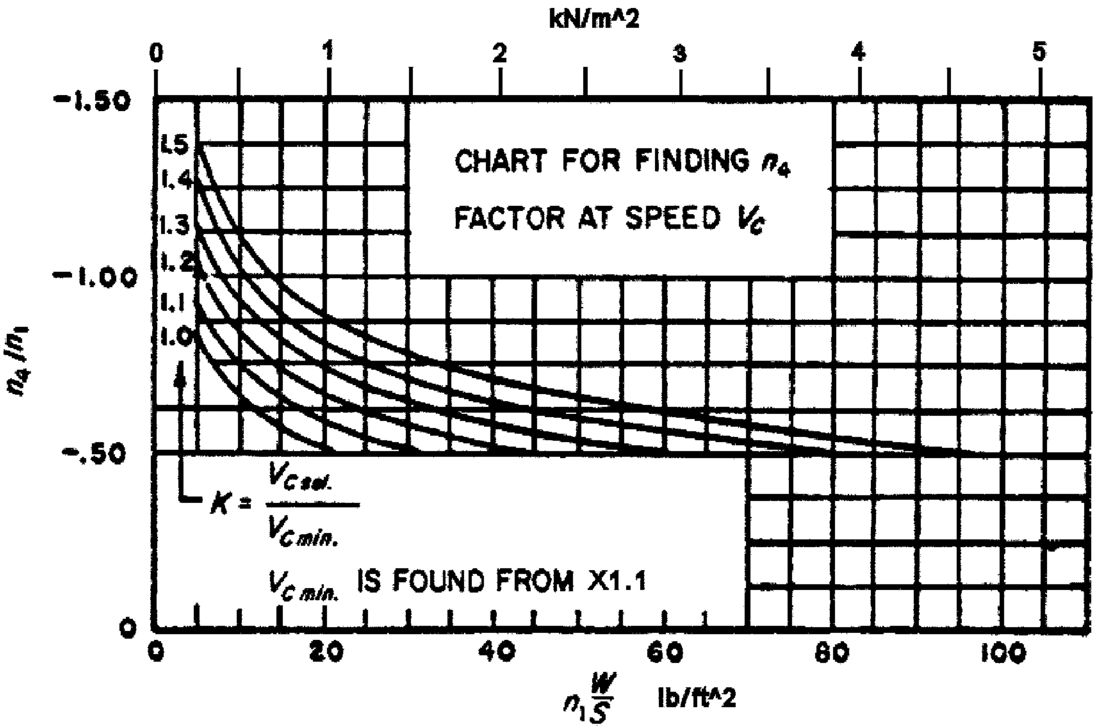


FIG. X1.3 Chart for Finding n_4 Factor at Speed V_c

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

X1.3.3.4 $\Delta_{critical}$, which is the sum of $\delta_u + \delta_d$ must be computed as follows:

(1) Compute Δ_a and Δ_b from the formulas:

$$\Delta_a = V_A/V_C \times \Delta_p, \text{ and} \tag{X1.1}$$

$$\Delta_b = 0.5 \times V_A/V_D \times \Delta_p$$

where:

Δ_p = maximum total deflection (sum of both aileron deflections) at V_A with V_A, V_C, and V_D described in X1.2.5.2

(2) Compute K from the formula:

$$K = \frac{(C_m - 0.01\delta_b)V_D^2}{(C_m - 0.01\delta_a)V_C^2} \tag{X1.2}$$

where:

δ_a = down aileron deflection corresponding to Δ_a, and
δ_b = down aileron deflection corresponding to Δ_b as computed in X1.3.3.4(*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 that 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 that must be used in computing the wing torsion loads over the aileron span.

X1.3.4 *Supplementary Conditions; Rear Lift Truss; Engine Torque; Side Load on Engine Mount*—Each of the following supplementary conditions must be investigated:

X1.3.4.1 In designing the rear lift truss, the following special condition may be investigated instead of Condition "G" of Fig. X1.1. The rear lift truss must be designed for conditions of reversed airflow at a design speed of V = 39 kts. Either aerodynamic data for a particular wing section used, or a value of C_L equaling −0.8 with a chord-wise distribution that is triangular between a peak at the trailing edge and zero at the leading edge, must be used.

X1.3.4.2 Each engine mount and its supporting structures must be designed for the maximum limit torque corresponding to maximum expected takeoff power and propeller speed acting simultaneously with the limit loads resulting from the maximum positive maneuvering flight load factory n_1. The limit torque must be obtained from 5.2.9.

X1.3.4.3 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.

