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Flight Maneuver and Gust Conditions

FLIGHT MANEUVER AND GUST CONDITIONS

§ 25.331 Symmetric maneuvering conditions.

(a) *Procedure.* For the analysis of the maneuvering flight conditions specified in paragraphs (b) and (c) of this section, the following provisions apply:

(1) Where sudden displacement of a control is specified, the assumed rate of control surface displacement may not be less than the rate that could be applied by the pilot through the control system.

(2) In determining elevator angles and chordwise load distribution in the maneuvering conditions of paragraphs (b) and (c) of this section, the effect of corresponding pitching velocities must be taken into account. The in-trim and out-of-trim flight conditions specified in § 25.255 must be considered.

(b) *Maneuvering balanced conditions.* Assuming the airplane to be in equilibrium with zero pitching acceleration, the maneuvering conditions A through I on the maneuvering envelope in § 25.333(b) must be investigated.

(c) *Maneuvering pitching conditions.* The following conditions must be investigated:

(1) *Maximum pitch control displacement at V_A.* The airplane is assumed to be flying in steady level flight (point A_1, § 25.333(b)) and the cockpit pitch control is suddenly moved to obtain extreme nose up pitching acceleration. In defining the tail load, the response of the airplane must be taken into

account. Airplane loads that occur subsequent to the time when normal acceleration at the c.g. exceeds the positive limit maneuvering load factor (at point A_2 in § 25.333(b)), or the resulting tailplane normal load reaches its maximum, whichever occurs first, need not be considered.

(2) *Checked maneuver between V_A and V_D.* Nose-up checked pitching maneuvers must be analyzed in which the positive limit load factor prescribed in § 25.337 is achieved. As a separate condition, nose-down checked pitching maneuvers must be analyzed in which a limit load factor of 0g is achieved. In defining the airplane loads, the flight deck pitch control motions described in paragraphs (c)(2)(i) through (iv) of this section must be used:

(i) The airplane is assumed to be flying in steady level flight at any speed between V_A and V_D and the flight deck pitch control is moved in accordance with the following formula:

$\delta(t) = \delta_1 \sin(\omega t)$ for $0 \leq t \leq t_{max}$

Where—

δ_1 = the maximum available displacement of the flight deck pitch control in the initial direction, as limited by the control system stops, control surface stops, or by pilot effort in accordance with § 25.397(b);

$\delta(t)$ = the displacement of the flight deck pitch control as a function of time. In the initial direction, $\delta(t)$ is limited to δ_1. In the reverse direction, $\delta(t)$ may be truncated at the maximum available displacement of the flight deck pitch control as limited by the control system stops, control surface stops, or by pilot effort in accordance with 25.397(b);

$t_{max} = 3\pi/2\omega$;

ω = the circular frequency (radians/second) of the control deflection taken equal to the undamped natural frequency of the short period rigid mode of the airplane, with active control system effects included where appropriate; but not less than:

$$\omega = \frac{\pi V}{2V_A} \text{ radians per second;}$$

Where

V = the speed of the airplane at entry to the maneuver.

V_A = the design maneuvering speed prescribed in § 25.335(c).

(ii) For nose-up pitching maneuvers, the complete flight deck pitch control displacement history may be scaled down in amplitude to the extent necessary to ensure that the positive limit load factor prescribed in § 25.337 is not exceeded. For nose-down pitching maneuvers, the complete flight deck control displacement history may be scaled down in amplitude to the extent necessary to ensure that the normal acceleration at the center of gravity does not go below 0g.

