§ 4b.116 Take-off path.

The take-off path shall be considered to consist of the following five consecutive elements:

(a) The distance required to accelerate the airplane to the speed V_2, assuming the critical engine to fail at the speed V_1.

(b) The horizontal distance traversed and the height attained by the airplane in the time required to retract the landing gear when operating at the speed V_2 with:

(1) The critical engine inoperative, its propeller:

(i) Windmilling with the propeller control in a position normally used during take-off until (if applicable) its rotation has been stopped (see paragraph (c) (1) of this section),

(ii) If applicable, stopped for the remainder of the gear retraction time.

(2) The landing gear extended.

(c) If applicable, the horizontal distance traversed and the height attained by the airplane in the time elapsed from the end of element (b) until the rotation of the inoperative propeller has been stopped when:

(1) The operation of stopping the propeller is initiated not earlier than the instant the airplane has attained a total height of 50 feet above the take-off surface,

(2) The airplane speed is equal to V_2,

(3) The landing gear is retracted,

(4) The inoperative propeller is windmilling with the propeller control in a position normally used during take-off.

(d) The horizontal distance traversed and the height attained by the airplane in the time elapsed from the end of element (c) until the time limit on the use of take-off power is reached, while operating at the speed V_2, with:

(1) The inoperative propeller stopped,

(2) The landing gear retracted.

(e) The slope of the flight path followed by the airplane in the configuration of element (d), but drawing not more than maximum continuous power on the operating engine(s).

§ 4b.116–1 Approval of automatic propeller feathering installations for use in establishing the take-off path (FAA policies which apply to § 4b.116).

The take-off path may be modified by permitting a feathered propeller instead of windmilling after the necessary time interval has elapsed from the instant of engine failure to complete feathering of the propeller. If it can be shown that the net work produced by the feathering propeller from the instant of engine failure to completion of feathering under all types of engine failure is positive using a datum based on feathered propeller drag, then it is permissible to assume that the propeller of the failed engine is in the feathered drag condition from the instant of attainment of the take-off climb speed V_2. (See §§ 4b.10–2, 4b.401–1, and 4b.700–1.)

[Supp. 23, 19 F. R. 1818, Apr. 2, 1954]

§ 4b.116–2 Determination of the take-off path (FAA policies which apply to § 4b.116).

(a) *Recommended procedure.* The recommended procedure for obtaining the take-off path is to determine the ground and climb portions separately and piece the corrected data together. The take-off flight path should be demonstrated in accordance with the following provisions:

(1) Three accelerations should be made during which the airplane is accelerated from a complete stop using all engines to speeds bracketing speed V_1 at which speed the critical engine fuel mixture is cut and the acceleration continued to speed V_2 with the inoperative engine propeller windmilling[1] in the take-off pitch setting. If V_1 is less than V_2, a take-off should be made on one of the above runs when the critical engine is failed at the lowest V_1 speed.

(2) The take-off flap setting should be maintained throughout the take-off flight path. If more than one flap setting is to be used for take-off, additional tests should be included to cover the flap range (see § 4b.118–1 (d) (2)).

[1] When a satisfactory fully automatic propeller feathering device is installed on the airplane, advantage of such a device may be used in showing compliance with this section. See § 4b.116–1 for policies covering automatic propeller feathering systems.

(3) See § 4b.115–1 (a) (4) for instrumentation requirements.

(4) A special tolerance of not greater than ±2 percent of the maximum take-off weight is allowable for the ground portion of the accelerate distance.

(b) *General test program*—(1) *Accelerate to take-off safety speed, V_2 § 4b.116 (a)*—(i) *Configuration.* These tests should be conducted in the configuration that follows:

Weight—Maximum take-off and one lower.
C. G. position—Most forward.
Wing flaps—Take-off position.
Landing gear—Extended.
Operating engine(s)—Take-off rpm and manifold pressure, cowl flaps in take-off position (see § 4b.118–1 (d) (1)).
Critical inoperative engine—Fuel mixture cut on engine most critical performancewise (see § 4b.118–1 (e) (2)), propeller windmilling in take-off pitch (feathered if automatic feathering device is installed) and cowl flaps in take-off position (see § 4b.118–1 (d) (1)).

(ii) *Test procedure and required data.* The airplane should be accelerated from a complete stop to the V_1 speed with all engines operating. The critical engine fuel mixture should be cut at the V_1 speed and the acceleration should be continued until V_2 speed is reached with the propeller of the inoperative engine windmilling in the take-off pitch. The airplane's path relative to the runway should be recorded against time in a manner to determine the horizontal distance-time history. In addition the following data should be recorded:

Pressure altitude.
Ambient air temperature.
Airplane gross weight.
Rpm.
Manifold pressure.
Torque pressure.
Mixture setting.
Cowl flap position.
Wing flap position.
Time, distance and speed at engine cut.
Time, distance and speed when V_2 is reached.
Slope of field.
Direction of run.

In addition, humidity, wind direction and wind velocity should be recorded adjacent to the runway at a height of 6 feet above the runway surface.

(2) *Initial take-off flight path segment test, § 4b.116 (b)*—(i) *Configuration.* This test should be conducted in the configuration that follows:

Weight—Maximum take-off and one lower.
C. G. position—Optional (see § 4b.118–1 (c) (2)).
Wing flaps—Take-off position.
Landing gear—Extended.
Operating engine(s)—Take-off rpm and manifold pressure or full throttle, mixture setting for take-off, carburetor air heat control at cold and cowl flaps in take-off position (see § 4b.118–1 (d) (1)).
Critical inoperative engine—Throttle closed on engine most critical performancewise (see § 4b.118–1 (e) (2)), propeller windmilling in take-off pitch, (feathered if automatic feathering device is installed, see § 4b.120–1), mixture setting at idle cut-off and cowl flaps in take-off position (see § 4b.118–1(d)(1)).

(ii) *Test procedure and required data.* The airplane should be climbed at the take-off safety speed, V_2. See § 4b.118–1 for test procedure and required data in connection with climb tests.

(3) *Second take-off flight path climb segment test, § 4b.116 (c)*—(i) *Configuration.* This test should be conducted in the configuration that follows:

Weight—Maximum take-off and one lower.
C. G. position—Optional (see § 4b.118–1 (c) (2)).
Wing flaps—Take-off position.
Landing gear—Retracted.
Operating engine(s)—Take-off rpm and manifold pressure or full throttle, mixture setting for take-off, carburetor air heat control at cold and cowl flaps in take-off position (see § 4b.118–1 (d) (1)).
Critical inoperative engine—Throttle closed on engine most critical performancewise (see § 4b.118–1 (a) (2)), propeller windmilling in take-off pitch (feathered if automatic feathering device is installed, see § 4b.120–1), mixture setting at idle cut-off and cowl flaps in take-off position (see § 4b.118–1(d)(1)).

(ii) *Test procedure and required data.* The airplane should be climbed at the take-off safety speed, V_2. See § 4b.118–1 for test procedure and required data in connection with climb tests.

(4) *Third take-off flight path climb segment test, § 4b.116 (d)*—(i) *Configuration.* This test should be conducted in the configuration that follows:

Weight—Maximum take-off and one lower.
C. G. position—Optional (see § 4b.118–1 (c) (2)).
Wing flaps—Take-off position.
Landing gear—Retracted.
Operating engine(s)—Take-off rpm and manifold pressure or full throttle, mixture setting for take-off, carburetor air heat control at cold and cowl flaps in take-off position (see § 4b.118–1 (d) (1)).
Critical inoperative engine—Throttle closed on engine most critical performancewise (see § 4b.118–1 (e) (2)), propeller feathered and cowl flaps in minimum drag position.

(ii) *Test procedure and required data.* The airplane should be climbed at the take-off safety speed, V_1. See §4b.118–1 for test procedure and required data in connection with climb tests.

(5) *Fourth take-off flight path climb segment test,* § *4b.116 (e)*—(i) *Configuration.* This configuration should be the same as for the third take-off flight path climb segment except that maximum continuous power is used on the operating engine(s).

(ii) *Test procedure and required data.* The airplane should be climbed at the take-off safety speed, V_1. See § 4b.118–1 for test procedure and required data in connection with climb tests.

[Supp. 24, 19 F.R. 4462, July 20, 1954, as amended by Supp. 34, 22 F.R. 6963, Aug. 29, 1957]

§ 4b.117 Temperature accountability.

Operating correction factors for take-off weight and take-off distance shall be determined to account for temperatures above and below standard, and when approved by the Administrator they shall be included in the Airplane Flight Manual. These factors shall be obtained as follows:

(a) For any specific airplane type, the average full temperature accountability shall be computed for the range of weights of the airplane, altitudes above sea level, and ambient temperatures required by the expected operating conditions. Account shall be taken of the temperature effect on both the aerodynamic characteristics of the airplane and on the engine power. The full temperature accountability shall be expressed per degree of temperature in terms of a weight correction, a take-off distance correction, and a change, if any, in the critical-engine-failure speed V_1.

(b) The operating correction factors for the airplane weight and take-off distance shall be at least one-half of the full accountability values. The value of V_1 shall be further corrected by the average amount necessary to assure that the airplane can stop within the runway length at the ambient temperature, except that the corrected value of V_1 shall not be less than a minimum at which the airplane can be controlled with the critical engine inoperative.

§ 4b.118 Climb; general.

Compliance shall be shown with the climb requirements of §§ 4b.119 through 4b.121.

§ 4b.118–1 Test procedures for the determination of climb performance (FAA policies which apply to § 4b.118).

The test conditions and methods in paragraphs (a) through (i) of this section assume that the flight test data do not exhibit excessive scatter in points and that if such scatter makes the accuracy of the climb slopes questionable, additional tests and/or the applicants' previous flight test data for the particular configuration involved should be available. The following methods are also based upon consistent flight test data that can be properly correlated, and the use of previous acceptable test and correction methods. All new tests and correction methods will be judged upon their own merits. Polar curve or other equivalent methods are acceptable. The number of climb tests recommended for each case in paragraphs (a) and (b) of this section represents a minimum, and in certain instances it is possible that more tests may be necessary.

(a) *For all take-off path segments, landing and approach climb.* (1) If it is desired to show that the required climb is met at the highest altitude and heaviest weight to be certificated, a constant rate of climb curve with altitude should be acceptable. One good climb should be satisfactory if it is 50 ft./min., or more, in excess of the required climb at the highest altitude for which certification is desired. Three climbs (at same altitude) should be made if the R/C is less than 50 ft./min. in excess of the required climb. (No climb variation with weight or altitude.)

(2) If it is desired to determine the rate of climb vs. altitude curve:

(i) For a sea level engine, or the critical altitude above the maximum altitude of certification (no breaks in the curve) where the altitude range is not in excess of 8,000 feet, two good climbs at each weight over the altitude range if weight spread is in excess of 10 percent but not less than 4 climbs if only one weight is used. If the altitude range is in excess of 8,000 feet, three climbs at each weight over the altitude range (6 climbs if weight range is in excess of 10 percent) should be conducted.

(ii) For installations where a critical altitude is within the altitude range (one break in the curve) two climbs on one slope of the curve should be conducted at each weight and at least one climb on the other slope (6 climbs for a weight spread of 10 percent or more). If the altitude range is in excess of 8,000 feet, consideration should be given to the need for a fourth climb at each weight.

(iii) For installations where two breaks in the R/C curve occur within the certification range, four climbs at each weight should be made.

(b) *For all engine enroute, one-engine-inoperative enroute and two-engines-inoperative enroute climbs.* (1) The regulations specify that these climbs should be determined at all altitudes of expected use and weight of certification. For each weight at least the following climbs should be conducted:

Altitude range	SL–8,000	SL–17,000	SL–25,000
No breaks in curve (1 slope)	[1] 2	[1] 3	3
1 break (2 slopes)	[1] 3	4	4
2 breaks (3 slopes)		4	4–5
3 breaks (4 slopes)		5	5

[1] But not less than 4 if only 1 weight is used.

(2) When an airplane is approved for more than one landing flap position, the climb requirements for the cruising and enroute configurations should be based upon the stall speed with the maximum sea level landing flap position for which the airplane is eligible.

(3) Where several climb points are available at the same test conditions the average point should be determined by averaging all consistent points not using the obviously erratic points (either high or low).

(c) *Weight and C. G. position.* (1) The climb tests should be conducted at maximum take-off weight for take-off climbs and at maximum landing weight for landing climbs. Climbs should also be made at an optional lower weight for both the take-off and landing configuration.

(2) Climbs may be made at any C. G. position except where the applicant elects to vary the stalling speed with C. G. position in which case the most critical C. G. position should be used. (See § 4b.112–1 (b).)

(d) *Airplane configuration.* (1) The cowl flaps should be set in the required position prior to conducting climb tests. The position of the cowl flaps for the take-off segments should comply with the provisions of the take-off cooling tests of § 4b.453.

(2) If more than one wing flap setting is to be used for take-off or landing, additional tests should be included to cover the flap range (see § 4b.111–1 (b)).

(e) *Engine power.* (1) The power should be stabilized prior to conducting the climb tests. The climbs should be made at constant power or at constant throttle setting. Unless limited by engine temperature, tests should be run for at least 3 minutes at take-off power. If limited by temperature, short duration tests of approximately 1 minute duration should be acceptable provided the stabilized climbing speed is attained by accelerating from a lower speed. Where maximum continuous power is required, climb tests should be of 5 minutes duration or not necessarily more than climbs of 2,000 feet but in any case not less than 3 minutes. If climb tests are conducted for short durations, such as take-off climbs which are limited by an engine rating of two minutes for take-off power, consideration should be given to the necessity of conducting an adequate number of tests in order to obtain results which are representative of the actual performance.

(2) For the one-engine-inoperative climb tests, it may be assumed that the critical inoperative engine, performance-wise, is the higher powered outboard engine unless there is evidence to indicate that another engine is more critical.

(3) For all climb tests, the power plant equipment and accessories appropriate to the specific configuration being tested should be in operation. During each test, a record should be made of such accessories in operation and of the particular engine from which power is being absorbed.