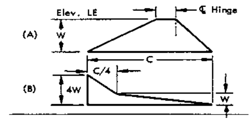
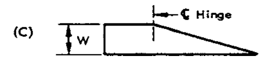
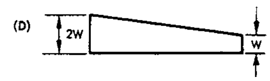
X1.4 Control Surface Loads

X1.4.1 *General*—Each control surface load must be determined using the criteria of X1.4.2 and must lie within the simplified loadings of X1.4.3.

X1.4.2 *Limit Pilot Forces*—In each control surface loading condition described in X1.4.3, the air loads on the movable surfaces and the corresponding deflections need not exceed those which could be obtained in flight by using the maximum limit pilot forces specified in 5.3.3.

X1.4.3 *Surface Loading Conditions*—Each surface loading condition must be investigated as follows: Simplified limit surface loadings and distributions for the horizontal tail, vertical tail, aileron, wing flaps, and trim tabs are specified in Table X1.2, and Figs. X1.4 and X1.5. If more than one

TABLE X1.2 Average Limit Control Surface Loading

Surface	Direction of Loading	Magnitude of Loading	Chord-wise Distribution
I. Horizontal tail	a) up and down	Fig. X1.4 Curve (2)	
	b) unsymmetrical loading (up and down)	100 % w on one side airplane 65 % w on other side airplane	
II. Vertical tail	a) right and left	Fig. X1.4 Curve (1)	same as (A)
	b) right and left	Fig. X1.4 Curve (1)	same as (B)
III. Aileron	a) up and down	Fig. X1.5 Curve (5)	
IV. Wing flap	a) up	Fig. X1.5 Curve (4)	
	b) down	0.25 × Up Load	
V. Trim tab	a) up and down	Fig. X1.5 Curve (3)	same as (D)

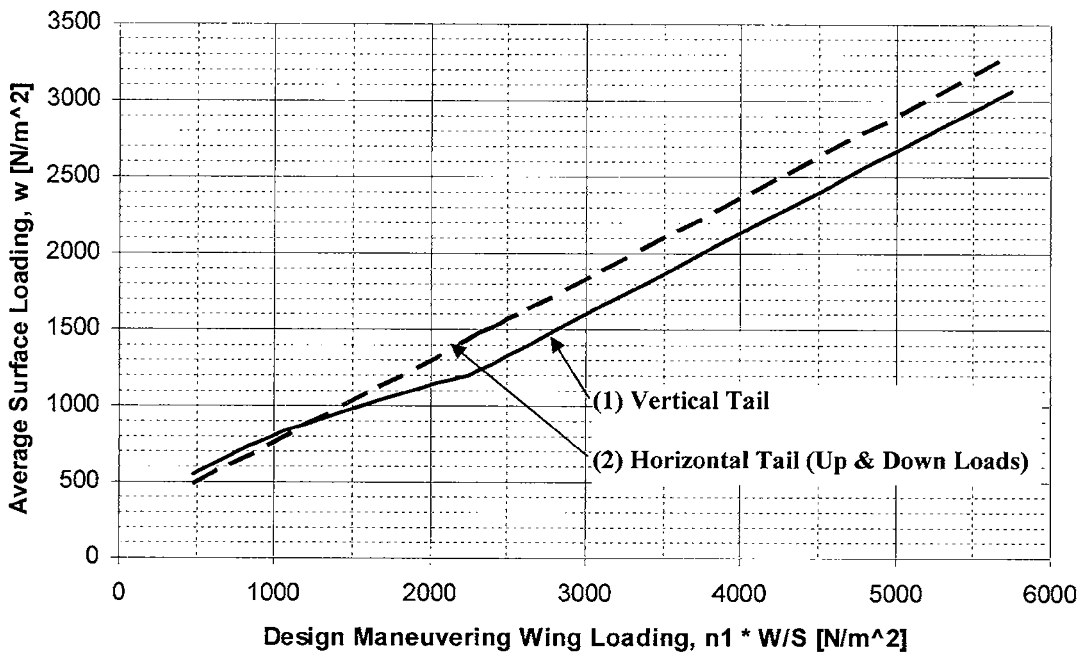


FIG. X1.4 Average Limit Control Surface Loading

distribution is given, each distribution must be investigated.