(iii) In addition, for cases where the airplane response to the specified flight deck pitch control motion does not achieve the prescribed limit load factors, then the following flight deck pitch control motion must be used:

$\delta(t) = \delta_1 \sin(\omega t)$ for $0 \le t \le t_1$

$\delta(t) = \delta_1$ for $t_1 \le t \le t_2$

$\delta(t) = \delta_1 \sin(\omega[t + t_1 - t_2])$ for $t_2 \le t \le t_{max}$
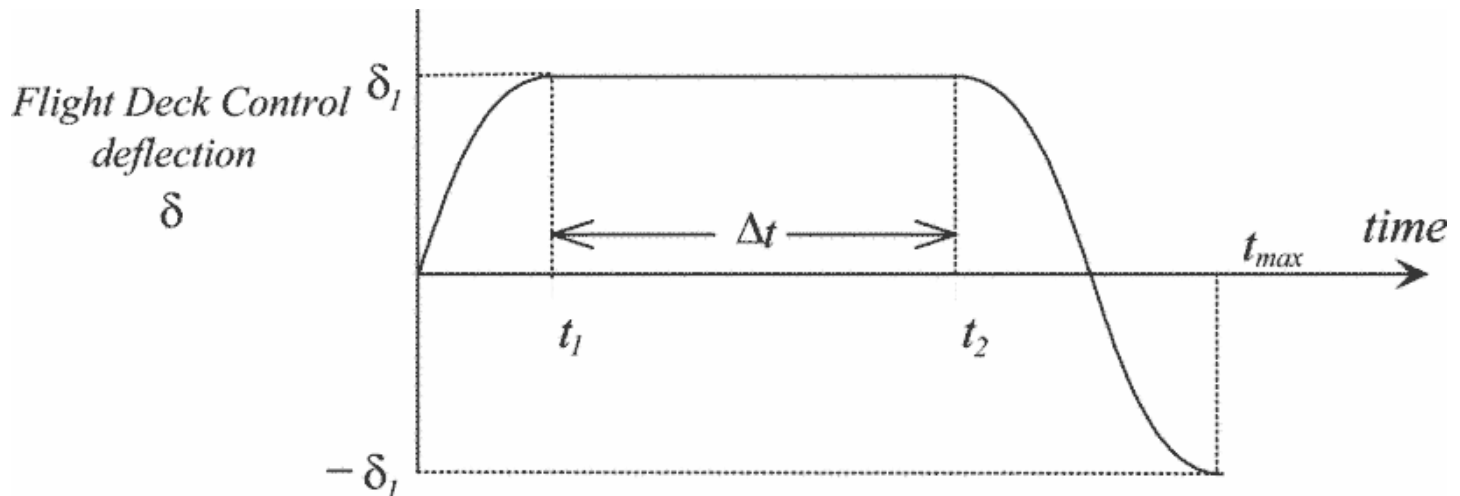
Where—

$t_1 = \pi/2\omega$

$t_2 = t_1 + \Delta t$

$t_{max} = t_2 + \pi/\omega;$

Δt = the minimum period of time necessary to allow the prescribed limit load factor to be achieved in the initial direction, but it need not exceed five seconds (see figure below).


(iv) In cases where the flight deck pitch control motion may be affected by inputs from systems (for example, by a stick pusher that can operate at high load factor as well as at 1g), then the effects of those systems shall be taken into account.

(v) Airplane loads that occur beyond the following times need not be considered:

(A) For the nose-up pitching maneuver, the time at which the normal acceleration at the center of gravity goes below 0g;

(B) For the nose-down pitching maneuver, the time at which the normal acceleration at the center of gravity goes above the positive limit load factor prescribed in § 25.337;

(C) $t_{max.}$.

§ 25.333 Flight maneuvering envelope.

(a) *General.* The strength requirements must be met at each combination of airspeed and load factor on and within the boundaries of the representative maneuvering envelope (*V-n* diagram) of paragraph (b) of this section. This envelope must also be used in determining the airplane structural operating limitations as specified in § 25.1501.