(f) *Climb speeds.* The climb speeds are to be selected by the applicant, but should be consistent with the performance and cooling requirements involved. The air speed should be stabilized prior to conducting the climb tests.

(g) *Position of wings.* The airplane's wings should be maintained in a level attitude during all take-off climb tests with one engine inoperative.

(h) *Climbs to be made in free air.* All climbs should be conducted in free air (without ground effect).

(i) *Data.* In addition to the following items, the data necessary to establish the weight and C. G. position during the climb tests should be recorded.

Pressure altitude recorded at 15 second intervals.
Humidity recorded at 15 second intervals.
Air speed recorded at 15 second intervals.
Rpm.
Manifold pressure.
Torque pressure.
Carburetor air temperature.
Mixture setting.
Throttle setting.
Cowl flap position.
Wing flap position.
Landing gear position.

[Supp. 24, 19 F. R. 4452, July 20, 1954]

§ 4b.119 Climb; all engines operating.

(a) *Cruising configuration.* In the cruising configuration the steady rate of climb in feet per minute at 5,000 feet shall not be less than $8V_{s_0}$. In addition the steady rate of climb shall be determined at any altitude at which the airplane is expected to operate and at any weight within the range of weights to be specified in the airworthiness certificate. The cruising configuration shall be with:

(1) Landing gear fully retracted,

(2) Wing flaps in the most favorable position,

(3) Cowl flaps (or other means of controlling the engine cooling) in the position which provides adequate cooling in the hot-day condition,

(4) Center of gravity in the most unfavorable position,

(5) All engines operating within the maximum continuous power limitations,

(6) Maximum take-off weight.

(b) *Landing configuration.* In the landing configuration the steady rate of climb in feet per minute shall not be less than $0.07\ V_{s_0}{}^2$ at any altitude within the range for which landing weight is to be specified in the certificate, with:

(1) Landing gear extended,

(2) Wing flaps in the landing position (see §§ 4b. 111 and 4b. 323),

(3) Cowl flaps in the position normally used in an approach to a landing,

(4) Center of gravity in the most unfavorable position permitted for landing,

(5) All engines operating at the take-off power available at such altitude,

(6) The weight equal to maximum landing weight for that altitude.

(7) A climb speed not in excess of $1.4\ V_{s_0}$.

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b–2, 20 F. R. 5304, July 26, 1955]

§ 4b.119–1 Determination of all engine climb (FAA policies which apply to § 4b.119).

(a) *Cruising configuration test, § 4b.-119 (a)*—(1) *Configuration.* This test should be conducted in the configuration that follows:

Weight—Maximum take-off and one lower.
C. G. position—Optional (see § 4b.118–1 (c) (2)).
Wing flaps—Optional.
Landing gear—Retracted.
Engines—Maximum continuous RPM and manifold pressure or full throttle, mixture setting in normal position, carburetor air heat control cold and cowl flaps in FAA hot day cooling position.

(2) *Test procedure and required data.* See § 4b.118–1 for test procedure and required data in connection with climb tests.

(b) *Landing configuration test, § 4b.-119 (b)*—(1) *Configuration.* This test should be conducted in the configuration that follows:

Weight—Maximum landing and one lower.
C. G. position—Optional (see § 4b.118–1 (c) (2)).
Wing flaps—Landing position.
Landing gear—Extended.
Engines—Take-off power and manifold pressure or full throttle, mixture setting in normal position, carburetor air heat control cold and cowl flaps in approach position.

(2) *Test procedures and required data.* See § 4b.118–1 for test procedure and required data in connection with climb tests.

[Supp. 24, 19 F. R. 4453, July 20, 1954]

§ 4b.120 One-engine-inoperative climb.

(a) *Flaps in take-off position; landing gear extended.* The steady rate of climb without ground effect shall not be less than 50 ft./min. at any altitude within the range for which take-off weight is to be specified in the certificate, with:

(1) Wing flaps in the take-off position (see §§ 4b.111 and 4b.323),

(2) Cowl flaps in the position normally used during take-off,

(3) Center of gravity in the most unfavorable position permitted for take-off,

(4) The critical engine inoperative, its propeller windmilling with the propeller control in a position normally used during take-off,

(5) All other engines operating at the take-off power available at such altitude,

(6) The speed equal to the minimum take-off safety speed V_2 (see § 4b.114 (b)),

(7) The weight equal to maximum take-off weight for that altitude,

(8) Landing gear extended.

(b) *Flaps in take-off position; landing gear retracted.* With the landing gear retracted the steady rate of climb in feet per minute shall not be less than $0.035\ V_{s_1}{}^2$ with all other conditions as described in paragraph (a) of this section.

(c) *Flaps in en route position.* The steady rate of climb in feet per minute at any altitude at which the airplane is expected to operate, at any weight within the range of weights to be specified in the airworthiness certificate, shall be determined and shall, at a standard altitude of 5,000 feet and at the maximum take-off weight, be at least

$$\left(0.06-\frac{0.08}{N}\right)V_{s_0}{}^2,$$

where N is the number of engines installed, with:

(1) The landing gear retracted,

(2) Wing flaps in the most favorable position,

(3) Cowl flaps or other means of controlling the engine cooling air supply in the position which provides adequate cooling in the hot-day condition,

(4) Center of gravity in the most unfavorable position,

(5) The critical engine inoperative, its propeller stopped,

(6) All remaining engines operating at the maximum continuous power available at the altitude.

(d) *Flaps in approach position.* The steady rate of climb in feet per minute shall not be less than $0.04\ V_{s_0}{}^2$ at any altitude within the range for which landing weight is to be specified in the certificate, with:

(1) The landing gear retracted,

(2) Wing flaps set in position such that V_{s_1} does not exceed $1.10\ V_{s_0}$,

(3) Cowl flaps in the position normally used during an approach to a landing,

(4) Center of gravity in the most unfavorable position permitted for landing,

(5) The critical engine inoperative, its propeller stopped,

(6) All remaining engines operating at the take-off power available at such altitude,

(7) The weight equal to the maximum landing weight for that altitude.

(8) A climb speed not in excess of $1.5\ V_{s_1}$.

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b–6, 17 F. R. 1089, Feb. 5, 1952; Amdt. 4b–2, 20 F. R. 5304, July 26, 1955]

§ 4b.120–1 Approval of automatic propeller feathering installations for use in establishing flaps in take-off position climb (FAA policies which apply to § 4b.120 (a) and (b)).

The propeller of the inoperative engine may be in the feathered condition during either or both of the landing gear extended or retracted conditions if:

(a) The propeller would be completely feathered at the beginning of these segments of the take-off flight path, or

(b) It can be shown that the network produced by the feathering propeller during the segment is positive using a datum based on feathered propeller drag. (See §§ 4b.10–2, 4b.401–1, and 4b.700–1.)

[Supp. 23, 19 F. R. 1818, Apr. 2, 1954]

§ 4b.120–2 Determination of one engine inoperative climb (FAA policies which apply to § 4b.120).

(a) *Flaps in take-off position; landing gear extended, § 4b.120(a).* Policies outlined in § 4b.116–2(b) (2) will apply.

(b) *Flaps in take-off position; landing gear retracted, § 4b.120 (b).* Policies outlined in § 4b.116–2 (b) (3) will apply.

(c) *Flaps in en route position, § 4b.120 (c)*—(1) *Configuration.* This test should be conducted in the configuration that follows:

Weight—Maximum take-off and one lower.
C. G. position—Optional (see § 4b.118–1 (c) (2)).
Wing flaps—Optional.
Landing gear—Retracted.
Operating engine(s)—Maximum continuous rpm and manifold pressure or full throttle, mixture setting at normal position, carburetor air heat control at cold and cowl flaps in FAA hot day cooling position.
Critical inoperative engine—throttle closed on engine most critical performancewise (see § 4b.118–1 (e) (2)), propeller feathered and cowl flaps in minimum drag position.

(2) *Test procedure and required data.* The airplane should be climbed at the en route climb speed. See § 4b.118–1 for test procedure and required data in connection with climb tests.

(d) *Flaps in approach position § 4b.120 (d)*—(1) *Configuration.* This test should be conducted in the configuration that follows:

Weight—Maximum landing and one lower.
C. G. position—Optional (see § 4b.118–1 (b) (2)).
Wing flaps—Approach position (V_{s_1} must not exceed 1.10 V_{s_0}).
Landing gear—Retracted.
Operating engine(s)—Take-off rpm and manifold pressure or full throttle, mixture setting at normal position, carburetor air heat control at cold and cowl flaps in approach position.
Critical inoperative engine—Throttle closed on engine most critical performancewise (see § 4b.118–1 (e) (2)), propeller feathered and cowl flaps position optional.

(2) *Test procedure and required data.* The airplane should be climbed at the approach climb speed. See § 4b.118–1 for test procedure and required data in connection with climb tests.

[Supp. 24, 19 F.R. 4453, July 20, 1954]

§ 4b.121 Two-engine-inoperative climb.

For airplanes with four or more engines, the steady rate of climb at any altitude at which the airplane is expected to operate, and at any weight within the range of weights to be specified in the Airplane Flight Manual, shall be determined with:

(a) The landing gear retracted,

(b) Wing flaps in the most favorable position,

(c) Cowl flaps or other means of controlling the engine cooling air supply in the position which will provide adequate cooling in the hot-day condition,

(d) Center of gravity in the most unfavorable position,

(e) The two critical engines on one side of the airplane inoperative and their propellers stopped,

(f) All remaining engines operating at the maximum continuous power available at that altitude.

§ 4b.121–1 Determination of two engine inoperative climb (FAA policies which apply to § 4b.121).

(a) *Configuration.* This test should be conducted in the configuration that follows:

Weight—Two optional weights.
C. G. position—Optional (see § 4b.118–1 (c) (2)).
Wing flaps—Optional.
Landing gear—Retracted.
Operating engines—Maximum continuous rpm and manifold pressure or full throttle, mixture setting at normal position, carburetor air heat control at cold and cowl flaps in FAA hot day cooling position.
Critical inoperative engines—Throttles closed on outboard engine most critical performancewise (see § 4b.118–1 (e) (2)), and on adjacent engine, propellers feathered and cowl flaps in minimum drag position.

(b) *Test procedure and required data.* See § 4b.118–1 for test procedure and required data in connection with climb tests.

[Supp. 24, 19 F. R. 4454, July 20, 1954]

§ 4b.122 Determination of the landing distance; general.

The horizontal distance required to land and to come to a complete stop (to a speed of approximately 3 m.p.h. for seaplanes or float planes) from a point at a height of 50 feet above the landing surface shall be determined for a range of weights and altitudes selected by the applicant. In making this determination the following conditions shall apply:

(a) A steady gliding approach shall have been maintained down to the 50-foot altitude with a calibrated air speed of not less than 1.3 V_{s_0}.

(b) The nose of the airplane shall not be depressed, nor the forward thrust increased by application of power after reaching the 50-foot altitude.

(c) At all times during and immediately prior to the landing, the flaps shall be in the landing position, except that after the airplane is on the landing surface and the calibrated air speed has been reduced to not more than 0.9 V_{s_0} the flap position may be changed.

(d) The landing shall be made in such manner that there is no excessive vertical acceleration, no tendency to bounce, nose over, ground loop, porpoise, or water loop, and in such manner that its reproduction shall not require any exceptional degree of skill on the part of the pilot, or exceptionally favorable conditions.

§ 4b.122–1 Downwind landings (FAA policies which apply to § 4b.122).

Downwind landing data will be approved on the following basis to provide for situations where geographic locations and terrain indicate they are desirable, as well as for use with ILS:

(a) *Performance.* In determining the required distances for landing downwind, the data should be substantiated

by actual flight tests. The general methods and procedures should be comparable to those for substantiating landing distances in no wind. The flight tests should be conducted in tailwind velocities up to the maximum velocity for which approval is desired except that the performance tests may be simulated in zero wind as outlined below:

(1) Landings should be demonstrated by approaching and contacting at speeds corresponding to the zero wind speed plus 150 percent of the tailwind velocity for which approval is desired.

(2) In determining the downwind landing distances for the Airplane Flight Manual data, 150 percent of the effect of the reported tailwind velocity should be taken into account. (See § 4b.740–1 (d) (2) (x).) This may, in some cases, permit calculating the required distances without further tests providing sufficiently high speed landings and decelerations were made in the original type tests. However, except in the cases outlined in paragraph (d) of this section, actual landings should be made under the conditions described in paragraph (b) of this section to check the flight and ground handling characteristics.

(b) *Controllability.* Landings should be made in steady downwind velocities equal to 1.5 times the maximum velocity for which approval is desired to check the controllability at the higher ground speed with correspondingly reduced aerodynamic control forces, dynamic balance of landing gear, nose gear shimmy or vibration, etc. Also actual approaches should be demonstrated under the above wind conditions at an approach angle corresponding to the maximum ILS beam angle (3° 18′) to determine the minimum altitude on the glide path from which the airplane can be readily flared for landing.

(c) *Brakes.* At present it is believed that for calculated landing distance based on actual airplane deceleration tests, the existing brake capacity requirements are sufficient to cover landings in downwind velocities of 10 m. p. h. measured at 50 feet. However, in wind velocities above 10 m. p. h. and in unusual cases or special types of operation, additional tests or substantiation of the adequacy of the brakes may be necessary and a revision to the braking system may be required. In determining the landing distances under paragraph (a) of this section, normal braking as outlined in § 4b.123 "Landplanes" should not be exceeded.

(d) *Tolerances.* (1) With regard to performance tests outlined in paragraph (a) of this section, approval will be given for calculated landing distances for reported tailwind velocities up to 10 mph. measured at 50 feet height without camera tests additional to those required for approval of the no wind data.

(2) With regard to controllability tests outlined in paragraph (b) of this section, approval will be given for reported downwind velocities up to 10 mph. measured at 50 feet without additional flight tests.

[Supp. 23, 19 F. R. 1818, Apr. 2, 1954, as amended by Supp. 25, 20 F. R. 2277, Apr. 8, 1955]

§ 4b.122–2 Determination of the landing distances (FAA policies which apply to § 4b.122).