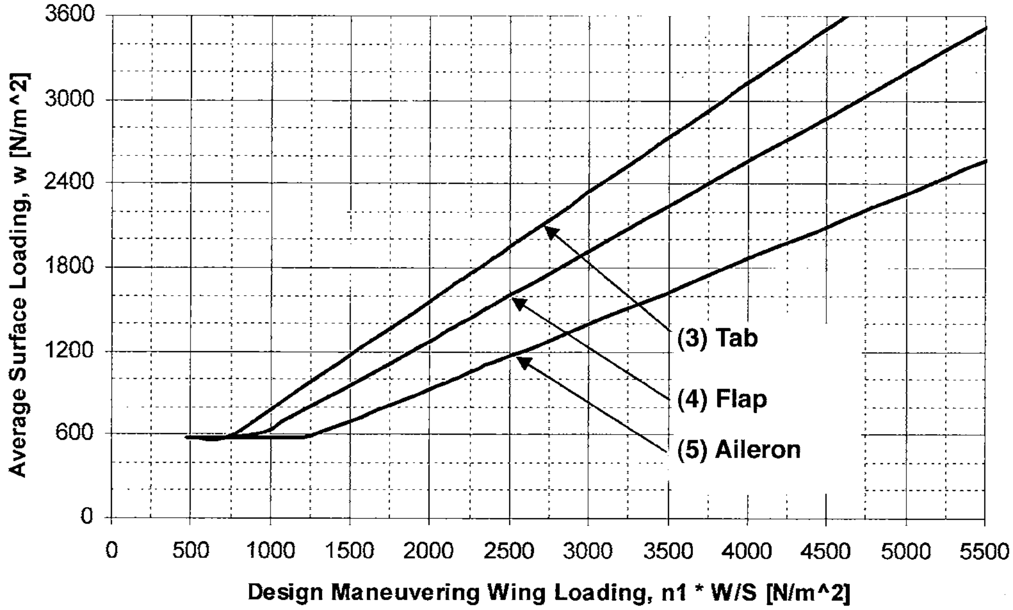


FIG. X1.5 Average Limit Control Surface Loading

X1.5 Control System Loads

X1.5.1 *Primary Flight Controls and Systems*—Each primary flight control and system must be designed as follows:

X1.5.1.1 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 X1.4. In addition, the system limit loads need not exceed those that could be produced by the pilot and automatic devices operating the controls, and the design must provide a rugged system for service use, including jamming, ground gusts, taxiing downwind, control inertia, and friction.

X1.5.2 Dual controls must meet 5.3.3 and 5.3.4.

X1.5.3 Ground gust condition must meet 5.3.7.

X2. ACCEPTABLE METHODS OF WING AND FUSELAGE LOAD CALCULATIONS

NOTE X2.1—These may not include all of the loads that are imposed on the wing or fuselage.

X2.1 Symmetrical Wing Loads

X2.1.1 As a minimum, the following four conditions need investigation:

Point A	normal load up	$= 4 \times W$
	tangential forward	$= W$
Point D	normal load up	$= 4 \times W$
	tangential rearward	$= W/5$
Point G	normal down	$= 2 \times W$
	tangential forward	$= 2 \times W/5 = 0.4 \times W$
	with flaps extended:	
Point F	normal up	$= 2 \times W$
	tangential forward	$= W$

X2.2 Instead of the above simplification, a more rational analysis using the following lift and drag components in Fig. X2.1 may be used:

X2.3 Unsymmetrical Wing Loads

X2.3.1 *Shear, Wing Carry Through*—Assume 100 % of Point A on one wing and apply 75 % of Point A on the other wing.

X2.3.2 *Torsion, Wing*—Assume 75 % of Point A or D on each wing and add the torsional loads because of the aileron deflection as shown in Fig. X2.2):

X2.3.3 *Torsion, Wing*—Assume 75 % of Point D on each wing and add the torsion loads as a result of ⅓ of the aileron deflection.

X2.3.4 If the landing gear is attached to the wing, the wing structure shall be justified for the ground loads as well.

X2.4 *Rear Fuselage Loads*—The rear fuselage shall be substantiated for:

X2.4.1 The symmetrical horizontal tail load of 5.4.2 and 5.4.3,

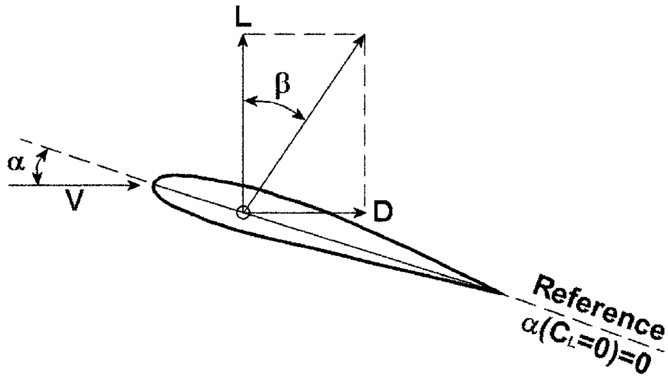
X2.4.2 The vertical tail loads of 5.5.1 and 5.5.2, and

X2.4.3 The tail wheel loads if applicable.