(b) *Maneuvering envelope.*
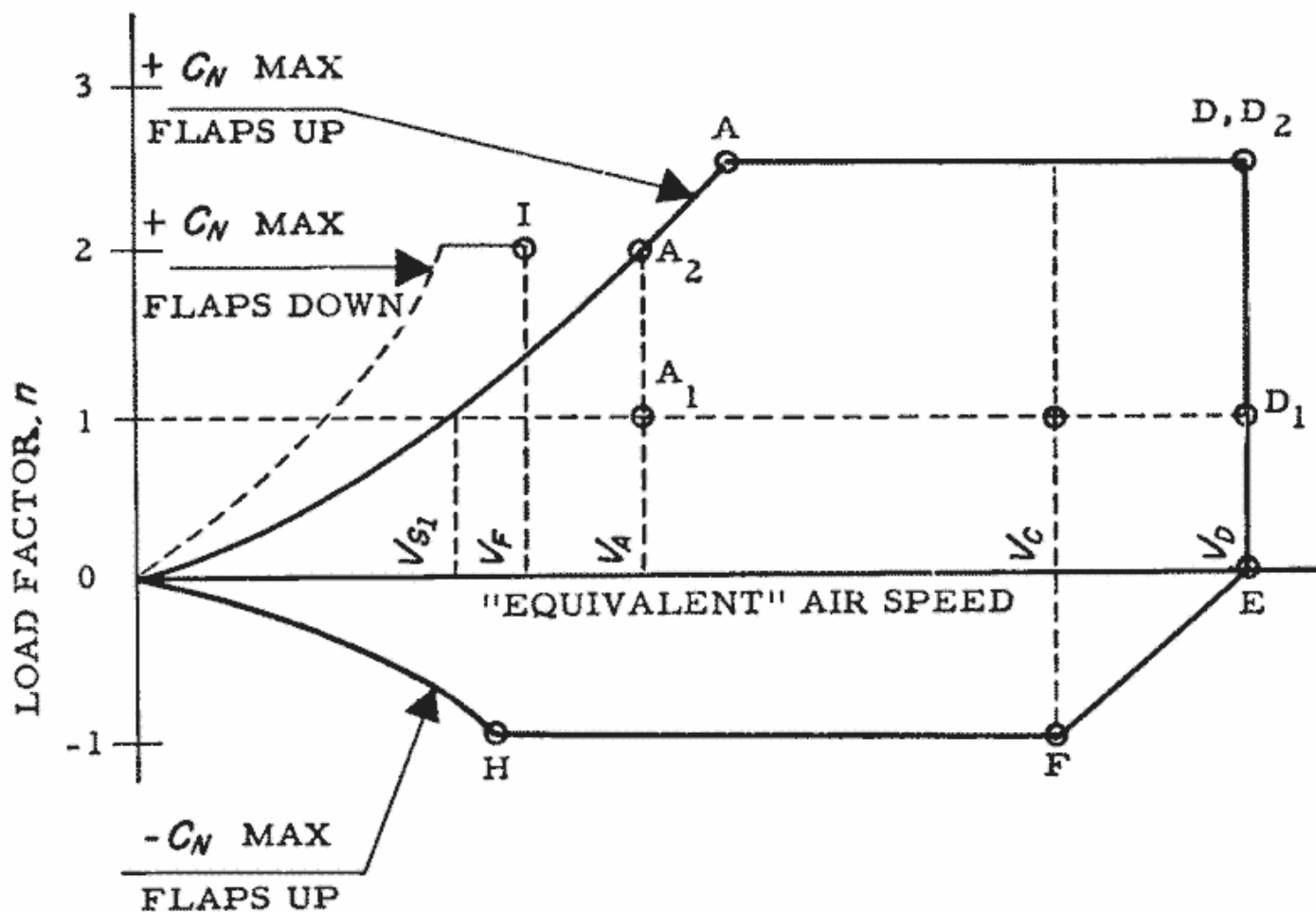
[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-86, 61 FR 5220, Feb. 9, 1996]

§ 25.335 Design airspeeds.

The selected design airspeeds are equivalent airspeeds (EAS). Estimated values of V_{S0} and V_{S1} must be conservative.

(a) *Design cruising speed, V_C.* For V_C, the following apply:

(1) The minimum value of V_C must be sufficiently greater than V_B to provide for inadvertent speed increases likely to occur as a result of severe atmospheric turbulence.

(2) Except as provided in § 25.335(d)(2), V_C may not be less than V_B + 1.32 U_{REF} (with U_{REF} as specified in § 25.341(a)(5)(i)). However V_C need not exceed the maximum speed in level flight at maximum continuous power for the corresponding altitude.

(3) At altitudes where V_D is limited by Mach number, V_C may be limited to a selected Mach number.