(a) When a particular airplane cannot comply with that part of § 4b.122(d) regarding exceptional degree of skill on the part of the pilot in landing from the 50-foot height with power off, compliance with the regulation should be shown by applying sufficient power during the approach to permit satisfactory landing.

(b) In the tests required by §§ 4b.123 through 4b.124 generally one set of data at one altitude should be sufficient to determine the landing distances for altitudes from sea level to 8,000 feet. If a greater range of airport altitudes is desired, the tests should be conducted at two or more altitudes.

[Supp. 23, 19 F. R. 4454, July 20, 1954]

§ 4b.122–3 Reverse thrust used in determination of landing distance (FAA policies which apply to § 4b.122).

The policies outlined in § 4b.402–1(1) will apply.

[Supp. 25, 20 F. R. 2277, Apr. 8, 1955]

§ 4b.122–4 Determination of the landing distance with an antiskid device installed (FAA policies which apply to § 4b.122).

The policies outlined in § 4b.337–4 will apply.

[Supp. 28, 21 F. R. 2558, Apr. 19, 1956]

§ 4b.123 Landplanes.

The landing distance referred to in § 4b.122 shall be determined on a dry, hard-surfaced runway in accordance with the following:

(a) The operating pressures on the braking system shall not be in excess of those approved by the manufacturer of the brakes.

(b) The brakes shall not be used in such manner as to produce excessive wear of brakes or tires.

(c) Means other than wheel brakes may be used in determining the landing distance: *Provided*, That:

(1) Exceptional skill is not required to control the airplane,

(2) The manner of their employment is such that consistent results could be expected under normal service, and

(3) They are regarded as reliable.

§ 4b.123–1 Excessive wear of brakes or tires (FAA interpretation which applies to § 4b.123(b)).

"Excessive wear" is interpreted as skidding of a tire or excessive heating of the brakes which requires replacement during a series of five official test landings.

[Supp. 24, 19 F. R. 4454, July 20, 1954]

§ 4b.123–2 Determination of the landing distances (landplanes) (FAA policies which apply to § 4b.123).

(a) *Landing tests.* The landing tests should be demonstrated in accordance with the following provisions:

(1) Landings should be made over an imaginary 50-foot obstacle at the maximum landing weight and at a lower weight at one altitude.

(2) During the landing demonstrations, the glide path should be established by the pilot as set forth in § 4b.122. The forward thrust should not be increased beyond the 50-foot obstacle. The ground roll should lie as close to a rectilinear path as possible including the airplane stop point. During each demonstration landing, the airplane should be brought to a complete stop.

(3) Instrumentation: (i) Instrumentation should include a means to record the airplane's glide path relative to the ground and the ground roll against time in a manner to determine the horizontal and vertical distance-time histories.

(ii) A means should be provided to measure the wind velocity and direction, pressure altitude, and ambient air temperature. The wind measurement should be made at the height of 6 feet above the runway surface. If wind effect on runway lengths is shown in the Airplane Flight Manual (see § 4b.740–1 (d) (2) (x)), the manual data should be based on reported wind velocities for a 50-foot tower height. Figure 2 should be used to calculate the wind velocity at the 50-foot height from the wind velocity measured at 6 feet above the runway surface.

(iii) The ground roll distance from contact to full stop should be established by observers if it is difficult to establish the exact contact point by graphical means.

(4) A special tolerance of not greater than ±2 percent of the maximum landing weight is allowable for the landing distance tests.

(b) *Configuration.* The landing tests should be demonstrated in the configuration that follows:

Weight—Maximum landing and one lower.
C. G. position—Most forward for braked landings.
Wing flaps—Landing position.
Landing gear—Extended.

(c) *Test procedure and required data.* (1) Three landings should be conducted at each weight with the airplane stabilized in a glide at a calibrated air speed of not less than 1.3 V_{s_0} approximately 1,000 feet (longitudinally) prior to reaching an altitude of 50 feet. Forward thrust should not be increased beyond the 50-foot obstacle; braking should not exceed manufacturer's approved maximum brake pressure and should be applied in such a manner as not to produce excessive wear of brakes and tires as evidenced by five measured landings. If more than one flap setting is to be used for landing, additional tests should be conducted to cover the flap range. (See § 4b.118–1 (d) (2).)

(2) The airplane path relative to the runway should be recorded against time in a manner to determine the horizontal and vertical distance-time history. In addition, the following data should be recorded:

Pressure altitude.
Ambient air temperature.
Airplane gross weight.
Rpm.
Manifold pressure.
Torque pressure.
Carburetor air temperature.
Mixture setting.
Cowl flap position.
Wing flap position.
Slope of field.
Direction of landing run.

(3) Wind direction and wind velocity should be recorded adjacent to the runway at a height of 6 feet.

[Supp. 24, 19 F.R. 4454, July 20, 1954, as amended by Supp. 34, 22 F.R. 6963, Aug. 29, 1957]

§ 4b.123–3 Reverse thrust used in landing distance—landplanes (FAA policies which apply to § 4b.123(c)).

The policies outlined in § 4b.402–1(1) will apply.

[Supp. 25, 20 F. R. 2277, Apr. 8, 1955]

§ 4b.123–4 Procedure in determination of landing distance with an antiskid device installed (FAA policies which apply to § 4b.123).

The policies outlined in § 4b.337–4 will apply.

[Supp. 28, 21 F. R. 2558, Apr. 19, 1956]

§ 4b.124 Seaplanes or float planes.

The landing distance referred to in § 4b.122 shall be determined on smooth water.

§ 4b.124–1 Determination of the landing distance (seaplanes) (FAA policies which apply to § 4b.124).

Policies outlined in §§ 4b.122–2 and 4b.123–2 will apply.

[Supp. 24, 19 F. R. 4454, July 20, 1954]

§ 4b.125 Skiplanes.

The landing distance referred to in § 4b.122 shall be determined on smooth, dry snow.

§ 4b.125–1 Determination of the landing distances (skiplanes) (FAA policies which apply to § 4b.125).

Policies outlined in §§ 4b.122–2 and 4b.123–2 will apply.

[Supp. 24, 19 F. R. 4454, July 20, 1954]

CONTROLLABILITY

§ 4b.130 Controllability; general.

(a) The airplane shall be safely controllable and maneuverable during takeoff, climb level flight, descent, and landing.

(b) It shall be possible to make a smooth transition from one flight condition to another, including turns and slips, without requiring an exceptional degree of skill, alertness, or strength on the part of the pilot and without danger of exceeding the limit load factor under all conditions of operation probable for the type, including those conditions normally encountered in the event of sudden failure of any engine.

(c) Compliance with the "strength of pilots" limits in paragraph (b) of this section need not be demonstrated unless the condition is found to be marginal. In the latter case, they shall not exceed the following pilot control force limits, expressed in pounds:

	Pitch	Roll	Yaw
(1) For temporary application	75	60	180
(2) For prolonged application	10	5	20

Pitch and roll forces shall be measured as applied to the control wheel.

(d) For the purpose of complying with the temporary control force limitations of paragraph (c) of this section, the airplane shall be operated in accordance with approved operating procedure or conventional operating practice including being as nearly trimmed as possible at the prior steady flight condition, except that in the case of takeoff the airplane shall be trimmed in accordance with approved operating procedures.

(e) For the purpose of complying with the prolonged control force limitations of paragraph (c) of this section, the airplane shall be as nearly trimmed as possible.

[15 F.R., 3543, June 8, 1950, as amended by Amdt. 4b–12, 27 F.R. 2989, Mar. 30, 1962]

§ 4b.130–1 Procedure for demonstrating controllability qualities (FAA policies which apply to § 4b.130).

The general controllability and maneuverability qualities of the airplane should be observed and noted throughout the flight test program. The amount of force required to be exerted on the controls in conducting such maneuvers specified in § 4b.130 should also be noted.

[Supp. 24, 19 F. R. 4454, July 20, 1954]

§ 4b.131 Longitudinal control.

(a) It shall be possible at all speeds between the trim speed prescribed in § 4b.112(c)(1) and V_{s_1} to pitch the nose downward so that a prompt recovery to this selected trim speed can be made with the following combination of airplane configurations:

(1) The airplane trimmed at the trim speed prescribed in § 4b.112(c)(1),

(2) The landing gear extended,

(3) The wing flaps in a retracted, and in an extended position.

(4) Power off, and maximum continuous power on all engines.

(b) During each of the following controllability demonstrations, a change in the trim control shall not be required. In addition, exertion of more than 50 pounds control force, representative of the maximum temporary force which can readily be applied by one hand, shall not be required. Each maneuver shall be performed with the landing gear extended.

(1) With power off, flaps retracted, and the airplane trimmed at 1.4 V_{s_1}, the flaps shall be extended as rapidly as possible while maintaining the air speed approximately 40 percent above the stalling speed prevailing at any instant throughout the maneuver.

(2) The maneuver of subparagraph (1) of this paragraph shall be repeated, except that it shall be started with flaps extended and the airplane trimmed at 1.4 V_{s_1}, after which the flaps shall be retracted as rapidly as possible.

(3) The maneuver of subparagraph (2) of this paragraph shall be repeated, except that take-off power shall be used.

(4) With power off, flaps retracted, and the airplane trimmed at 1.4 V_{s_1}, take-off power shall be applied quickly while maintaining the same air speed.

(5) The maneuver of subparagraph (4) of this paragraph shall be repeated, except that the flaps shall be extended.

(6) With power off, flaps extended, and the airplane trimmed at 1.4 V_{s_1}, air speeds within the range of 1.1 V_{s_1} to 1.7 V_{s_1} or to V_{FE}, whichever of the two is the lesser, shall be obtained and maintained.

(c) It shall be possible without the use of exceptional piloting skill to prevent loss of altitude when wing flap retraction from any position is initiated during steady straight level flight at a speed equal to 1.1 V_{s_1} with simultaneous application of not more than maximum continuous power, with the landing gear extended, and with the airplane weight equal to the maximum sea level landing weight. (See also § 4b.323.)

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b-6, 17 F.R. 1089, Feb. 5, 1952; Amdt. 4b-11, 24 F.R. 7068, Sept. 1, 1959; Amdt. 4b-12, 27 F.R. 2989, Mar. 30, 1962]

§ 4b.131-1 Procedure for demonstrating longitudinal control (FAA policies which apply to § 4b.131).

The flight tests specified in paragraphs (a) through (h) of this section should be made in demonstrating compliance with § 4b.131. These tests may be conducted at an optional altitude (see § 4b.100-3(c)). Where applicable, the following conditions should be maintained on the engines: propellers in low pitch; throttles—closed except as noted; cowl flaps—appropriate for the flight condition.

(a) *Longitudinal control recovery § 4b.131(a).* The test specified by this requirement should be demonstrated with power off and also with maximum continuous power.

(1) *Configuration.* This test should be conducted in the configuration that follows:

(i) Maximum take-off weight, flaps and landing gear retracted, C. G. position—most aft.

(ii) Maximum landing weight, flaps extended in the maximum landing position; landing gear extended, C. G. position—most aft.

(The flap extended portion of this test may be combined with tests under paragraph (g) of this section.)

(2) *Test procedure and required data.* The airplane should be trimmed at the speed prescribed in § 4b.112(c)(1); the nose should be pitched downward starting from any speed between that prescribed in § 4b.112(c)(1), and V_{s_1}. The rate of increase in airspeed should be satisfactory for prompt acceleration to the trim speed prescribed in § 4b.112 (c)(1). The following data should be recorded:

Weight.
C.G. position.
Wing flap position.
Landing gear position.
Engines, r.p.m. and manifold pressure.
Pressure altitude.
Ambient air temperature.
Trim speed.
Lowest speed from which pitch is satisfactory.
Altitude lost to regain trim speed.

(b) *Longitudinal control, flap extension, § 4b.131 (b) (1)—(1) Configuration.* This test should be conducted in the configuration that follows:

Weight—Maximum landing.
C. G. position—Most forward and most aft.
Wing flaps—Retracted.
Landing gear—Extended.
Engines—Power off.

(2) *Test procedure and required data.* The airplane should be trimmed at a speed of 1.4 V_{s_1}; the flaps should be extended to the maximum landing position as rapidly as possible. During this maneuver, it should be possible to control the airplane with one hand for a reasonable time without the necessity of changing the trim control. The following data should be recorded:

Weight.
C. G. position.
Wing flap position.
Landing gear position.
Engines, rpm and manifold pressure.
Pressure altitude.
Ambient air temperature.
Trim speed at 1.4 V_{s_1}.

(c) *Longitudinal control, flap retraction § 4b.131 (b) (2)—(1) Configuration. This test should be conducted in the configuration that follows:*

Weight—Maximum landing.
C. G. position—Most forward and most aft.
Wing flaps—Maximum landing position.
Landing gear—Extended.
Engines—Power off.

(2) *Test procedures and required data.* The airplane should be trimmed at a speed of 1.4 V_{s_1}; the flaps should be retracted as rapidly as possible. During this maneuver, it should be possible to control the airplane with one hand for a reasonable time without the necessity of changing the trim control. The same data as are specified in paragraph (b) (2) of this section should be recorded.

(d) *Longitudinal control, flap retraction § 4b.131 (b) (3).* The maneuver shown in paragraph (c) (2) of this section should be repeated with take-off power on all engines.

(e) *Longitudinal control, power application, § 4b.131 (b) (4)—(1) Configuration.* This test should be conducted in the configuration that follows:

Weight—Maximum landing.
C. G. position—Most forward and most aft.
Wing flaps—Retracted.
Landing gear—Extended.
Engines—Power off.

(2) *Test procedure and required data.* The airplane should be trimmed at a speed of 1.4 V_{s_1}; take-off power should be applied quickly without changing the air speed. During this maneuver, it should be possible to control the airplane with one hand for a reasonable time without the necessity of changing the trim control. The same data as are specified in paragraph (b) (2) of this section should be recorded.

(f) *Longitudinal control, power application, § 4b.131 (b) (5)—(1) Configuration.* This test should be conducted in the configuration that follows:

Weight—Maximum landing.
C. G. position—Most forward and most aft.
Wing flaps—Maximum landing position.
Landing gear—Extended.
Engines—Power off.