X2.5 *Forward Fuselage Loads*—The forward fuselage shall be substantiated for each of the following conditions:

X2.5.1 Inertia forces of $n = 4$ and $n = -2$ (or for n_3 and n_4 if they are larger than 4 and −2) (see also 5.8 if n_j is larger than 3.33), and

X2.5.2 Engine limit torque ($N \times m$) equal to the values specified in 5.2.9.



$$L = Lift = C_L \times S \times q$$

$$D = Drag = C_D \times S \times q$$

$$\text{with } C_L = \frac{n \times W}{S \times q}$$

$$C_D = .01 + \frac{C_L^{\,2}}{3.14 \times AR}$$

$$\beta = (\tan)^{-1} \frac{C_D}{C_L}$$

$$\alpha = \frac{C_L}{\frac{dC_L}{d\alpha}}$$

$$\frac{dC_L}{d\alpha}\left[\deg^{-1}\right] = .1 \times \frac{AR}{AR+2}$$

NOTE 1—Both components (normal and tangential) must be considered simultaneously.
NOTE 2—The aerodynamic loads shall be considered to be located at the aerodynamic center.
NOTE 3—The wing normal and tangential loads are balanced by the inertia loads at the corresponding load factors.
NOTE 4—If wing flaps are installed, the resulting loads shall also be investigated at Point F (symmetrical load condition).

FIG. X2.1 Normal and Tangential Loads

$$C_{m\frac{c}{4}} = C_{m\frac{c}{4}}(\text{basic airfoil}) + .01 \times \delta\, up$$

$$C_{m\frac{c}{4}} = C_{m\frac{c}{4}}(\text{basic airfoil}) - .01 \times \delta\, dn$$

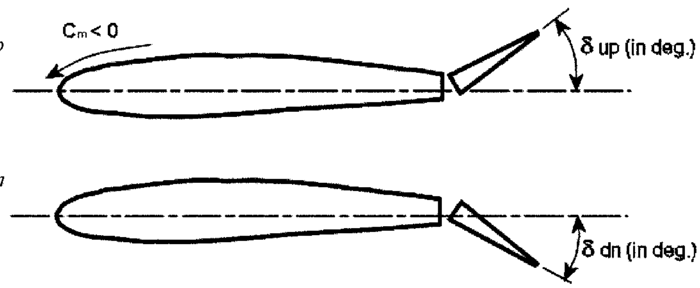


FIG. X2.2 Unsymmetrical Wing Loads

X3. ACCEPTABLE MEANS OF GUST LOAD FACTOR CALCULATIONS

X3.1 In the absence of a more rational analysis, the gust load factors may be computed as follows:

$$n = 1 + \frac{\frac{1}{2} \cdot \rho \cdot V \cdot K_g \cdot a \cdot U_{de}}{\left(\frac{W}{S}\right)} \tag{X3.1}$$

where:

K_g = $0.88\mu_g / 5.3 + \mu_g$ = gust alleviation factor,
μ_g = $2(W/S) / \rho \cdot C \cdot a \cdot g$ = airplane mass ratio,
U_{de} = derived gust velocities referred to in 5.2.3.3, m/s,
ρ = density of air, kg/m^3,
W/S = wing loading, N/m^2,
C = mean geometric chord of wing, m,
g = acceleration of gravity, m/s^2,
V = airplane equivalent airspeed (or CAS for LSA), m/s, and
a = slope of the airplane normal force coefficient curve, C_{NA} per radian.

X3.2 The wing lift curve slope, C_L per radian, may be used when the gust load applied to the wing only and the horizontal tail gust loads are treated as separate condition. The value of n calculated from the preceding expression need not exceed:

$$n = 1.25 \cdot \left(\frac{V}{V_{S1}}\right)^2 \tag{X3.2}$$

X4. ACCEPTABLE MEANS FOR CALCULATING GUST LOADS ON STABILIZING SURFACES

X4.1 In the absence of a more rational analysis, the horizontal stabilizing surfaces gust loads may be computed as follows:

$$\Delta L_{HT} = \frac{K_g \cdot U_{de} \cdot V \cdot a_{HT} \cdot S_{HT}}{16.3}\left(1 - \frac{d\epsilon}{d\alpha}\right) \tag{X4.1}$$

where:

ΔL_{HT} = incremental horizontal surface load, daN,
K_g = gust alleviation factor; same value used in 5.2.6,

U_{de} = derived gust velocity, m/s,
V = airplane airspeed (CAS for LSA), m/s,
a_{HT} = slope of horizontal surface lift curve, per radian,
S_{HT} = area of horizontal surface, m^2, and
$(1-{}^{d\epsilon}/_{d\alpha})$ = downwash factor.