(b) *Design dive speed, V_D. V_D* must be selected so that V_C/M_C is not greater than 0.8 V_D/M_D, or so that the minimum speed margin between V_C/M_C and V_D/M_D is the greater of the following values:

(1) From an initial condition of stabilized flight at V_C/M_C, the airplane is upset, flown for 20 seconds along a flight path 7.5° below the initial path, and then pulled up at a load factor of 1.5*g* (0.5*g* acceleration increment). The speed increase occurring in this maneuver may be calculated if reliable or conservative aerodynamic data is used. Power as specified in § 25.175(b)(1)(iv) is assumed until the pullup is initiated, at which time power reduction and the use of pilot controlled drag devices may be assumed;

(2) The minimum speed margin must be enough to provide for atmospheric variations (such as horizontal gusts, and penetration of jet streams and cold fronts) and for instrument errors and airframe production variations. These factors may be considered on a probability basis. The margin at altitude where M_C is limited by compressibility effects must not less than 0.07M unless a lower margin is determined using a rational analysis that includes the effects of any automatic systems. In any case, the margin may not be reduced to less than 0.05M.

(c) *Design maneuvering speed V_A*. For V_A, the following apply:

(1) V_A may not be less than V_{S1} √n where—

(i) *n* is the limit positive maneuvering load factor at V_C; and

(ii) V_{S1} is the stalling speed with flaps retracted.

(2) V_A and V_S must be evaluated at the design weight and altitude under consideration.

(3) V_A need not be more than V_C or the speed at which the positive $C_{N\,max}$ curve intersects the positive maneuver load factor line, whichever is less.

(d) *Design speed for maximum gust intensity, V_B.*

(1) V_B may not be less than

$$V_{S1}\left[1 + \frac{K_g U_{ref} V_c a}{498w}\right]^{1/2}$$

where—

V_{S1} = the 1-g stalling speed based on C_{NAmax} with the flaps retracted at the particular weight under consideration;

V_c = design cruise speed (knots equivalent airspeed);

U_{ref} = the reference gust velocity (feet per second equivalent airspeed) from § 25.341(a)(5)(i);

w = average wing loading (pounds per square foot) at the particular weight under consideration.

$$K_g = \frac{.88\mu}{5.3+\mu}$$

$$\mu = \frac{2w}{\rho cag}$$

ρ = density of air (slugs/ft^3);

c = mean geometric chord of the wing (feet);

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

a = slope of the airplane normal force coefficient curve, C_{NA} per radian;

    (2) At altitudes where V_C is limited by Mach number—

        (i) V_B may be chosen to provide an optimum margin between low and high speed buffet boundaries; and,

        (ii) V_B need not be greater than V_C.

(e) *Design flap speeds, V_F.* For V_F, the following apply:

    (1) The design flap speed for each flap position (established in accordance with § 25.697(a)) must be sufficiently greater than the operating speed recommended for the corresponding stage of flight (including balked landings) to allow for probable variations in control of airspeed and for transition from one flap position to another.

    (2) If an automatic flap positioning or load limiting device is used, the speeds and corresponding flap positions programmed or allowed by the device may be used.

    (3) *V_F* may not be less than—

        (i) 1.6 V_{S1} with the flaps in takeoff position at maximum takeoff weight;

        (ii) 1.8 V_{S1} with the flaps in approach position at maximum landing weight, and

        (iii) 1.8 V_{S0} with the flaps in landing position at maximum landing weight.

(f) *Design drag device speeds, V_{DD}.* The selected design speed for each drag device must be sufficiently greater than the speed recommended for the operation of the device to allow for probable variations in speed control. For drag devices intended for use in high speed descents, V_{DD} may not be less than V_D. When an automatic drag device positioning or load limiting means is used, the speeds and corresponding drag device positions programmed or allowed by the automatic means must be used for design.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-23, 35 FR 5672, Apr. 8, 1970; Amdt. 25-86, 61 FR 5220, Feb. 9, 1996; Amdt. 25-91, 62 FR 40704, July 29, 1997]

§ 25.337 Limit maneuvering load factors.