(2) *Test procedure and required data.* The airplane should be trimmed at a speed of 1.4 V_{s_1}; take-off power should be applied quickly without changing the air speed. During this maneuver, it should be possible to control the airplane with one hand for a reasonable time without the necessity of changing the trim control. The same data as are specified in paragraph (b) (2) of this section should be recorded.

(g) *Longitudinal control, air speed variation, § 4b.131 (b) (6)—(1) Configuration.* This test should be conducted in the configuration that follows:

Weight—Maximum landing.
C. G. position—Most forward.
Wing flaps—Maximum landing position.
Landing gear—Extended.
Engines—Power off.

(2) *Test procedure and required data.* The airplane should be trimmed at a speed of 1.4 V_{s_1} and it should be possible to control the airplane with one hand for a reasonable time without the necessity of changing the trim control when

(i) The speed is reduced to 1.1 V_{s_1}.

(ii) The speed is increased to 1.7 V_{s_1} or to the placard flap speed.

In addition to the data specified in (b) (2) of this section, the recorded data should also include the air speeds at 1.1 V_{s_1} and 1.7 V_{s_1} (or V_{FE} whichever is the lesser).

(h) *Longitudinal control, flap retraction and power application, § 4b.131 (c)—(1) Configuration.* This test should be conducted in the configuration that follows:

Weight—Maximum landing.
C. G. position—Most forward and most aft.
Wing flaps—Maximum landing position.
Landing gear—Extended.
Engines—Power noted.

(2) *Test procedure and required data.* The airplane should be maintained in a steady horizontal flight at a speed of 1.1 V_{s_1}; the flaps should be retracted from the maximum landing position with simultaneous application of not more than maximum continuous power. During this maneuver, it should be possible to prevent a loss of altitude without

the use of exceptional piloting skill. The following data should be recorded:

Weight.
C. G. position.
Wing flap position.
Landing gear position.
Engines, rpm and manifold pressure before and after tests.
Pressure altitude.
Ambient air temperature.
Air speed at 1.1 V_{s_1}.

[Supp. 24, 19 F.R. 4454, July 20, 1954, as amended by Supp. 42, 24 F.R. 7072, Sept. 1, 1959]

§ 4b.132 Directional and lateral control.

(a) *Directional control; general.* It shall be possible, while holding the wings approximately level, to execute reasonably sudden changes in heading in either direction without encountering dangerous characteristics. Heading changes up to 15° shall be demonstrated, except that the heading change at which the rudder pedal force is 180 pounds need not be exceeded. The control shall be demonstrated at a speed equal to 1.4 V_{s_1}, under the following conditions:

(1) The critical engine inoperative and its propeller in the minimum drag position,

(2) Power required for level flight at 1.4 V_{s_1}, but not greater than maximum continuous power,

(3) Most unfavorable center of gravity position,

(4) Landing gear retracted.

(5) Wing flaps in the approach position,

(6) Maximum landing weight.

(b) *Directional control; four or more engines.* Airplanes with four or more engines shall comply with paragraph (a) of this section, except that:

(1) The two critical engines shall be inoperative, their propellers in the minimum drag position,

(2) The center of gravity shall be in the most forward position,

(3) The wing flaps shall be in the most favorable climb position.

(c) *Lateral control; general.* It shall be possible to execute 20° banked turns with and against the inoperative engine from steady flight at a speed equal to 1.4 V_{s_1} with:

(1) The critical engine inoperative and its propeller in the minimum drag position,

(2) Maximum continuous power on the operating engines,

(3) Most unfavorable center of gravity position,

(4) Landing gear retracted and extended,

(5) Wing flaps in the most favorable climb position,

(6) Maximum take-off weight.

(d) *Lateral control; four or more engines.* It shall be possible to execute 20° banked turns with and against the inoperative engines from steady flight at a speed equal to 1.4 V_{s_1} with maximum continuous power and with the airplane in the configuration prescribed by paragraph (b) of this section.

(e) *Lateral control; all engines operating.* Roll response shall be rapid and of sufficient magnitude to perform normal maneuvers, such as recovery from upsets produced by gusts and the initiation of evasive maneuvers. In sideslips up to sideslip angles which might be required in normal operation, sufficient excess lateral control shall remain to perform a limited amount of maneuvering and to correct for gusts. Lateral control shall be sufficient at all speeds up to V_{FC}/M_{FC} to provide a peak roll rate necessary for safety without requiring excessive control forces or excessive control travel.

[15 F.R. 3543, June 8, 1950, as amended. 15 F.R. 4171, June 29, 1950; Amdt. 4b-8, 23 F.R. 2590, Apr. 19, 1958; Amdt. 4b-12, 27 F.R. 2989, Mar. 30, 1962]

§ 4b.132-1 Procedure for demonstrating directional and lateral control (FAA policies which apply to § 4b.132).

(a) *Appropriate instrumentation.* When conducting directional and lateral control tests, the airplane should contain appropriate instrumentation in order to obtain the following data:

(1) *Section 4b.132 (a) and (b).* Instrumentation to indicate airplane heading angle and rudder force.

(2) *Section 4b.132 (c) and (d).* Instrumentation to measure bank angle.

(b) *General test program*—(1) *Directional control; general, § 4b.132 (a)*—(i) *Configuration.* This test should be conducted in the configuration that follows:

Weight—Maximum landing.
C. G. position—Most aft.
Wing flaps—Approach position.
Landing gear—Retracted.
Cowl flaps—Appropriate for flight condition.
Operating engine(s)—Power to maintain level flight at 1.4 V_{s_1} but not more than maximum continuous power.
Critical inoperative engine—Propeller feathered on engine most critical for controllability.

(ii) *Test procedure and required data.* The airplane should be trimmed as indicated above at any optional altitude (see § 4b.100-3 (c)). Reasonable sudden changes in heading to the left and right, using ailerons to maintain approximate level flight, should be made demonstrating a change of 15°, or the angle at which a dangerous condition is encountered, or at which 180 lbs. rudder force is required. The airplane should be satisfactorily controllable during this maneuver. The following data should be recorded:

Weight.
C. G. position.
Wing flap position.
Landing gear position.
Operating engine(s), rpm and manifold pressure.
Position of critical inoperative engine and its propeller.
Pressure altitude.
Ambient air temperature.
Trim speed at 1.4 V_{s_1}.
Rudder force at maximum deflection.

(2) *Directional control; four or more engines, § 4b.132 (b)*—(i) *Configuration.* This test should be conducted in the configuration that follows:

Weight—Maximum landing.
C. G. position—Most forward.
Wing flaps—Climb position.
Landing gear—Retracted.
Operating engines—Power required for level flight at 1.4 V_{s_1} but not more than maximum continuous power.
Critical inoperative engines—Propellers feathered on outboard engine most critical for controllability and on adjacent engine.

(ii) *Test procedure and required data.* The test procedure shown in subparagraph (1) (ii) of this paragraph should be repeated with two critical engines inoperative. In addition to the data specified in subparagraph (1) (ii) of this paragraph, the position of the critical inoperative engines and the propeller configuration should also be recorded.

(3) *Lateral control; general, § 4b.132 (c)*—(i) *Configuration.* This test should be conducted in the configuration that follows:

Weight—Maximum take-off.
C. G. position—Most aft.
Wing flaps—Climb position.
Landing gear—Retracted and extended.
Operating engine(s)—Maximum continuous power.
Critical inoperative engine—Throttle closed on engine most critical for controllability, propeller feathered.

(ii) *Test procedure and required data.* Banked turns of 20° should be demonstrated with and against the inoperative engine from a steady climb at 1.4 V_{s_1}. The following data should be recorded:

Weight.
C. G. position.
Wing flap position.
Landing gear position.
Engine(s), rpm and manifold pressure.
Position of critical inoperative engine and its propeller.
Pressure altitude.
Ambient air temperature.
Trim speed at 1.4 V_{s_1}.
Rudder force at maximum deflection.
Aileron force at maximum deflection.

(4) *Lateral control; four or more engines, § 4b.132 (d)*—(i) *Configuration* This test should be conducted in the configuration that follows:

Weight—Maximum landing.
C. G. position—Most forward.
Wing flaps—Climb position.
Landing gear—Retracted.
Operating engines—Power required for level flight at 1.4 V_{s_1} but not more than maximum continuous power.
Critical inoperative engines—Propellers feathered on outboard engine most critical for controllability and on adjacent engine.

(ii) *Test procedure and required data.* Banked turns of 20° should be demonstrated with and against inoperative engines from steady flight at 1.4 V_{s_1}. The same data as are specified in subparagraph (3) (ii) of this paragraph should be recorded.

[Supp. 24, 19 F. R. 4455, July 20, 1954]

§ 4b.133 Minimum control speed, V_{MC}.

(a) A minimum speed shall be determined under the conditions specified in this paragraph, so that when the critical engine is suddenly made inoperative at that speed it shall be possible to recover control of the airplane, with the engine still inoperative, and maintain it in straight flight at that speed, either with zero yaw or, at the option of the applicant, with an angle of bank not to excess of 5°. Such speed shall not exceed 1.2 V_{s_1} with:

(1) Take-off or maximum available power on all engines,

(2) Rearmost center of gravity,

(3) Flaps in take-off position,

(4) Landing gear retracted,

(5) Cowl flaps in the position normally used during take-off,

(6) Maximum sea level take-off weight, or such lesser weight as may be necessary to demonstrate V_{MC}.

(7) The airplane trimmed for take-off.

(8) The propeller of the inoperative engine windmilling, except that a different position of the propeller shall be acceptable if the specific design of the propeller control makes it more logical to assume the different position.

(9) The airplane airborne and the ground effect negligible.

(b) In demonstrating the minimum speed of paragraph (a) of this section, the rudder force required to maintain control shall not exceed 180 pounds, and it shall not be necessary to throttle the remaining engines.

(c) During recovery of the maneuver of paragraph (a) of this section the airplane shall not assume any dangerous attitude, nor shall it require exceptional skill, strength, or alertness on the part of the pilot to prevent a change of heading in excess of 20° before recovery is complete.

[15 F. R. 3543, June 6, 1950, as amended by Amdt. 4b-6, 17 F. R. 1089, Feb. 5, 1952]

NOTE: Interpretation No. 1 (17 F. R. 2112, Mar. 12, 1952), adopted by the Civil Aeronautics Board, Mar. 7, 1952, provides as follows:

(1) The Board interprets and construes subparagraph (8) of § 4b.133 (a) as requiring the Administrator to accept for the purposes of § 4b.133 a value for the one-engine-inoperative minimum control speed which has been established in accordance with the provisions of that section with the propeller of the inoperative engine feathered: *Provided*, That the airplane involved is equipped with an automatic feathering device acceptable to the Administrator under § 4b.10 for demonstrating compliance with the take-off path and climb requirement of §§ 4b.116 and 4b.120 (a) and (b).

CROSS REFERENCE: For Special Civil Air Regulation applicable to turbine-powered transport category airplanes of current design, in lieu of the requirements contained in § 4b.133, see SR-422B, *supra*.

§ 4b.133-1 Determination of the minimum control speed, V_{MC} (FAA policies which apply to § 4b.133).

(a) When demonstrating the minimum control speed, the applicant may choose one of three basic methods dependent on the inherent characteristics of the airplane or a combination of the methods is acceptable provided that the combination chosen does not allow the aircraft to exceed any of the limtiing factors specified in § 4b.133. These methods are:

(1) With wings level and 180 pounds rudder force, or full rudder travel causing airplane to deviate from a constant heading, or airplane stall.

(2) With the wing on the engine operating side lowered 5° and 180 pounds rudder force, or full rudder travel causing airplane to deviate from a constant heading, or airplane stall.

(3) At 0° yaw and full rudder travel causing airplane to deviate from 0° yaw, or 180 pounds rudder force, or airplane stall.

(b) When it has been found that the aircraft is limited by the 180 pounds rudder force in any of the methods in paragraph (a) of this section, a plot of force vs. air speed should be made through a suitable range of speeds to substantiate the speed chosen at V_{MC}.

(c) Generally speaking, in aircraft equipped with right-hand rotating propellers, the left-hand outboard engine is the most critical when inoperative from the standpoint of control. This condition should be substantiated, however, by a comparative test with both the right and then the left outboard engines inoperative, measuring the force necessary to hold the airplane within the limits as specified in § 4b.133 at or slightly above the minimum control speed. When conducting this test it is imperative to hold all remaining factors equal so that a true comparison may be accomplished.

(d) All testing should be accomplished at the appropriate weights and powers for the range of approval desired. The minimum control speed should be determined for each take-off flap position selected for approval if the take-off flap is made variable with altitude.

(e) Civil Air Regulation Part 4b, Interpretation No. 1, interprets § 4b.133 (a) (8) as requiring the establishment of one engine inoperative minimum control speed with the propeller of the inoperative engine feathered providing that the airplane is equipped with an automatic feathering device acceptable to the Administrator under § 4b.10 for demonstrating compliance with the take-off path and climb requirements of §§ 4b.116 and 4b.120 (a) and (b). In such cases where the applicant chooses to demonstrate V_{MC} with the propeller feathered, the value of V_{MC} with the propeller windmilling should also be obtained and included in the Airplane Flight Manual.

(f) Configuration: This test should be conducted in the configuration that follows:

Weight—Maximum take-off. (If stall occurs prior to reaching V_{MC}, applicant may choose to demonstrate a lower V_{MC} at a reduced weight.)
C. G.—Most aft.
Wing flaps—Take-off position.
Landing gear—Retracted.
Operating engine(s)—Take-off rpm and manifold pressure or full throttle, cowl flaps in take-off position.
Inoperative engine—Throttle closed, propeller windmilling or any other logical position, cowl flaps in take-off position.