X4.2 In the absence of a more rational analysis, the vertical stabilizing surfaces gust loads may be computed as follows:

$$\Delta L_{VT} = \frac{K_{gt} \cdot U_{de} \cdot V \cdot a_{VT} \cdot S_{VT}}{16.3} \quad \text{(X4.2)}$$

$$K_{gt} = \frac{.88 \cdot \mu_{gt}}{5.3 + \mu_{gt}} \quad \text{(X4.3)}$$

$$\mu_{gt} = \frac{2 \cdot M}{\rho \cdot C_{vt} \cdot g \cdot a_{vt} \cdot S_{vt}} \left(\frac{K}{l_{vt}} \right)^2 \quad \text{(X4.4)}$$

where:
L_{VT} = incremental vertical surface load, daN,
K_{gt} = gust alleviation factor,
μ_{gt} = lateral mass ratio,
U_{de} = derived gust velocity, m/s,
M = airplane mass, kg,
ρ = density of air, kg/m^3,
V = airplane equivalent airspeed (CAS may be used for LSA), m/s,
a_{VT} = slope of vertical surface lift curve, per radian,
S_{VT} = area of vertical surface, m^2,
C_{vt} = mean geometric chord of vertical surface, m,
K = Radius of gyration in yaw, m,
l_{vt} = distance from airplane c.g. to lift center of vertical surface, m, and
g = acceleration due to gravity, m/s^2.

X5. ACCEPTABLE MEANS FOR CALCULATION OF WATER LOADS

NOTE X5.1—In the absence of a more rational analysis, the water loads may be calculated as follows:

X5.1 Water Load Conditions

X5.1.1 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.

X5.1.2 In the absence of a more rational analysis of the water loads, X5.2 through X5.9 apply.

X5.2 Design Weights and Center of Gravity Positions

X5.2.1 *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 X5.6, the design water takeoff weight (the maximum weight for water taxi and takeoff run) must be used.

X5.2.2 *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.

X5.3 Application of Loads

X5.3.1 Unless otherwise prescribed, the seaplane as a whole is assumed to be subjected to the loads corresponding to the load factors specified in X5.4.

X5.3.2 In applying the loads resulting from the load factors prescribed in X5.4, 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 X5.7.3.

X5.3.3 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.

X5.3.4 Except in the takeoff condition of X5.6, the aerodynamic lift on the seaplane during the impact is assumed to be two thirds of the weight of the seaplane.

X5.4 Hull and Main Float Load Factors

X5.4.1 Water reaction load factors n_w must be computed in the following manner:

X5.4.1.1 For the step landing case:

$$n_w = \frac{C_1 V_{S0}{}^2}{(\tan^{\frac{2}{3}} \beta) \left(\frac{W}{4.448} \right)^{\frac{1}{3}}} \quad \text{(X5.1)}$$

X5.4.1.2 For the bow and stern landing cases:

$$n_w = \frac{C_1 V_{S0}{}^2}{(\tan^{\frac{2}{3}} \beta) \left(\frac{W}{4.448} \right)^{\frac{1}{3}}} \times \frac{K_1}{(1 + r_x{}^2)^{\frac{2}{3}}} \quad \text{(X5.2)}$$

X5.4.2 The following values are used:

n_w = water reaction load factor (that is, the water reaction divided by seaplane weight),
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),
Vs_0 = seaplane stalling speed in knots with flaps extended in the appropriate landing position and with no slipstream effect,
β = angle of dead rise at the longitudinal station at which the load factor is being determined in accordance with Fig. X1.1,
W = seaplane design landing weight in Newtons,
K_1 = empirical hull station weighing factor, in accordance with Fig. X1.2, and
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.

X5.4.3 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. X1.2. This reduction applies only to the design of the carry-through and seaplane structure.

X5.5 Hull and Main Float Landing Conditions

X5.5.1 *Symmetrical Step, Bow, and Stern Landing*—For symmetrical step, bow, and stern landings, the limit water reaction load factors are those computed in X5.4.

X5.5.1.1 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;

X5.5.1.2 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

X5.5.1.3 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.

X5.5.2 *Unsymmetrical Landing for Hull and Single Float Seaplanes*—Unsymmetrical step, bow, and stern landing conditions must be investigated.

X5.5.2.1 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

X5.5.2.2 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.

X5.5.3 *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 in X5.4. 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.

X5.6 Hull and Main Float Takeoff Condition

X5.6.1 For the wing and its attachment to the hull or main float:

X5.6.1.1 The aerodynamic wing lift is assumed to be zero; and

X5.6.1.2 A downward inertia load, corresponding to a load factor computed from the following formula, must be applied:

$$n = \frac{C_{TO} V_{S1}{}^2}{(\tan^{\frac{2}{3}} \beta)\left(\frac{W}{4.448}\right)^{\frac{1}{3}}} \tag{X5.3}$$

where:

n = inertia load factor,
C_{TO} = empirical seaplane operations factor equal to 0.004,
VS_1 = 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 Newtons.