(a) Except where limited by maximum (static) lift coefficients, the airplane is assumed to be subjected to symmetrical maneuvers resulting in the limit maneuvering load factors prescribed in this section. Pitching velocities appropriate to the corresponding pull-up and steady turn maneuvers must be taken into account.

(b) The positive limit maneuvering load factor n for any speed up to Vn may not be less than 2.1 + 24,000/ (W + 10,000) except that n may not be less than 2.5 and need not be greater than 3.8—where W is the design maximum takeoff weight.

(c) The negative limit maneuvering load factor—

(1) May not be less than −1.0 at speeds up to V_C; and

(2) Must vary linearly with speed from the value at V_C to zero at V_D.

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

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-23, 35 FR 5672, Apr. 8, 1970]

§ 25.341 Gust and turbulence loads.

(a) *Discrete Gust Design Criteria.* The airplane is assumed to be subjected to symmetrical vertical and lateral gusts in level flight. Limit gust loads must be determined in accordance with the provisions:

(1) Loads on each part of the structure must be determined by dynamic analysis. The analysis must take into account unsteady aerodynamic characteristics and all significant structural degrees of freedom including rigid body motions.

(2) The shape of the gust must be:

$$U = \frac{U_{ds}}{2} \left[1 - \text{Cos} \left(\frac{\pi s}{H} \right) \right]$$

for 0 ≤s ≤2H

where—

s = distance penetrated into the gust (feet);

U_{ds} = the design gust velocity in equivalent airspeed specified in paragraph (a)(4) of this section; and

H = the gust gradient which is the distance (feet) parallel to the airplane's flight path for the gust to reach its peak velocity.

(3) A sufficient number of gust gradient distances in the range 30 feet to 350 feet must be investigated to find the critical response for each load quantity.

(4) The design gust velocity must be:

$$U_{ds} = U_{ref}F_g \left(\frac{H}{350}\right)^{1/6}$$

where—

U_{ref} = the reference gust velocity in equivalent airspeed defined in paragraph (a)(5) of this section.

F_g = the flight profile alleviation factor defined in paragraph (a)(6) of this section.

(5) The following reference gust velocities apply:

(i) At airplane speeds between V_B and V_C: Positive and negative gusts with reference gust velocities of 56.0 ft/sec EAS must be considered at sea level. The reference gust velocity may be reduced linearly from 56.0 ft/sec EAS at sea level to 44.0 ft/sec EAS at 15,000 feet. The reference gust velocity may be further reduced linearly from 44.0 ft/sec EAS at 15,000 feet to 20.86 ft/sec EAS at 60,000 feet.

(ii) At the airplane design speed V_D: The reference gust velocity must be 0.5 times the value obtained under § 25.341(a)(5)(i).

(6) The flight profile alleviation factor, F_g, must be increased linearly from the sea level value to a value of 1.0 at the maximum operating altitude defined in § 25.1527. At sea level, the flight profile alleviation factor is determined by the following equation:

$$F_g = 0.5\left(F_{gz} + F_{gm}\right)$$

Where:

$$F_{gz} = 1 - \frac{Z_{mo}}{250000};$$

$$F_{gm} = \sqrt{R_2 \mathrm{Tan}\left(\frac{\pi R_1}{4}\right)};$$

$$R_1 = \frac{\mathrm{Maximum\ Landing\ Weight}}{\mathrm{Maximum\ Take\text{-}off\ Weight}};$$

$$R_2 = \frac{\mathrm{Maximum\ Zero\ Fuel\ Weight}}{\mathrm{Maximum\ Take\text{-}off\ Weight}};$$

Z_{mo} = Maximum operating altitude defined in § 25.1527 (feet).

(7) When a stability augmentation system is included in the analysis, the effect of any significant system nonlinearities should be accounted for when deriving limit loads from limit gust conditions.

(b) *Continuous turbulence design criteria.* The dynamic response of the airplane to vertical and lateral continuous turbulence must be taken into account. The dynamic analysis must take into account unsteady aerodynamic characteristics and all significant structural degrees of freedom including rigid body motions. The limit loads must be determined for all critical altitudes, weights, and weight distributions as specified in § 25.321(b), and all critical speeds within the ranges indicated in § 25.341(b)(3).

