(2) The deformations involved are fully accounted for in the analysis; or

(3) The methods and assumptions used are sufficient to cover the effects of these deformations.

(c) Where structural flexibility is such that any rate of load application likely to occur in the operating conditions might produce transient stresses appreciably higher than those corresponding to static loads, the effects of this rate of application must be considered.

(d) [Reserved]

(e) The airplane must be designed to withstand any vibration and buffeting that might occur in any likely operating condition up to V_D/M_D, including stall and probable inadvertent excursions beyond the boundaries of the buffet onset envelope. This must be shown by analysis, flight tests, or other tests found necessary by the Administrator.

(f) Unless shown to be extremely improbable, the airplane must be designed to withstand any forced structural vibration resulting from any failure, malfunction or adverse condition in the flight control system. These must be considered limit loads and must be investigated at airspeeds up to V_C/M_C.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25–23, 35 FR 5672, Apr. 8, 1970; Amdt. 25–54, 45 FR 60172, Sept. 11, 1980; Amdt. 25–77, 57 FR 28949, June 29, 1992; Amdt. 25–86, 61 FR 5220, Feb. 9, 1996]

§25.307 Proof of structure.

(a) Compliance with the strength and deformation requirements of this subpart must be shown for each critical loading condition. Structural analysis may be used only if the structure conforms to that for which experience has shown this method to be reliable. In other cases, substantiating tests must be made to load levels that are sufficient to verify structural behavior up to loads specified in §25.305.

(b)–(c) [Reserved]

(d) When static or dynamic tests are used to show compliance with the requirements of §25.305(b) for flight structures, appropriate material correction factors must be applied to the test results, unless the structure, or part thereof, being tested has features such that a number of elements contribute to the total strength of the structure and the failure of one element results in the redistribution of the load through alternate load paths.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25–23, 35 FR 5672, Apr. 8, 1970; Amdt. 25–54, 45 FR 60172, Sept. 11, 1980; Amdt. 25–72, 55 FR 29775, July 20, 1990; Amdt. 25–139, 79 FR 59429, Oct. 2, 2014]

FLIGHT LOADS

§25.321 General.

(a) Flight load factors 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 load factor is one in which the aerodynamic force acts upward with respect to the airplane.

(b) Considering compressibility effects at each speed, compliance with the flight load requirements of this subpart must be shown—

(1) At each critical altitude within the range of altitudes selected by the applicant;

(2) At each weight from the design minimum weight to the design maximum weight appropriate to each particular flight load condition; and

(3) For each required altitude and weight, for any practicable distribution of disposable load within the operating limitations recorded in the Airplane Flight Manual.

(c) Enough points on and within the boundaries of the design envelope must be investigated to ensure that the maximum load for each part of the airplane structure is obtained.

(d) The significant forces acting on the airplane must be placed in equilibrium in a rational or conservative manner. The linear inertia forces must be considered in equilibrium with the thrust and all aerodynamic loads, while the angular (pitching) inertia forces must be considered in equilibrium with thrust and all aerodynamic moments, including moments due to loads on components such as tail surfaces and nacelles. Critical thrust values in the range from zero to maximum continuous thrust must be considered.

[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]

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:

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

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