(g) Test procedure and required data: After establishing the critical inoperative engine and the choice of method for demonstration, the tests for establishing the minimum control speed may be conducted. Using the configuration specified in § 4b.133, all engines should be adjusted for take-off power and a series of engine cuts made by moving the mixture control of the critical engine in idle cut-off position at consecutively lower air speeds until one of the limiting factors specified in § 4b.133 is experienced. When the minimum control speed is determined, a minimum of five demonstrations should be made to provide adequate proof that the chosen value meets the requirement. The following data should be recorded:

Pressure altitude.
Ambient air temperature.
Indicated air speed.
Engines, rpm and manifold pressure.
Torque pressure.
Carburetor air temperature.
Rudder force.
Bank angle.
Gyro direction indicator.
Yaw—If method is chosen where loss of airplane's ability to maintain 0° yaw is limiting factor specified in (a) (3) of this section.

[Supp. 24, 19 F. R. 4456, July 20, 1954]

TRIM

§ 4b.140 General.

The means used for trimming the airplane shall be such that after being trimmed and without further pressure upon, or movement of, either the primary control or its corresponding trim control by the pilot or the automatic pilot, the airplane shall comply with the trim requirements of §§ 4b.141 through 4b.144.

§ 4b.140-1 General trim qualities (FAA policies which apply to § 4b.140).

It should be possible to trim the airplane completely for any flight condition which is reasonable to assume will be maintained steadily for any appreciable time. Compliance for unsymmetrical power should be demonstrated with "wings level" or "zero yaw" when a yawmeter is installed as a part of the required equipment.

[Supp. 24, 19 F. R. 4456, July 20, 1954]

§ 4b.141 Lateral and directional trim.

The airplane shall maintain lateral and directional trim under the most adverse lateral displacement of the center of gravity within the relevant operating limitations, under all normally expected conditions of operation, including operation at any speed from 1.4 V_{s_1} to V_{MO}/M_{MO}.

[Amdt. 4b-2, 20 F.R. 5305, July 26, 1955; Amdt. 4b-12, 27 F.R. 2989, Mar. 30, 1962]

§ 4b.141-1 Procedure for demonstrating lateral and directional trim (FAA policies which apply to § 4b.141).

(a) *Configuration.* This test should be conducted in the configuration that follows:

Weight—Maximum take-off and maximum landing.
C. G. position—Most forward and most aft with greatest lateral variation in useful load. Asymmetrical fuel loading should be considered.
Wing flaps—Retracted and maximum landing position.
Landing gear—Retracted and extended.
Engines—Power required for level flight.
Cowl flaps—Appropriate for flight condition.

(b) *Test procedure and required data.* It should be possible to maintain hands-off lateral and directional trim when demonstrating compliance with § 4b.141. The following data should be recorded:

Weight.
C. G. position.
Wing flap position.
Landing gear position.
Engines, rpm and manifold pressure.
Pressure altitude.
Ambient air temperature.
Trim speed at 1.4 V_{s_1}.

(Additional lateral and directional trim should be demonstrated in other configurations in conjunction with tests in § 4b.150.)

[Supp. 24, 19 F.R. 4456, July 20, 1954, as amended by Supp. 32, 22 F.R. 6963, Aug. 29, 1957]

§ 4b.142 **Longitudinal trim.**

The airplane shall maintain longitudinal trim under the following conditions:

(a) During a climb with maximum continuous power at a speed not in excess of 1.4 V_{s_1} with the landing gear retracted and the wing flaps both retracted and in the take-off position,

(b) During a glide with power off at a speed not in excess of 1.4 V_{s_1} with the landing gear extended and the wing flaps both retracted and extended, with the forward center of gravity position approved for landing with the maximum landing weight, and with the most forward center of gravity position approved for landing regardless of weight,

(c) During level flight at any speed from 1.4 V_s to V_{MO}/M_{MO} with the landing gear and wing flaps retracted, and from 1.4 V_{s_1} to V_{LE} with the landing gear extended.

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b-2, 20 F.R. 5305, July 26, 1955; Amdt. 4b-12, 27 F.R. 2989, Mar. 30, 1962]

§ 4b.142-1 **Procedure for demonstrating longitudinal trim (FAA policies which apply to § 4b.142).**

(a) *Longitudinal trim during climb, § 4b.142(a)*—(1) *Configuration.* This test should be conducted in the configuration that follows:

Weight—Maximum take-off.
C. G. position—Most forward.
Wing flaps—Retracted and take-off position.
Landing gear—Retracted.
Engines—Maximum continuous power.
Cowl flaps—Optional.

(2) *Test procedure and required data.* It should be possible to maintain hands-off longitudinal trim at a speed not in excess of 1.4 V_{s_1} with the wing flaps retracted and in the take-off position. The same data specified in § 4b.141-1 (b) should be recorded.

(b) *Longitudinal trim during glide, § 4b.142 (b)*—(1) *Configuration.* This test should be conducted in the configurations that follow:

Weight—Maximum landing.
C. G. position—Most forward for maximum landing weight.
Wing flaps—Retracted and maximum landing position.
Landing gear—Extended.
Engines—Power off, propellers windmilling.

(2) *Test procedure and required data.* It should be possible to maintain hands-off longitudinal trim at a speed not in excess of 1.4 V_{s_1} with the wing flaps retracted and extended. This test should be repeated with the most forward C. G. position for landing regardless of weight. The same data specified in § 4b.141-1 (b) should be recorded.

(c) *Longitudinal trim during level flight, § 4b.142 (c)*—(1) *Configuration.* This test should be conducted in the configuration that follows:

Weight—maximum takeoff.
C. G. position—most forward and most aft.
Wing flaps—retracted.
Landing gear—retracted and extended.
Engines—power required for level flight.
Cowl flaps—appropriate for flight condition.

(2) *Test procedure and required data.* It should be possible to maintain hands-off longitudinal trim when demonstrating compliance with § 4b.142 (c).

[Supp. 24, 19 F.R. 4456, July 20, 1954, as amended by Supp. 34, 22 F.R. 6963, Aug. 29, 1957]

§ 4b.143 **Longitudinal, directional, and lateral trim.**

(a) The airplane shall maintain longitudinal, directional, and lateral trim at a speed equal to 1.4 V_{s_1} during climbing flight with the critical engine inoperative, with

(1) The remaining engine(s) operating at maximum continuous power,
(2) Landing gear retracted,
(3) Wing flaps retracted.

(b) In demonstrating compliance with the lateral trim requirement of paragraph (a) of this section, the angle of bank of the airplane shall not be in excess of 5 degrees.

[Amdt. 4b-6, 17 F. R. 1089, Feb. 5, 1952]

§ 4b.143-1 **Procedure for demonstrating longitudinal, directional, and lateral trim (FAA policies which apply to § 4b.143).**

(a) *Configuration.* This test should be conducted in the configuration that follows:

Weight—Maximum take-off.
C. G. position—Most forward.
Wing flaps—Retracted.
Landing gear—Retracted.
Cowl flaps—Appropriate for flight condition.
Operating engine(s)—Maximum continuous power.
Critical inoperative engine—Throttle closed on engine most critical for trim, propeller feathered.

(b) *Test procedure and required data.* It should be possible to maintain hands-off longitudinal, directional, and lateral trim during climb at a speed of 1.4 V_{s_1}. In addition to the data specified in § 4b.141-1 (b), the position of the critical inoperative engine and its corresponding propeller should be recorded.

[Supp. 24, 19 F. R. 4457, July 20, 1954]

§ 4b.144 **Trim for airplanes with four or more engines.**

The airplane shall maintain trim in rectilinear flight at the climb speed, configuration, and power used in establishing the rates of climb in § 4b.121, with the most unfavorable center of gravity position, and at the weight at which the two-engine-inoperative climb is equal to at least 0.01 $V_{s_0}^2$ at an altitude of 5,000 feet.

§ 4b.144-1 **Procedure for demonstrating trim for airplanes with four or more engines (FAA policies which apply to § 4b.144).**

(a) *Configuration.* This test should be conducted in the configuration that follows:

Weight—At which climb is equal to at least 0.01 $V_{s_0}^2$ at an altitude of 5,000 feet.
C. G. position—Most forward.
Wing flaps—Optional.
Landing gear—Retracted.
Cowl flaps—Appropriate for flight condition.
Operating engines—Maximum continuous power.
Inoperative engines—Throttles closed on outboard engine most critical for trim and on adjacent engine, propellers feathered.

(b) *Test procedure and required data.* It should be possible to maintain hands-off longitudinal, lateral, and directional trim at the same air speed used in demonstrating the two-engine-inoperative climb (see § 4b.121). The following data should be recorded:

Weight.
C. G. position.
Wing flap position.
Landing gear position.
Operating engines, rpm, manifold pressure, and cowl flap position.
Position of critical inoperative engines and their propellers.
Pressure altitude.
Ambient air temperature.
Climb speed.

[Supp. 24, 19 F. R. 4457, July 20, 1954]

STABILITY

§ 4b.150 **General.**

The airplane shall be longitudinally, directionally, and laterally stable in accordance with §§ 4b.151 through 4b.158. Suitable stability shall be required in other conditions normally encountered in service if flight tests show such stability to be necessary for safe operation.

[Amdt. 4b-12, 27 F.R. 2989, Mar. 30, 1962]

§ 4b.151 **Static longitudinal stability.**

In the conditions outlined in §§ 4b.152 through 4b.155, the characteristics of the elevator control forces including friction and the elevator control surface displacement shall comply with paragraphs (a) through (c) of this section.

(a) A pull shall be required to obtain and maintain speeds below the specified trim speed, and a push shall be required to obtain and maintain speeds above the specified trim speed, except that if the elevator control forces are not dependent upon the hinge moments of the elevator control surface it shall also be shown that an upward displacement of the elevator trailing edge is required to obtain and maintain speeds below the specified trim speed and a downward displacement of the elevator trailing edge is required to obtain and maintain speeds above the specified trim speed. These criteria shall apply to any speed which can be obtained, except that such speeds need not be greater than the landing gear or the wing flap operating limit speed or V_{FC}/M_{FC}, whichever is appropriate, or need not be less than the minimum speed in steady unstalled flight.

(b) The air speed shall return to within 10 percent of the original trim speed when the control force is slowly released from any speed within the limits defined in paragraph (a) of this section.

(c) The stable slope of the stick force versus speed curve shall not be less than 0.5 pound per 3 knots nor shall it exceed a value beyond which control of the airplane is difficult.

[15 F.R. 3543, June 8, 1950, as amended by Amdt. 4b-11, 24 F.R. 7068, Sept. 1, 1959; Amdt. 4b-12, 27 F.R. 2989, Mar. 30, 1962]

§ 4b.152 **Stability during landing.**

The stick force curve and, if required by § 4b.151(a), the elevator angle curve shall have stable slopes and the stick force shall not exceed 80 pounds at any speed between 1.1 V_{s_0} and 1.8 V_{s_0} with:

(a) Wing flaps in the landing position;
(b) The landing gear extended;
(c) Maximum landing weight;
(d) Power, or thrust, off on all engines; and

(e) The airplane trimmed at 1.4 V_{s_0} with power or thrust off.

[Amdt. 4b–12, 27 F.R. 2989, Mar. 30, 1962]

§ 4b.153 Stability during approach.

The stick force curve and, if required by § 4b.151(a), the elevator angle curve shall have stable slopes at all speeds between 1.1 V_{s_1} and 1.8 V_{s_1} with:

(a) Wing flaps in the approach position;

(b) Landing gear retracted;

(c) Maximum landing weight; and

(d) The airplane trimmed at 1.4 V_{s_1} and with power sufficient to maintain level flight at this speed.

[Amdt. 4b–12, 27 F.R. 2989, Mar. 30, 1962]

§ 4b.154 Stability during climb.

The stick force curve and, if required by § 4b.151(a), the elevator angle curve shall have stable slopes at all speeds between 85 and 115 percent of the speed at which the airplane is trimmed with:

(a) Wing flaps retracted;

(b) Landing gear retracted;

(c) Maximum takeoff weight;

(d) 75 percent of maximum continuous power for reciprocating engines; maximum power or thrust selected by the applicant as an operating limitation for use during climb (see § 4b.718) for turbine engines; and

(e) The airplane trimmed at the best rate-of-climb speed except that the speed need not be less than 1.4 V_{s_1}.

[Amdt. 4b–12, 27 F.R. 2989, Mar. 30, 1962]

§ 4b.155 Stability during cruising.

(a) *Landing gear retracted; high speed.* The stick force curve and, if required by § 4b.151(a), the elevator angle curve shall have stable slopes at all speeds from V_{FC}/M_{FC} to the speed equal to $V_{FC}-\left(\frac{V_{FC}-1.4\,V_{s_1}}{2}\right)$ or to 50 knots less than the trim speed specified in subparagraph (4) of this paragraph, whichever is the lesser speed except that it need not be less than 1.4 V_{s_1}, and the stick force shall not exceed 50 pounds with:

(1) Wing flaps retracted;

(2) The most critical weight between maximum landing weight and maximum takeoff weight;

(3) 75 percent of maximum continuous power for reciprocating engines; maximum cruising power selected by the applicant as an operating limitation (see § 4b.718) for turbine engines, except that the power need not exceed that required at V_{MO}/M_{MO}; and

(4) The airplane trimmed for level flight with the power required in subparagraph (3) of this paragraph.

(b) *Landing gear retracted; low speed.* The stick force curve and, if required by § 4b.151(a), the elevator angle curve shall have stable slopes at all speeds from a speed equal to

$$V_{FC}-\left(\frac{V_{FC}-1.4\,V_{s_1}}{2}\right)$$

to 1.4 V_{s_1} and the stick force shall not exceed 50 pounds with the wing flaps and weight as specified in paragraph (a) of this section and with:

(1) Power required for level flight at a speed equal to $V_{FC}-\left(\frac{V_{FC}-1.4\,V_{s_1}}{2}\right)$; and

(2) The airplane trimmed for level flight with the power required in subparagraph (1) of this paragraph.

NOTE: At altitudes where Mach number is critical, the calibrated airspeed corresponding with M_{FC} may be used to calculate the speed $V_{FC}-\left(\frac{V_{FC}-1.4\,V_{s_1}}{2}\right)$.