(1) Except as provided in paragraphs (b)(4) and (5) of this section, the following equation must be used:

$$P_L = P_{L-1g} \pm U_\sigma \bar{A}$$

Where—

P_L = limit load;

P_{L-1g} = steady 1g load for the condition;

$\bar{A}$ = ratio of root-mean-square incremental load for the condition to root-mean-square turbulence velocity; and

U_σ = limit turbulence intensity in true airspeed, specified in paragraph (b)(3) of this section.

(2) Values of $\bar{A}$ must be determined according to the following formula:

$$\bar{A} = \sqrt{\int_0^\infty |H(\Omega)|^2 \, \Phi(\Omega) \, d\Omega}$$

Where—

$H(\Omega)$ = the frequency response function, determined by dynamic analysis, that relates the loads in the aircraft structure to the atmospheric turbulence; and

$\Phi(\Omega)$ = normalized power spectral density of atmospheric turbulence given by—

$$\Phi(\Omega) = \frac{L}{\pi} \frac{1 + \frac{8}{3}(1.339\Omega L)^2}{\left[1 + (1.339\Omega L)^2\right]^{11/6}}$$

Where—

Ω = reduced frequency, radians per foot; and

L = scale of turbulence = 2,500 ft.

(3) The limit turbulence intensities, U_σ, in feet per second true airspeed required for compliance with this paragraph are—

   (i) At airplane speeds between V_B and V_C:

      $U_\sigma = U_{\sigma ref} F_g$

   Where—

   $U_{\sigma ref}$ is the reference turbulence intensity that varies linearly with altitude from 90 fps (TAS) at sea level to 79 fps (TAS) at 24,000 feet and is then constant at 79 fps (TAS) up to the altitude of 60,000 feet.

   F_g is the flight profile alleviation factor defined in paragraph (a)(6) of this section;

   (ii) At speed V_D: U_σ is equal to $^1/_2$ the values obtained under paragraph (b)(3)(i) of this section.

   (iii) At speeds between V_C and V_D: U_σ is equal to a value obtained by linear interpolation.

   (iv) At all speeds, both positive and negative incremental loads due to continuous turbulence must be considered.

(4) When an automatic system affecting the dynamic response of the airplane is included in the analysis, the effects of system non-linearities on loads at the limit load level must be taken into account in a realistic or conservative manner.

(5) If necessary for the assessment of loads on airplanes with significant non-linearities, it must be assumed that the turbulence field has a root-mean-square velocity equal to 40 percent of the U_σ values specified in paragraph (b)(3) of this section. The value of limit load is that load with the same probability of exceedance in the turbulence field as $\overline{A}U_\sigma$ of the same load quantity in a linear approximated model.

(c) *Supplementary gust conditions for wing-mounted engines.* For airplanes equipped with wing-mounted engines, the engine mounts, pylons, and wing supporting structure must be designed for the maximum response at the nacelle center of gravity derived from the following dynamic gust conditions applied to the airplane:

(1) A discrete gust determined in accordance with § 25.341(a) at each angle normal to the flight path, and separately,

(2) A pair of discrete gusts, one vertical and one lateral. The length of each of these gusts must be independently tuned to the maximum response in accordance with § 25.341(a). The penetration of the airplane in the combined gust field and the phasing of the vertical and lateral component gusts must be established to develop the maximum response to the gust pair. In the absence of a more rational analysis, the following formula must be used for each of the maximum engine loads in all six degrees of freedom:

$$P_L = P_{L-1g} \pm 0.85\sqrt{L_V^2 + L_L^2}$$

Where—

P_L = limit load;

P_{L-1g} = steady 1g load for the condition;

L_V = peak incremental response load due to a vertical gust according to § 25.341(a); and

L_L = peak incremental response load due to a lateral gust according to § 25.341(a).

[Doc. No. 27902, 61 FR 5221, Feb. 9, 1996; 61 FR 9533, Mar. 8, 1996; Doc. No. FAA-2013-0142; 79 FR 73467, Dec. 11, 2014; Amdt. 25-141, 80 FR 4762, Jan. 29, 2015; 80 FR 6435, Feb. 5, 2015]

§ 25.343 Design fuel and oil loads.