X5.7 Hull and Main Float Bottom Pressures

X5.7.1 *General*—The hull and main float structure, including frames and bulkheads, stringers, and bottom plating, must be designed under this section.

X5.7.2 *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:

X5.7.2.1 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. X5.3. The pressure at the keel (kPa) is computed as follows:

$$P_k = \frac{C_2 K_2 Vs_1^2}{\tan \beta_k} \times 6.895 \tag{X5.4}$$

where:

P_k = pressure at the keel, kPa,
C_2 = 0.00213,
K_2 = hull station weighing factor, in accordance with Fig. X5.2,
Vs_1 = 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. X5.1.

X5.7.2.2 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. X5.3. The pressure distribution is the same at that prescribed in X5.7.2.1 for an unflared bottom except that the pressure at the chine is computed as follows:

$$P_{ch} = \frac{C_3 K_2 Vs_1^2}{\tan \beta} \times 6.895 \tag{X5.5}$$

where:

P_{ch} = pressure at the chine, kPa,
C_3 = 0.0016,
K_2 = hull station weighing factor, in accordance with Fig. X5.2,
Vs_1 = 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.

NOTE X5.2—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.

X5.7.3 *Distributed Pressures*—For the design of the frames, keel, and chine structure, the following pressure distributions apply:

X5.7.3.1 Symmetrical pressures as computed as follows:

$$P = \frac{C_4 K_2 Vs_0^2}{\tan \beta} \times 6.895 \tag{X5.6}$$

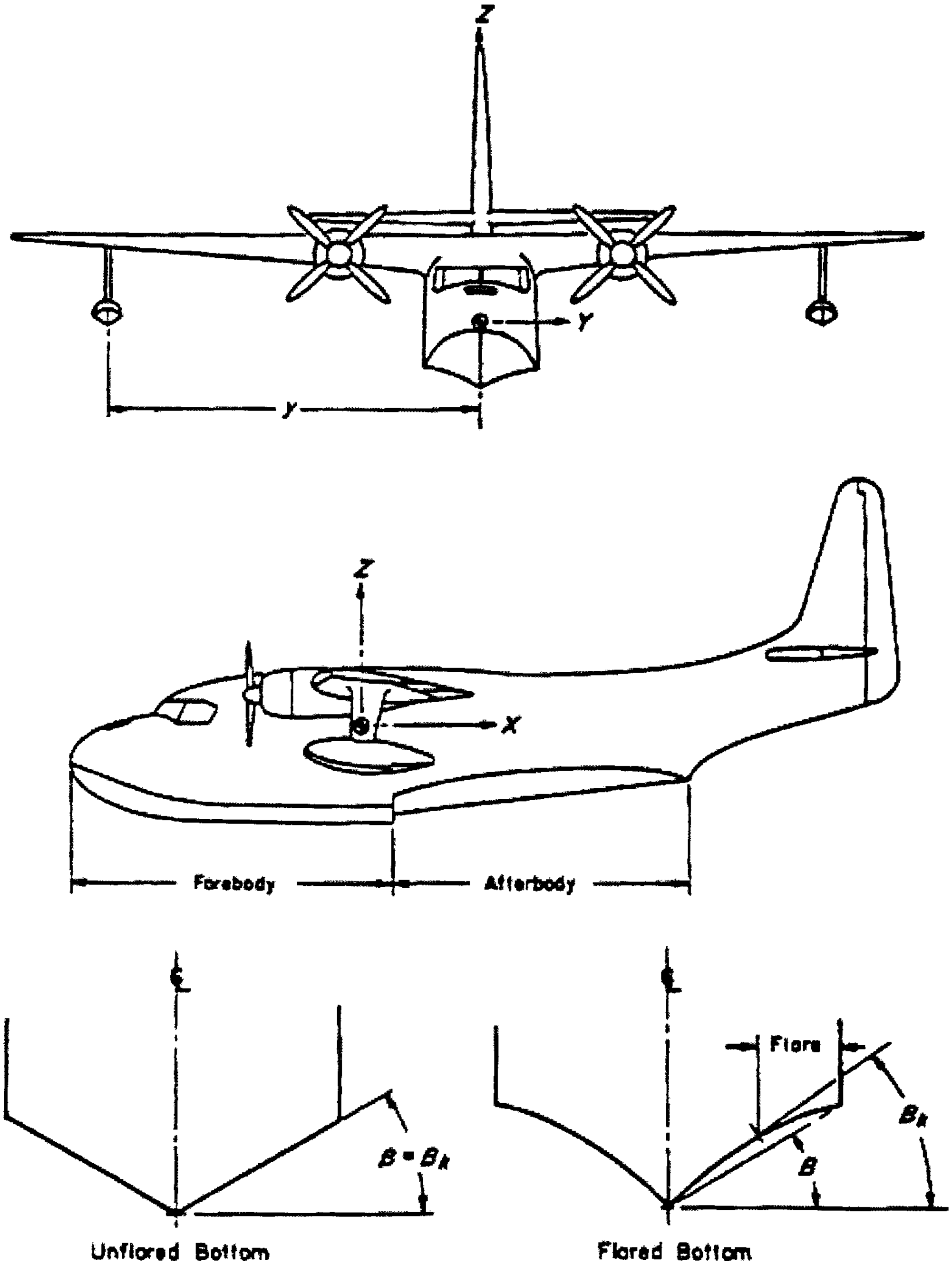


FIG. X5.1 Pictorial Definition of Angles, Dimensions, and Directions on a Seaplane

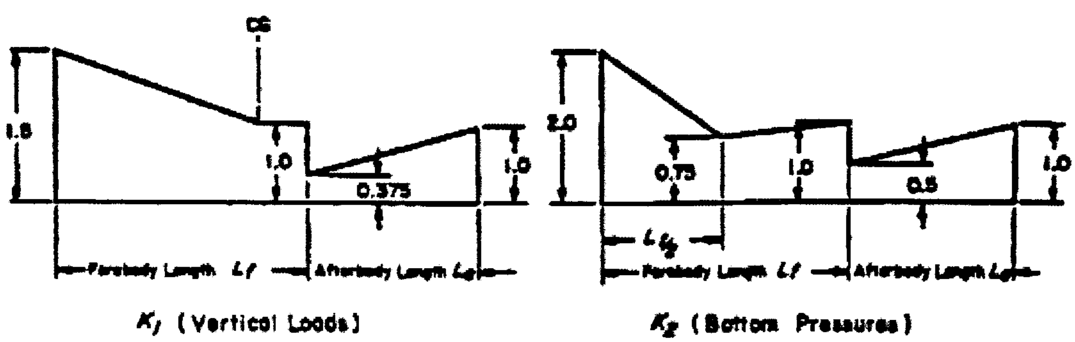


FIG. X5.2 Hull Station Weighing Factor

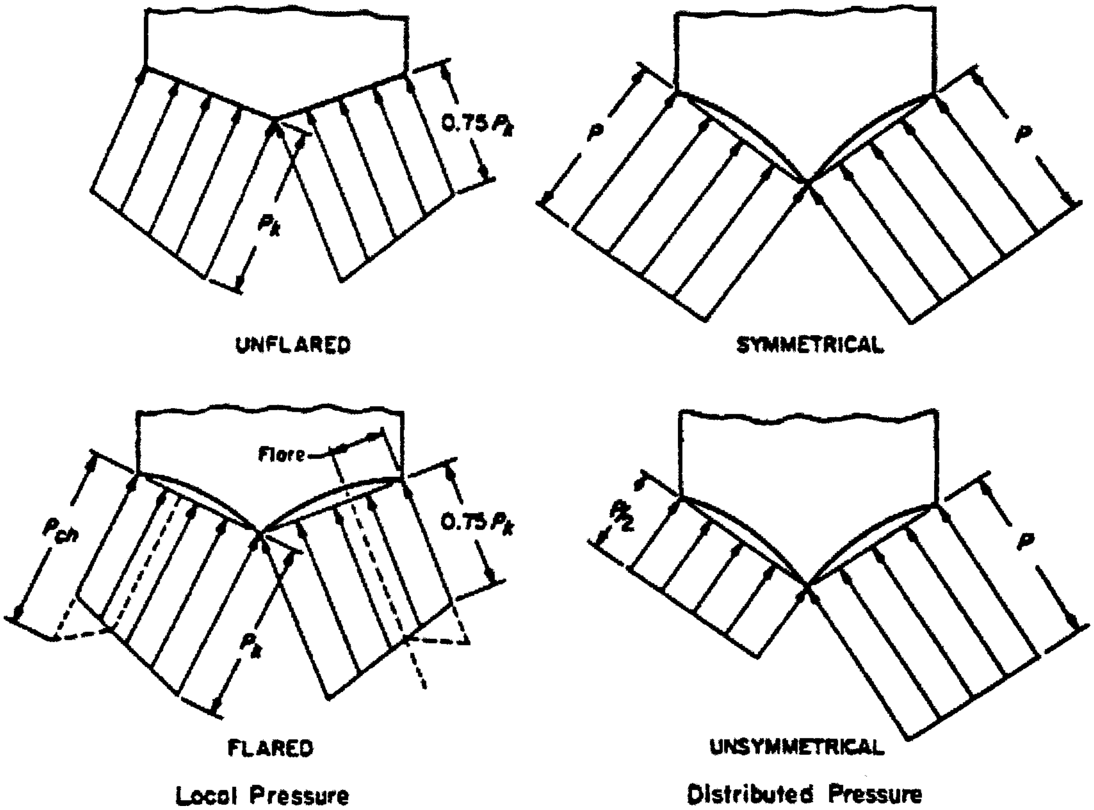


FIG. X5.3 Transverse Pressure Distributions

where:

P = pressure, kPa,
C_4 = 0.078 C_1(with C_1 computed in X5.4),
K_2 = hull station weighing factor, determined in accordance with Fig. X5.2,
Vs_0 = 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.