(c) *Landing gear extended.* The stick force curve and, if required by § 4b.151(a), the elevator angle curve shall have stable slopes at all speeds between 1.4 V_{s_1} and V_{LE} and the stick force shall not exceed 50 pounds with the wing flaps and the weight as specified in paragraph (a) of this section and with:

(1) Power required for level flight at V_{LE}; and

(2) The airplane trimmed for level flight with the power required in subparagraph (1) of this paragraph.

[Amdt. 4b–12, 27 F.R. 2989, Mar. 30, 1962]

§ 4b.156 Dynamic longitudinal stability.

Any short period oscillation occurring between stalling speed and maximum permissible speed appropriate to the configuration of the airplane (e.g., V_{FE}, V_{LE}, or V_{FC}/M_{FC}) shall be heavily damped with the primary controls free and in a fixed position.

[15 F.R. 3543, June 8, 1950, as amended by Amdt. 4b–12, 27 F.R. 2990, Mar. 30, 1962]

§ 4b.156–1 Procedure for demonstrating dynamic longitudinal stability (FAA policies which apply to § 4b.156).

Damping of accelerations and movement of the control should be noted when:

(a) The control column is quickly offset and immediately released and

(b) The control column is quickly offset and immediately returned to the trim position and held in this position.

[Supp. 24, 19 F. R. 4458, July 20, 1954]

§ 4b.157 Static directional and lateral stability.

(a) The static directional stability, as shown by the tendency to recover from a skid with rudder free, shall be positive with all landing gear and flap positions and symmetrical power conditions, at all speeds from 1.2 V_{s_1} up to V_{FE}, V_{LE}, or V_{FC}/M_{FC}, whichever is appropriate to the airplane configuration.

(b) The static lateral stability, as shown by the tendency to raise the low wing in a sideslip with the aileron controls free and with all landing gear and flap positions and symmetrical power conditions, shall:

(1) Be positive at V_{FE}, V_{LE}, or V_{FC}/M_{FC}, whichever is appropriate to the airplane configuration.

(2) Not be negative at a speed equal to 1.2 V_{s_1}.

(c) In straight steady sideslips (unaccelerated forward slips) the aileron and rudder control movements and forces shall be substantially proportional to the angle of sideslip, and the factor of proportionality shall lie between limits found necessary for safe operation throughout the range of sideslip angles appropriate to the operation of the airplane. At greater angles up to that at which the full rudder control is employed or a rudder pedal force of 180 pounds is obtained, the rudder pedal forces shall not reverse, and increased rudder deflection shall produce increased angles of sideslip. Sufficient bank shall accompany sideslipping to indicate clearly any departure from steady unyawed flight, unless a yaw indicator is provided.

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b–6, 17 F.R. 1089, Feb. 5, 1952; Amdt. 4b–12, 27 F.R. 2990, Mar. 30, 1962]

§ 4b.157–1 Procedure for demonstrating static directional and lateral stability (FAA policies which apply to § 4b.157).

(a) *Motion involving roll.* No real motion of the airplane involving roll is possible without yaw also being involved, and vice versa. In showing compliance with § 4b.157 the rolling and yawing stability should be investigated separately.

(b) *Directional stability.* Directional stability should be investigated by starting from steady flight in the required configuration and deflecting the rudder at a fairly rapid rate by the amount required to maintain a steady skid with the airplane yawed approximately 20° (as read on the directional gyro) while the wings are maintained level by use of the ailerons, and the speed held constant by means of the elevator control. When the steady condition has been established, the rudder should be released and, if the airplane is directionally stable, it should cease to skid; i. e., the yaw should decrease to approximately zero and, if also laterally stable the aileron deflection and force required to hold the wings level should also approach zero. The test should be conducted by executing skids both to the right and left, recording in each case the time required from the release of the rudder controls and the number of oscillations, if any, involved to recover to steady level flight.

(c) *Lateral stability.* Lateral stability should be investigated by starting from steady flight in the required configuration and banking the airplane approximately 20° (as read on the gyro horizon) by means of the ailerons, while maintaining the original heading by means of the rudder, and the original speed by means of the longitudinal trimming device. When the steady slipping condition has been established, the aileron control should be released. If the airplane is laterally stable, it should cease to slip; i. e., the wing should return to an approximately level attitude, and the rudder deflection and pedal force required to maintain the heading should approach zero. The test should be conducted by executing slips from both to right and left, and in each case the time required from the release of the aileron control and the number of oscillations, if any, involved to recover to steady level flight should be recorded.

(d) *Additional test.* In addition to the directional and lateral stability tests, § 4b.157 (c) contains provisions which should be used to test the airplane for rudder over balance.

(e) *Static directional stability test,* § 4b.157 (a) and (c).

CAUTION: Prior to conducting this test and that in paragraph (f) of this section, complete agreement should be reached between the applicant and the FAA Flight Test Agent to insure that the severity of control application will not result in loads exceeding the design limitations.

(1) *Configuration.* This test should be conducted in the configurations that follow:

Maximum take-off weight with wing flaps retracted.
Maximum landing weight with wing flaps extended.
C. G. position—Most aft.
Wing flaps—Retracted and maximum landing position.
Landing gear—Retracted and extended.

(2) *Test procedure and required data.* The following tests should be conducted at the altitude deemed most critical for the combination of power and aerodynamic damping effect:

(i) The airplane should be yawed slowly to the left and right using ailerons to hold wings level, and, when controls are released slowly, the tendency of airplane to recover from the skid should be noted.

(ii) The qualitative proportionality of rudder and aileron deflection and force during steady straight sideslips should be noted.

(iii) Damping of yawing and movement of control should be noted when the rudder is quickly offset and immediately released and when the rudder is quickly offset and immediately returned and held in the trim position.

(f) *Static lateral stability test, § 4b.157 (b)*—(1) *Configuration.* This test should be conducted in the configurations that follow:

Maximum take-off weight with flaps retracted.
Maximum landing weight with flaps extended.
C. G. position—Most aft.
Wing flaps—Retracted and maximum landing position.
Landing gear—Retracted and extended.

[Supp. 24, 19 F.R. 4458, July 20, 1954, as amended by Supp. 34, 22 F.R. 6963, Aug. 29, 1957; Amdt. 4b-12, 27 F.R. 2990, Mar. 30, 1962]

§ 4b.158 Dynamic directional and lateral stability.

Any short period oscillation occurring between stalling speed and maximum permissible speed appropriate to the configuration of the airplane (e.g., V_{FE}, V_{LE}, or V_{FC}/M_{FC}) shall be heavily damped with the primary controls free and in a fixed position.

[15 F.R. 3543, June 8, 1950, as amended by Amdt. 4b-12, 27 F.R. 2990, Mar. 30, 1962]

§ 4b.158-1 Procedure for demonstrating dynamic directional and lateral stability (FAA policies which apply to § 4b.158).

Damping of yawing and movement of the control should be noted during the test procedure in § 4b.157-1(e)(2)(iii).

[Supp. 24, 19 F.R. 4459, July 20, 1954]

STALLING CHARACTERISTICS

§ 4b.160 Stalling; symmetrical power.

(a) Stalls shall be demonstrated with the airplane in straight flight and in banked turns at 30 degrees, both with power off and with power on. In the power-on conditions the power shall be that necessary to maintain level flight at a speed of 1.6 V_{s_1}, where V_{s_1} corresponds with the stalling speed with flaps in the approach position, the landing gear retracted, and maximum landing weight.

(b) The stall demonstration shall be in the following configurations:

(1) Wing flaps and landing gear in any likely combination of positions,

(2) Representative weights within the range for which certification is sought,

(3) The center of gravity in the most adverse position for recovery.

(c) The stall demonstration shall be conducted as follows:

(1) With the airplane trimmed for straight flight at the speed prescribed in § 4b.112(c)(1), the speed shall be reduced by means of the elevator control until it is steady at slightly above stalling speed; after which the elevator control shall be applied at a rate such that the airplane speed reduction does not exceed one mile per hour per second until the airplane is stalled or, if the airplane is not stalled, until the control reaches the stop.

(2) The airplane shall be considered stalled when, at an angle of attack measurably greater than that of maximum lift, the inherent flight characteristics give a clear indication to the pilot that the airplane is stalled, except that for airplanes demonstrating unmistakable inherent aerodynamic warning associated with the stall in all required configurations, the speed need not be reduced below a value which provides an adequate stall warning margin as defined in § 4b.162.

NOTE: A nose-down pitch or a roll which cannot be readily arrested are typical indications that the airplane is stalled. Other indications, such as marked loss of control effectiveness, abrupt change in control force or motion, characteristic buffeting, or a distinctive vibration of the pilot's controls, may be accepted if found in a particular case to be sufficiently clear. Types of inherent aerodynamic warning considered acceptable include characteristics such as buffeting, small amplitude pitch or roll oscillations, distinctive shaking of the pilots' controls, etc.

(3) Recovery from the stall shall be effected by normal recovery techniques, starting as soon as the airplane is stalled.

(d) During stall demonstration it shall be possible to produce and to correct roll and yaw by unreversed use of the aileron and rudder controls up to the moment the airplane is stalled; there shall occur no abnormal nose-up pitching; and the longitudinal control force shall be positive up to and including the stall.

(e) Straight flight stalls shall be entered with wings level. The roll occurring between the stall and the completion of the recovery shall not exceed approximately 20 degrees.

(f) In turning flight stalls the action of the airplane following the stall shall not be so violent or extreme as to make it difficult with normal piloting skill to effect a prompt recovery and to regain control of the airplane.

(g) In both the straight flight and the turning flight stall demonstrations it shall be possible promptly to prevent the airplane from stalling and to recover from the stall condition by normal use of the controls.

[Amdt. 4b-5, 16 F. R. 12220, Dec. 4, 1951, as amended by Amdt. 4b-3, 21 F. R. 990, Feb. 11, 1956; Amdt. 4b-12, 27 F.R. 2990, Mar. 30, 1962]

§ 4b.160-1 Procedure for demonstrating stall tests, symmetrical power (FAA policies which apply to § 4b.160(c)(2)).

(a) *Angle-of-attack.* The angle-of-attack during the stall maneuver should be increased at least to the point where the following two conditions are satisfied:

(1) Attainment of an angle-of-attack measurably greater than that for maximum lift.

(2) Clear indication to the pilot through the inherent flight characteristics that the airplane is stalled.

(b) *Procedure to be used.* The following procedure may be used to demonstrate that these two conditions are fulfilled.

(1) A photopanel or equivalent method of obtaining continuous records of the following variables at not greater than ¼ second intervals should be provided: indicated angle-of-attack, swivel static and shielded total pressure head or equivalent, pressure altitude, pitch and bank angle, normal acceleration, elevator position and force, aileron and rudder position.

(2) If it is evident that longitudinal stick force is always positive, that is, no reversal exists down to the stall, then time history of this item should not be required.

(3) In order to insure that an accurate recording of the indicated angle-of-attack is obtained, the sensing device should be located in a region where tuft surveys show that the streamlines undergo no radical change in direction up to the maximum angle contemplated. Regions well forward of the wing leading edge are desirable to keep the angular difference between true angle and indicated angle as small as possible.

(4) Means for indicating the stall warning point and the point at which the pilot is informed by the inherent flight characteristics that the airplane is stalled should be provided. This may consist of a light on the photopanel operated by a switch mounted on the control wheel. In order to insure that the camera records the light image a time delay device may need to be incorporated in the light circuit in cases where camera speed is low.

(c) *Configuration.* Stalls should be conducted in the configurations noted in the following listings and with cowl flaps appropriate for the flight condition. Power off stalls should be conducted with the engines idling and propellers in low pitch. For power on conditions, stalls should be conducted with that power necessary to maintain level flight at a speed of 1.6 V_{s_1} with flaps in the approach position, landing gear retracted and maximum landing weight.

(1) *Stalls—straight flight.*

Gross weight	C. G.	Power	Positions	
			Flap	Gear
Maximum landing	Most forward	Off	Retracted	Retracted.[1]
Do	do	Off	Take-off	Do.[1]
Do	do	Off	Approach	Do.[1]
Do	do	Off	Landing	Extended.[1]
Do	do	On	Approach	Do.[2]
Do	do	On	Landing	Do.[2]
Do	Most aft	Off	Approach	Do.[2]
Do	do	Off	Landing	Do.[2]
Do	do	On	Approach	Retracted.
Do	do	On	do	Extended.
Do	do	On	Landing	Retracted.
Do	do	On	do	Extended.
Maximum take-off	do	Off	Retracted	Do.[2]
Do	do	Off	Take-off	Do.[2]
Do	do	On	Retracted	Do.[2]
Do	do	On	Take-off	Do.[2]

[1] May be demonstrated during stalling speed tests. See § 14b.112.
[2] Use extended, unless, due to direction of C. G. shift with gear, retracted gear is more critical. If retracted is more critical use retracted position for these stalls.

(2) *Stalls—30° banked turns.*

Gross weight	C. G.	Power	Positions		Direction
			Flap	Gear	
Maximum take-off	Most aft	Off	Retracted	Retracted	To right.
Do	do	Off	do	do	To left.
Do	do	On	do	do	To right.
Do	do	On	do	do	To left.
Maximum landing	do	Off	Landing	Extended	To right.
Do	do	Off	do	do	To left.
Do	do	On	do	do	Do.
Do	do	On	do	do	To right.

(d) *Test procedure and required data.* The stall tests may be conducted at any optional altitude (see § 4b.100-3 (d)). The flight test procedure should be conducted in accordance with § 4b.160 (c) (1). The pilot should be provided with a yawmeter or equivalent means for maintaining the angle of yaw as near zero as possible. The operation of the photopanel recording system, previously described in paragraph (b) (1) of this section, should be started at least 20 mph above the stall speed and allowed to operate continuously until the stall recovery is completed. The pilot's indication of stall warning and the actual occurrence of the stall should be obtained. In addition to the data obtained on the photopanel shown in paragraph (b) (1) of this section, the following information should also be recorded:

Weight.
C. G. position.
Ambient air temperature.
Wing flap position.
Landing gear position.
Engines, rpm and manifold pressure.

(e) *Data analysis.* Time histories should be plotted of the photopanel instruments. The stalling warning point (see § 4b.162-1) and the point at which "the inherent flight characteristics give a clear indication that the airplane is stalled" should be noted on the plots.