(a) The disposable load combinations must include each fuel and oil load in the range from zero fuel and oil to the selected maximum fuel and oil load. A structural reserve fuel condition, not exceeding 45 minutes of fuel under the operating conditions in § 25.1001(e) and (f), as applicable, may be selected.

(b) If a structural reserve fuel condition is selected, it must be used as the minimum fuel weight condition for showing compliance with the flight load requirements as prescribed in this subpart. In addition—

(1) The structure must be designed for a condition of zero fuel and oil in the wing at limit loads corresponding to—

(i) A maneuvering load factor of + 2.25; and

(ii) The gust and turbulence conditions of § 25.341(a) and (b), but assuming 85% of the gust velocities prescribed in § 25.341(a)(4) and 85% of the turbulence intensities prescribed in § 25.341(b)(3).

(2) Fatigue evaluation of the structure must account for any increase in operating stresses resulting from the design condition of paragraph (b)(1) of this section; and

(3) The flutter, deformation, and vibration requirements must also be met with zero fuel.

§ 25.345 High lift devices.

(a) If wing flaps are to be used during takeoff, approach, or landing, at the design flap speeds established for these stages of flight under § 25.335(e) and with the wing flaps in the corresponding positions, the airplane is assumed to be subjected to symmetrical maneuvers and gusts. The resulting limit loads must correspond to the conditions determined as follows:

    (1) Maneuvering to a positive limit load factor of 2.0; and

    (2) Positive and negative gusts of 25 ft/sec EAS acting normal to the flight path in level flight. Gust loads resulting on each part of the structure must be determined by rational analysis. The analysis must take into account the unsteady aerodynamic characteristics and rigid body motions of the aircraft. The shape of the gust must be as described in § 25.341(a)(2) except that—

U_{ds} = 25 ft/sec EAS;

H = 12.5 c; and

c = mean geometric chord of the wing (feet).

(b) The airplane must be designed for the conditions prescribed in paragraph (a) of this section, except that the airplane load factor need not exceed 1.0, taking into account, as separate conditions, the effects of—

    (1) Propeller slipstream corresponding to maximum continuous power at the design flap speeds V_F, and with takeoff power at not less than 1.4 times the stalling speed for the particular flap position and associated maximum weight; and

    (2) A head-on gust of 25 feet per second velocity (EAS).

(c) If flaps or other high lift devices are to be used in en route conditions, and with flaps in the appropriate position at speeds up to the flap design speed chosen for these conditions, the airplane is assumed to be subjected to symmetrical maneuvers and gusts within the range determined by—

    (1) Maneuvering to a positive limit load factor as prescribed in § 25.337(b); and

    (2) The vertical gust and turbulence conditions prescribed in § 25.341(a) and (b).

(d) The airplane must be designed for a maneuvering load factor of 1.5 g at the maximum take-off weight with the wing-flaps and similar high lift devices in the landing configurations.

§ 25.349 Rolling conditions.

The airplane must be designed for loads resulting from the rolling conditions specified in paragraphs (a) and (b) of this section. Unbalanced aerodynamic moments about the center of gravity must be reacted in a rational or conservative manner, considering the principal masses furnishing the reacting inertia forces.

(a) *Maneuvering.* The following conditions, speeds, and aileron deflections (except as the deflections may be limited by pilot effort) must be considered in combination with an airplane load factor of zero and of two-thirds of the positive maneuvering factor used in design. In determining the required aileron deflections, the torsional flexibility of the wing must be considered in accordance with § 25.301(b):

  (1) Conditions corresponding to steady rolling velocities must be investigated. In addition, conditions corresponding to maximum angular acceleration must be investigated for airplanes with engines or other weight concentrations outboard of the fuselage. For the angular acceleration conditions, zero rolling velocity may be assumed in the absence of a rational time history investigation of the maneuver.

  (2) At V_A, a sudden deflection of the aileron to the stop is assumed.