X5.7.3.2 The unsymmetrical pressure distribution consists of the pressures prescribed in X5.7.3.1 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. X5.3.

X5.7.3.3 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.

X5.8 Auxiliary Float Loads

X5.8.1 *General*—Auxiliary floats and their attachments and supporting structures must be designed for the conditions prescribed in this section. In the cases specified in X5.8.2 through X5.8.5, 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 X5.8.7.

X5.8.2 *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:

$$L = 4.448 \times \frac{C_5 V_{S0}{}^2 \left(\frac{W}{4.448}\right)^{\frac{2}{3}}}{(\tan^{\frac{2}{3}}\beta_S)(1 + r_y{}^2)^{\frac{2}{3}}} \quad \text{(X5.7)}$$

where:

L = limit load, N,
C_5 = 0.0053,
Vs_0 = seaplane stalling speed (knots) with landing flaps extended in the appropriate position and with no slipstream effect,
W = seaplane design landing weight, N,
β_S = angle of dead rise at a station three-fourths 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.

X5.8.3 *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 X5.8.2.

X5.8.4 *Unsymmetrical Step Loading*—The resultant water load consists of a component equal to 0.75 times the load specified in X5.8.1 and a side component equal to 0.25 tan β

times the load specified in X5.8.2. 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.

X5.8.5 *Unsymmetrical Bow Loading*—The resultant water load consists of a component equal to 0.75 times the load specified in X5.8.2 and a side component equal to 0.25 tan β times the load specified in X5.8.3. The side load must be applied perpendicularly to the plane of symmetry at a point midway between the keel and the chine.

X5.8.6 *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 (N) are as follows:

$$vertical = \rho g V \tag{X5.8}$$

$$aft = \frac{C_x \rho V^{\frac{2}{3}} (KV_{S0})^2}{2}$$

$$side = \frac{C_y \rho V^{\frac{2}{3}} (KV_{S0})^2}{2}$$

where:

ρ = mass density of water, kg/m^3,
V = volume of float, m^3,
C_x = coefficient of drag force, equal to 0.01236,
C_y = coefficient of side force, equal to 0.00985,
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^{Vs_0} in normal operations,
Vs_0 = seaplane stalling speed (knots) with landing flaps extended in the appropriate position and with no slipstream effect, and
g = acceleration due to gravity, m/s^2.

X5.8.7 *Float Bottom Pressures*—The float bottom pressures must be established in X5.7, 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 X5.8.2.

X5.9 *Seawing Loads*—Seawing design loads must be based on applicable test data.

X6. IMPERIAL AND METRIC UNITS

X6.1 : Only those units relevant to this specification are listed as follows, with a conversion accuracy adequate for the intended use.

Length	1 ft = 12 in. = 0.305 m
	1 in. = 2.54 cm
	1 m = 100 cm = 1000 mm = 39.37 in. = 3.28 ft
Surface	1 ft^2 = 0.093 m^2
	1 m^2 = 10.76 ft^2
Volume	1 U.S. gal = 3.78 L
	1 L = 0.264 U.S. gal
	(1 British gal = 1.2 U.S. gal = 4.5 L)
Weight	1 lb = 0.454 kg
	1 kg = 2.205 lb
Pressure	1 PSF = 4.88 kg/m^2
	1 kg/m^2 = 0.205 PSF
	1 psi = 2.3-ft water column = 0.000 703 kg/m^2
	1 ksi = 1000 psi = 0.703 kg/m^2
	1 kg/mm^2 = 1.43 ksi = 1430 psi
Dynamic pressure in standard atmosphere, at sea level	$q = V^2/391$ in lb/ft^2 when V in mph
	$q = (V/14.4)^2$ in kg/m^2 when V in km/h
Speeds	1 mph = 1.61 km/h
	1 knot = 1.15 mph = 1.85 km/h
	1 km/h = 0.62 mph = 0.54 knots
	1 fpm = 0.005 08 m/s (rate of climb)
	1 m/s = 197 FPM
Earth acceleration	g = 32.2 ft/s^2 = 9.81 m/s^2
Fuel density	6 lb/U.S. gal
	0.72 kg/L