(1) Inspection of the plots will then show if the following two conditions are fulfilled:

(i) The indicated angle-of-attack increases steadily to a value measurably beyond that for maximum lift, and

(ii) The stall is evident to the pilot prior to initiation of recovery.

(2) Consideration should be given to the following points in the time-history analysis:

(i) The direction in which the elevator is moving, i. e., any nose down pitch or decrease in load factor not induced by inadvertent elevator motion,

(ii) Rudder and aileron movement with respect to uncontrollable roll,

(iii) The effect of lag in the air-speed system,

(iv) Rate of air speed change,

(v) Effect of pitching velocity or rolling velocity on indicated angle-of-attack. If possible, angle-of-attack time history should be drawn through points where pitching or rolling velocity are small. If corrections are unavoidable, the angular correction is simply $\tan^{-1} pd/V$ where p is the rolling or pitching velocity, V is the true air speed, and d is the distance from the pitch or rolling axis, as the case may be, to the sensing device. It should be noted that this correction is applicable to either true or indicated angle-of-attack.

(vi) The indicated acceleration is a function of the angle the vertical axis of the accelerometer makes with the perpendicular to the earth's surface. Therefore, bank angle will seriously affect the maximum lift point. For example, a bank of 45° without loss of lift will result in a drop in indicated acceleration from 1.00 to .707. As a result acceleration data obtained in the region of $C_{L\max}$ should be disregarded or corrected for bank angle when the bank angle exceeds nine or ten degrees.

[Supp. 24, 19 F. R. 4459, July 20, 1954]

§ 4b.161 Stalling; asymmetrical power.

(a) The airplane shall be safely recoverable without applying power to the inoperative engine when stalled with:

(1) The critical engine inoperative,

(2) Flaps and landing gear retracted,

(3) The remaining engines operating up to 75 percent of maximum continuous power, except that the power need not be greater than that at which the wings can be held level laterally with the use of maximum control travel.

(b) It shall be acceptable to throttle back the operating engines during the recovery from the stall.

§ 4b.161-1 Procedure for demonstrating stall test, asymmetrical power (FAA policies which apply to § 4b.161).

(a) *Control of airplane.* During this test the airplane should not become uncontrollable or lose an excessive amount of altitude when so stalled.

(b) *Configuration.* This test should be conducted in the configuration that follows:

Weight—Maximum take-off.
C. G. position—Most aft.
Wing flaps—Retracted.
Landing gear—Retracted.
Operating engine(s)—Power up to 75 percent maximum continuous power, cowl flaps optional.
Critical inoperative engine—Propeller optional, feathered or windmilling, cowl flaps appropriate for flight condition.
Trim speed—1.4 V_{s_1}.

(c) *Test procedure and required data.* This test may be conducted at any optional altitude (see § 4b.100-3 (c)). See § 4b.160 (c) regarding test procedure.

(1) The speed of the airplane should be reduced from the trim condition with the wings held level until the first of the following occurs:

(i) Full rudder or aileron deflection.

(ii) 180 lbs. rudder force.

(iii) Stall is reached.

(2) If full rudder or aileron deflection, or the 180 lbs. rudder force occurs first, the power should be reduced and the test repeated until sufficient control is available to complete the stall. The power may be reduced on the operating engine(s) before reapplying power on the operating engine or engines for the purpose of regaining level flight. The following data should be recorded at that point:

Pressure altitude.
Ambient air temperature.
Indicated air speed.
Engines, rpm and manifold pressure.
Torque pressure.
Carburetor air temperature.
Rudder force (if desirable).

(3) If stall is reached first, the same data should be recorded.

[Supp. 24, 19 F. R. 4460, July 20, 1954]

§ 4b.162 Stall warning.

Clear and distinctive stall warning shall be apparent to the pilot with sufficient margin to prevent inadvertent stalling of the airplane with flaps and landing gear in all normally used positions, both in straight and in turning flight. It shall be acceptable for the warning to be furnished either through the inherent aerodynamic qualities of the airplane or by a device which will give clearly distinguishable indications under all expected conditions of flight.

NOTE: A stall warning beginning at a speed 7 percent above the stalling speed is normally considered sufficient margin. Other margins may be acceptable depending upon the degree of clarity, duration, and distinctiveness of the warning and upon other characteristics of the airplane evidenced during the approach to the stall.

[Amdt. 4b-5, 16 F. R. 12220, Dec. 4, 1951]

§ 4b.162-1 Stall warning (FAA policies which apply to § 4b.162).

(a) The adequacy of stall warning should depend on the relative ease with which an airplane might be inadvertently stalled following the occurrence of stall warning. For example, if unmistakable warning occurs only 2 percent above the stall speed but undue pilot effort is required to reduce the air speed to the stall, the speed margin of 2 percent may be adequate. On the other hand if conscious effort is required to avoid stalling the airplane, a positive type of warning initiated at a relatively high speed above the stall may be required.

(b) Suggested suitable stall warnings are, buffeting which may be defined as general shaking or vibration of the airplane or elevator shake of sufficient magnitude to be unmistakable; or a stall warning instrument such as a stick shaker. A visual stall warning device which requires the attention of the crew within the cockpit is not considered acceptable by itself.

[Supp. 24, 19 F. R. 4460, July 20, 1954]

GROUND HANDLING CHARACTERISTICS

§ 4b.170 Longitudinal stability and control.

(a) There shall be no uncontrollable tendency for landplanes to nose over in any reasonably expected operating condition or when rebound occurs during the landing or take-off.

(b) Wheel brakes shall operate smoothly and shall exhibit no undue tendency to induce nosing over.

(c) When a tail-wheel landing gear is used it shall be possible during the take-off ground run on concrete to maintain any attitude up to thrust line level at 80 percent of V_{s_1}.

§ 4b.170-1 Procedure for demonstrating longitudinal stability and control on the ground (FAA policies which apply to § 4b.170).

Taxiing tests at velocities up to 70 percent of the stalling speed should be conducted on smooth and rough ground which may likely be encountered under normal operating conditions. Particular attention should be paid to the following:

(a) *Taxiing over rough ground.* There is some evidence to indicate that critical loads can be built up in taxiing over rough ground, even when the shock-absorbing system is entirely satisfactory with respect to capacity for landing purposes.

(b) *Brakes.* Their adequacy when maneuvering on the ground and their tendency to cause nosing-over should be investigated. Any bad tendency will normally be exaggerated when taxiing in a strong side or tail wind.

[Supp. 24, 19 F. R. 4460, July 20, 1954]

§ 4b.170-2 Longitudinal stability and control with reverse thrust (FAA policies which apply to § 4b.170).

The policies outlined in § 4b.402-1 (a), (d), and (e) will apply.

[Supp. 25, 20 F. R. 2277, Apr. 8, 1955]

§ 4b.170-3 Longitudinal stability and control with an antiskid device installed (FAA policies which apply to § 4b.170).

The policies outlined in § 4b.337-4 will apply.

[Supp. 28, 21 F. R. 2558, Apr. 19, 1956]

§ 4b.171 Directional stability and control.

(a) There shall be no uncontrollable ground-looping tendency in 90° cross winds of velocity up to 0.2 V_{s_0} at any ground speed at which the airplane is expected to operate.

(b) All landplanes shall be demonstrated to be satisfactorily controllable with no exceptional degree of skill or alertness on the part of the pilot in power-off landings at normal landing speed during which brakes or engine power are not used to maintain a straight path.

(c) Means shall be provided for directional control of the airplane during taxiing.

§ 4b.171-1 Procedure for demonstrating direction stability and control on the ground (FAA policies which apply to § 4b.171).

(a) Compliance with the requirement of § 4b.171(a) may be demonstrated during tests for the establishment of the cross wind component velocity in accordance with § 4b.173.

(b) Compliance with the requirement of § 4b.171 (b) may be demonstrated during power off landings in other tests.

(c) Compliance with the requirement of § 4b.171 (c) may be demonstrated during taxiing prior to take-off or after landing from other flight tests.

[Supp. 24, 19 F. R. 4460, July 20, 1954]

§ 4b.171-2 Directional stability and control with reverse thrust (FAA policies which apply to § 4b.171).

The policies outlined in § 4b.402-1 (a), (d), and (e) will apply.

[Supp. 25, 20 F. R. 2278, Apr. 8, 1955]

§ 4b.171-3 Directional stability and control with an antiskid device installed (FAA policies which apply to § 4b.170).

The policies outlined in § 4b.337-4 will apply.

[Supp. 28, 21 F. R. 2558, Apr. 19, 1956]

§ 4b.172 Shock absorption.

The shock absorbing mechanism shall not produce damage to the structure when the airplane is taxied on the roughest ground which it is reasonable to expect the airplane to encounter in normal operation.

§ 4b.172-1 Shock absorbing mechanism tests (FAA policies which apply to § 4b.172).

The shock absorbing mechanism should be checked for satisfactory operation while taxiing, taking-off and landing during other tests in the type certification program.

[Supp. 24, 19 F. R. 4460, July 20, 1954]

§ 4b.173 Demonstrated cross wind.

There shall be established a cross component of wind velocity at which it has been demonstrated to be safe to take off or land.

§ 4b.173-1 Cross wind demonstration (FAA policies which apply to § 4b.173).

(a) *Cross wind component.* A cross wind component of not less than 0.2 V_{s_0} should be established during type tests. Consequently, two results are possible:

(1) A cross wind component may be established at a value which is not marginal with the airplane's handling characteristics. This value should be included in the Operating Procedures section of the Airplane Flight Manual. The operation of the aircraft in cross winds greater than the value specified is not necessarily a hazard. Thus operation in cross winds of a greater value is entirely within the discretion of the operator.

(2) A critical cross wind component may be established at a value which is considered the maximum up to which it is safe to operate the airplane on the ground, including take-offs and landings. This value should be shown in the Oper ating Limitations section of the Airplan Flight Manual. Operation of the ai plane in cross winds above the maxi mum safe value is considered hazardou and the operator should do so only o the same emergency basis that a pilc would be justified in exceeding any o the operating limitations such as ai speed, engine rpm, C. G. limitations, etc

(3) An operator may of course restric the operation of his airplane to cros wind components of any value equal t or less than that established during th type certification tests.

(b) *Configuration.* This test shoulc be conducted in the configurations that follow:

Weight—Maximum take-off and landing.
C. G. position—Most aft.
Flaps—Take-off and maximum landing positions.

(c) *Test procedure and required data.* At least three take-offs and landings should be made in cross wind components of 0.2 V_{s_0} mph (or greater at applicant's option) to demonstrate satisfactory controllability and handling characteristics. The magnitude and direction of the cross wind should be established by the use of appropriate meteorological instruments.

[Supp. 24, 19 F. R. 4460, July 20, 1954]

§ 4b.173-2 Ground handling characteristics with reverse thrust (FAA policies which apply to § 4b.173).

The policies outlined in § 4b.402-1 (a) and (d) will apply.

[Supp. 25, 20 F. R. 2278, Apr. 8, 1955]

WATER HANDLING CHARACTERISTICS

§ 4b.180 Water conditions.

The most adverse water conditions in which the seaplane has been demonstrated to be safe for take-off, taxiing, and alighting shall be established.

[Amdt. 4b.6, 17 F.R. 1089, Feb. 5, 1952]

§ 4b.180-1 Water handling qualities (FAA policies which apply to § 4b.180).

Policies outlined in § 4b.182-1 will apply.

[Supp. 24, 19 F. R. 4461, July 20, 1954]

§ 4b.181 Wind conditions.

The following wind velocities shall be established:

(a) A lateral component of wind velocity not less than $0.2V_{s_0}$ at and below which it has been demonstrated that the seaplane is safe for taking off and alighting under all water conditions in which the seaplane is likely to be operated;

(b) A wind velocity at and below which it has been demonstrated that the seaplane is safe in taxiing in all directions, under all water conditions in which the seaplane is likely to be operated.

[Amdt. 4b–6, 17 F.R. 1089, Feb. 5, 1952]

§ 4b.181–1 Cross wind demonstration (FAA policies which apply to § 4b.181).

Policies outlined in § 4b.173–1 will apply.

[Supp. 24, 19 F. R. 4461, July 20, 1954]

§ 4b.182 Control and stability on the water.

(a) In taking off, taxiing, and alighting, the seaplane shall not exhibit the following:

(1) Any dangerously uncontrollable porpoising, bouncing, or swinging tendency;

(2) Any submerging of auxiliary floats or sponsons, any immersion of wing tips, propeller blades, or other parts of the seaplane which are not designed to withstand the resulting water loads;

(3) Any spray forming which would impair the pilot's view, cause damage to the seaplane, or result in ingress of an undue quantity of water.

(b) Compliance with paragraph (a) of this section shall be shown under the following conditions:

(1) All water conditions from smooth to the most adverse condition established in accordance with § 4b.180;

(2) All wind and cross-wind velocities, water currents, and associated waves and swells which the seaplane is likely to encounter in operation on water;

(3) All speeds at which the seaplane is likely to be operated on the water;

(4) Sudden failure of the critical engine, occurring at any time while the airplane is operated on water;

(5) All seaplane weights and center of gravity positions within the range of loading conditions for which certification is sought, relevant to each condition of operation.

(c) In the water conditions of paragraph (b) of this section and the corresponding wind conditions the seaplane shall be able to drift for 5 minutes with engines inoperative, aided if necessary by a sea anchor.

[Amdt. 4b–6, 17 F. R. 1089, Feb. 5, 1952]

§ 4b.182–1 Procedure for demonstrating control and stability on the water (FAA policies which apply to § 4b.182).

(a) In order to check water stability, taxiing tests should be made in a cross wind determined in accordance with § 4b.181.

(b) Porpoising tendencies should be investigated and reported for extreme loading conditions.

(c) The ability to maneuver up to and while on the step should be investigated and the results reported.

(d) Compliance with the spray requirements may be substantiated while taxiing, taking off, and landing during other tests in the type certification program.

(e) If water rudders are provided, their effectiveness should be checked.