  (3) At V_C, the aileron deflection must be that required to produce a rate of roll not less than that obtained in paragraph (a)(2) of this section.

  (4) At V_D, the aileron deflection must be that required to produce a rate of roll not less than one-third of that in paragraph (a)(2) of this section.

(b) *Unsymmetrical gusts.* The airplane is assumed to be subjected to unsymmetrical vertical gusts in level flight. The resulting limit loads must be determined from either the wing maximum airload derived directly from § 25.341(a), or the wing maximum airload derived indirectly from the vertical load factor calculated from § 25.341(a). It must be assumed that 100 percent of the wing air load acts on one side of the airplane and 80 percent of the wing air load acts on the other side.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-23, 35 FR 5672, Apr. 8, 1970; Amdt. 25-86, 61 FR 5222, Feb. 9, 1996; Amdt. 25-94, 63 FR 8848, Feb. 23, 1998]

§ 25.351 Yaw maneuver conditions.

The airplane must be designed for loads resulting from the yaw maneuver conditions specified in paragraphs (a) through (d) of this section at speeds from V_{MC} to V_D. Unbalanced aerodynamic moments about the center of gravity must be reacted in a rational or conservative manner considering the airplane inertia forces. In computing the tail loads the yawing velocity may be assumed to be zero.

(a) With the airplane in unaccelerated flight at zero yaw, it is assumed that the cockpit rudder control is suddenly displaced to achieve the resulting rudder deflection, as limited by:

  (1) The control system on control surface stops; or

  (2) A limit pilot force of 300 pounds from V_{MC} to V_A and 200 pounds from V_C/M_C to V_D/M_D, with a linear variation between V_A and V_C/M_C.

(b) With the cockpit rudder control deflected so as always to maintain the maximum rudder deflection available within the limitations specified in paragraph (a) of this section, it is assumed that the airplane yaws to the overswing sideslip angle.

(c) With the airplane yawed to the static equilibrium sideslip angle, it is assumed that the cockpit rudder control is held so as to achieve the maximum rudder deflection available within the limitations specified in paragraph (a) of this section.

(d) With the airplane yawed to the static equilibrium sideslip angle of paragraph (c) of this section, it is assumed that the cockpit rudder control is suddenly returned to neutral.

[Amdt. 25-91, 62 FR 40704, July 29, 1997]

§ 25.353 Rudder control reversal conditions.

Airplanes with a powered rudder control surface or surfaces must be designed for loads, considered to be ultimate, resulting from the yaw maneuver conditions specified in paragraphs (a) through (e) of this section at speeds from V_{MC} to V_C/M_C. Any permanent deformation resulting from these ultimate load conditions must not prevent continued safe flight and landing. The applicant must evaluate these conditions with the landing gear retracted and speed brakes (and spoilers when used as speed brakes) retracted. The applicant must evaluate the effects of flaps, flaperons, or any other aerodynamic devices when used as flaps, and slats-extended configurations, if they are used in en route conditions. Unbalanced aerodynamic moments about the center of gravity must be reacted in a rational or conservative manner considering the airplane inertia forces. In computing the loads on the airplane, the yawing velocity may be assumed to be zero. The applicant must assume a pilot force of 200 pounds when evaluating each of the following conditions:

(a) With the airplane in unaccelerated flight at zero yaw, the flightdeck rudder control is suddenly and fully displaced to achieve the resulting rudder deflection, as limited by the control system or the control surface stops.

(b) With the airplane yawed to the overswing sideslip angle, the flightdeck rudder control is suddenly and fully displaced in the opposite direction, as limited by the control system or control surface stops.

(c) With the airplane yawed to the opposite overswing sideslip angle, the flightdeck rudder control is suddenly and fully displaced in the opposite direction, as limited by the control system or control surface stops.

(d) With the airplane yawed to the subsequent overswing sideslip angle, the flightdeck rudder control is suddenly and fully displaced in the opposite direction, as limited by the control system or control surface stops.

(e) With the airplane yawed to the opposite overswing sideslip angle, the flightdeck rudder control is suddenly returned to neutral.

[Amdt. No. 25-147, 87 FR 71210, Nov. 22, 2022]