(f) Water taxiing ability should be investigated by actually taxiing the seaplane with appropriate use of power.

[Supp. 24, 19 F. R. 4461, July 20, 1954]

§ 4b.182–2 Control and stability on the water with reverse thrust (FAA policies which apply to § 4b.182).

The policies outlined in § 4b.402–1 (a), (d), (f), and (h) will apply.

[Supp. 25, 20 F. R. 2278, Apr. 8, 1955]

Miscellaneous Flight Requirements

§ 4b.190 Flutter and vibration.

(a) All parts of the airplane shall be demonstrated in flight to be free from flutter and excessive vibration under all speed and power conditions appropriate to the operation of the airplane up to at least the minimum value permitted for V_D in § 4b.210(b)(5). The maximum speeds so demonstrated shall be used in establishing the operating limitations of the airplane in accordance with § 4b.711.

(b) There shall be no buffeting condition in normal flight severe enough to interfere with the control of the airplane, to cause excessive fatigue to the crew, or to cause structural damage.[1] (See also § 4b.308.)

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b–7, 17 F. R. 11631, Dec. 20, 1952]

[1] It is not the intent of this requirement to discourage such stall warning buffeting as does not contradict these provisions.

§ 4b.190–1 Determination of flutter and vibration qualities during dive (FAA policies which apply to § 4b.190).

(a) *Observation for tendencies.* The airplane should be observed for flutter and vibration tendencies during other tests in the type certification program. In case the design speed is limited at altitude by Mach number, the dive should be conducted at a combination of pressure altitudes and equivalent air speed to permit attaining the desired maximum Mach number and dynamic pressure simultaneously. Stability and control qualities should be noted during the dive.

(b) *Configuration.* This test should be conducted in the configurations that follow:

(1) *Maximum take-off weight.*

C. G. Position—Most aft.
Wing flaps—Retracted and take-off position.
Landing gear—Retracted.
Engines—Power as desired.
Cooling controls—Optional.
Pneumatic boots—Inoperative.

(2) *Maximum landing weight.*

C. G. position—Most rearward.
Wing flaps—Approach and landing positions.
Landing gear—Extended.
Engines—Power as desired.
Cooling controls—Optional.
Pneumatic boots—Inoperative.

(c) *Test procedure and required data.* The speed of the airplane should be slowly increased, from a steady flight high speed condition, until the maximum calibrated design dive speed for maximum take-off weight is attained. The power and trim may be adjusted during the dive. The dive should be entered at a sufficiently high altitude to insure safe recovery. In case the design speed is limited at altitude by Mach number, the airplane should be dived at constant Mach number (maximum design or highest desired by the applicant—but in no case less than that specified in § 4b.210 (b) (5)) until the maximum equivalent design dive speed is attained. The test should be repeated at maximum landing weight with flaps and gear extended diving to the maximum design flap speed or speeds.

CAUTION: Throughout these tests any control displacements should be executed gently.

(1) The following data should be recorded for each test:

Pressure altitude.
Ambient air temperature.
Indicated air speed.
Machmeter reading (if applicable).
Engines, rpm and manifold pressure.
Wing flap position.
Landing gear position.
Weight.
C. G. position.

[Supp. 24, 19 F. R. 4461, July 20, 1954]

§ 4b.191 High-speed characteristics.

(a) *Speed increase and recovery characteristics.* (1) Operating conditions or characteristics likely to cause inadvertent speed increases, including upsets in pitch and roll, shall be simulated with the airplane trimmed at any likely cruise speed up to V_{MO}/M_{MO}. Allowing for pilot reaction time after effective inherent or artificial speed warning occurs (see § 4b.603(k)), it shall be demonstrated that the airplane can be recovered to a normal attitude and its speed reduced to V_{MO}/M_{MO} without requiring exceptional strength or skill on the part of the pilot, without exceeding V_D/M_D, V_{DF}/M_{DF}, or the structural limitations, and without producing buffeting which would cause structural damage.

NOTE: Examples of operating conditions or characteristics likely to cause speed increases are: gust upsets, inadvertent control movements, low stick force gradient in relation to control friction, passenger movement, leveling off from climb, and descent from Mach to airspeed limit altitudes.

(2) At all speeds up to V_{DF}/M_{DF}, there shall be no control reversal. Any reversal of elevator control force or tendency of the airplane to pitch, roll, or yaw, shall be mild and readily controllable using normal piloting technique.

(b) *Maximum speed for stability characteristics, V_{FC}/M_{FC}.* V_{FC}/M_{FC} shall be the maximum speed at which the requirements of §§ 4b.132(e), 4b.155(a), 4b.156, 4b.157(a), 4b.157(b), and 4b.158 are required to be met with flaps and landing gear retracted. It shall not be less than a speed halfway between V_{MO}/M_{MO} and V_{DF}/M_{DF}, except that in the altitude range where Mach number is the limiting factor, M_{FC} need not exceed the Mach number at which effective speed warning occurs.

[Amdt. 4b–12, 27 F.R. 2990, Mar. 30, 1962]

Subpart C—Structure

General

§ 4b.200 Loads.

Strength requirements of this subpart are specified in terms of limit and ultimate loads. Unless otherwise stated, the specified loads shall be considered as limit loads. In determining compliance with these requirements the following shall be applicable:

(a) The factor of safety shall be 1.5 unless otherwise specified.

(b) Unless otherwise provided, the specified air, ground, and water loads shall be placed in equilibrium with inertia forces, considering all items of mass in the airplane.

(c) All loads shall be distributed in a manner closely approximating or conservatively representing actual conditions.

(d) If deflections under load significantly change the distribution of external or internal loads, the redistribution shall be taken into account.

§ 4b.201 Strength and deformation.

(a) The structure shall be capable of supporting limit loads without suffering detrimental permanent deformations.

(b) At all loads up to limit loads the deformation shall be such as not to interfere with safe operation of the airplane.

(c) The structure shall be capable of supporting ultimate loads without failure. It shall support the load for at least 3 seconds, unless proof of strength is demonstrated by dynamic tests simulating actual conditions of load application.

(d) 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 with static loads, the effects of such rate of application shall be considered.

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b–6, 17 F. R. 1089, Feb. 5, 1952]

§ 4b.202 Proof of structure.

(a) Proof of compliance of the structure with the strength and deformation requirements of § 4b.201 shall be made for all critical loading conditions.

(b) Proof of compliance by means of structural analysis shall be acceptable only when the structure conforms to types for which experience has shown such methods to be reliable. In all other cases substantiating tests shall be required.

(c) In all cases certain portions of the structure shall be tested as specified in § 4b.300.

(d) Proof of compliance of the structure with the fatigue evaluation requirements of § 4b.270 shall be made.

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b–3, 21 F. R. 990, Feb. 11, 1956]

Flight Loads

§ 4b.210 General.

Flight load requirements shall be complied with at critical altitudes within the range selected by the applicant at all weights from the design minimum weight to the maximum weight appropriate to each particular flight condition, with any practicable distribution of disposable load within prescribed operating limitations stated in the Airplane Flight Manual. (See § 4b.740.) Compressibility effects shall be taken into account at all speeds.

(a) *Flight load factor.* The flight load factors specified in this subpart shall represent the component of acceleration in terms of the gravitational constant. The flight load factor shall be assumed to act normal to the longitudinal axis of the airplane, shall be equal in magnitude, and shall be opposite in direction to the airplane inertia load factor at the center of gravity.

(b) *Design air speeds.* The design air speeds shall be equivalent air speeds (EAS) and shall be chosen by the applicant, except that they shall not be less than the speeds defined in subparagraphs (1) through (5) of this paragraph. Where estimated values of the speeds V_{s_0} and V_{s_1} are used, such estimates shall be conservative.

(1) *Design flap speeds, V_F.* The design flap speed for each flap position established in accordance with § 4b.323 (a) shall 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. V_F shall be not less than:

(i) 1.6 V_{s_1} with flaps in takeoff position at maximum takeoff weight;

(ii) 1.8 V_{s_1} with flaps in approach position at maximum landing weight; and

(iii) 1.8 V_{s_0} with flaps in landing position at maximum landing weight.

Where an automatic flap positioning or load limiting device is employed, it shall be permissible to use the speeds and corresponding flap positions programmed or permitted by the device. (See § 4b.323 (c).)

(2) *Design maneuvering speed, V_A.* The design maneuvering speed, V_A, shall be equal to $V_{s_1}\sqrt{n}$ where n is the limit positive maneuvering load factor at V_C (see § 4b.211(a)) and V_{s_1} is the stalling speed with flaps retracted. Both V_A and V_S shall be evaluated at the design weight and altitude under consideration. V_A need not be greater than V_C or the speed at which the positive $C_{N_{max}}$ curve intersects the positive maneuver load factor line, whichever is the lesser. (See figure 4b–2.)

(3) *Design speed for maximum gust intensity, V_B.* V_B shall be either the speed determined by the intersection of the line representing the maximum positive lift $C_{N_{max}}$ and the line representing the rough air gust velocity on the gust V–n diagram or $(\sqrt{n_g})\ V_{s_1}$, whichever is the lesser; where n_g is the positive airplane gust load factor due to gust at speed V_C in accordance with § 4b.211 (b) (2) at the particular weight under consideration and V_{s_1} is the stalling speed with flaps retracted at the particular weight under consideration. V_B need not be greater than V_C.

(4) *Design cruising speed, V_C.* The minimum design cruising speed V_C shall be sufficiently greater than V_B to provide for inadvertent speed increases likely to occur as a result of severe atmospheric turbulence. In the absence of a rational investigation substantiating the use of other values, V_C shall not be less than V_B+50 (m. p. h.), except that it need not exceed the maximum speed in level flight at maximum continuous power for the corresponding altitude. At altitudes where V_D is limited by Mach number, it shall be acceptable to limit V_C to a Mach number selected by the applicant. (See § 4b.711.)

(5) *Design dive speed, V_D.* The design dive speed chosen by the applicant shall be used in determining the maximum operating limit speed for the airplane in accordance with § 4b.711.

(c) *Design fuel and oil loads.* The disposable load combinations shall include all fuel and oil loads in the range from zero fuel and oil to the maximum fuel and oil load selected by the applicant. It shall be permissible for the applicant to select a structural reserve fuel condition not exceeding 45 minutes of fuel under operating conditions defined in § 4b.437 (c). If a structural reserve fuel condition is selected, it shall be used as the minimum fuel weight condition for showing compliance with the flight load requirements as prescribed in this subpart, in which case, the provisions of subparagraphs (1) through (3) of this paragraph shall apply.

(1) The structure shall be designed for a condition of zero fuel and oil in the wing at limit loads corresponding with:

(i) A maneuver load factor of +2.25, and

(ii) Gust intensities equal to 85 percent of the values prescribed in § 4b.211 (b).

(2) Fatigue evaluation of the structure shall take into account any increase in operating stresses resulting from the design condition of subparagraph (1) of this paragraph (see § 4b.270).

(3) The flutter, deformation, and vibration requirements shall also be met with zero fuel (see § 4b.308).

[15 F. R. 3543, June 8, 1950; 15 F. R. 4171, June 29, 1950, as amended by Amdt. 4b–8, 18 F. R. 2214, Apr. 18, 1953; Amdt. 4b–2, 20 F. R. 5305, July 26, 1955; Amdt. 4b–3, 21 F. R. 990, Feb. 11, 1956; Amdt. 4b–6, 22 F.R. 5564, July 16, 1957; Amdt. 4b–11, 24 F.R. 7068, Sept. 1, 1959; Amdt. 4b–12, 27 F.R. 2990, Mar. 30, 1962]

§ 4b.211 Flight envelopes.

The strength requirements shall be met at all combinations of air speed and load factor on and within the boundaries of the V–n diagrams of figures 4b–2 and 4b–3 which represent the maneuvering and gust envelopes. These envelopes shall also be used in determining the airplane structural operating limitations as specified in § 4b.710.

(a) *Maneuvering load factors.* (See fig. 4b–2.) The airplane shall be assumed to be subjected to symmetrical maneuvers resulting in the limit load factors prescribed in subparagraphs (1) and (2) of this paragraph, except where limited by maximum (static) lift coefficients. Pitching velocities appropriate to the corresponding pull-up and steady turn maneuvers shall be taken into account. Lower values of maneuvering load factor shall be acceptable only if it is shown that the airplane embodies features of design which make it impossible to exceed such values in flight.

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(1) The positive maneuvering load factor n for any flight speed up to V_D shall be selected by the applicant, except that it shall not be less than 2.5.

(2) The negative maneuvering load factor shall have a minimum value of −1.0 at all speeds up to V_C, and it shall vary linearly with speed from the value at V_C to zero at V_D.

(b) *Gust load factors.* The airplane shall be assumed to be subjected to symmetrical vertical gusts while in level flight. The resulting limit load factors shall correspond with the conditions prescribed in subparagraphs (1) through (5) of this paragraph. The shape of the gust shall be assumed to be:

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

where:

s=distance penetrated into gust (ft.);
$\bar{C}$=mean geometric chord of wing (ft.);
U_{de}=derived gust velocity referred to in subparagraphs (1) through (3) of this paragraph (fps).

(1) Positive (up) and negative (down) rough air gusts of 66 fps at the speed V_B shall be considered at altitudes between sea level and 20,000 feet. At altitudes above 20,000 feet, it shall be acceptable to reduce the gust velocity linearly from 66 fps at 20,000 feet to 38 fps at 50,000 feet.

(2) Positive and negative gusts of 50 fps at the speed V_C shall be considered at altitudes between sea level and 20,000 feet. At altitudes above 20,000 feet, it shall be acceptable to reduce the gust velocity linearly from 50 fps at 20,000 feet to 25 fps at 50,000 feet.

(3) Positive and negative gusts of 25 fps at the speed V_D shall be considered at altitudes between sea level and 20,000 feet. At altitudes above 20,000 feet, it shall be acceptable to reduce the gust velocity linearly from 25 fps at 20,000 feet to 12.5 fps at 50,000 feet.

(4) Gust load factors shall be assumed to vary linearly between the specified conditions B′ through G′, as shown on the gust envelope of Figure 4b–3.

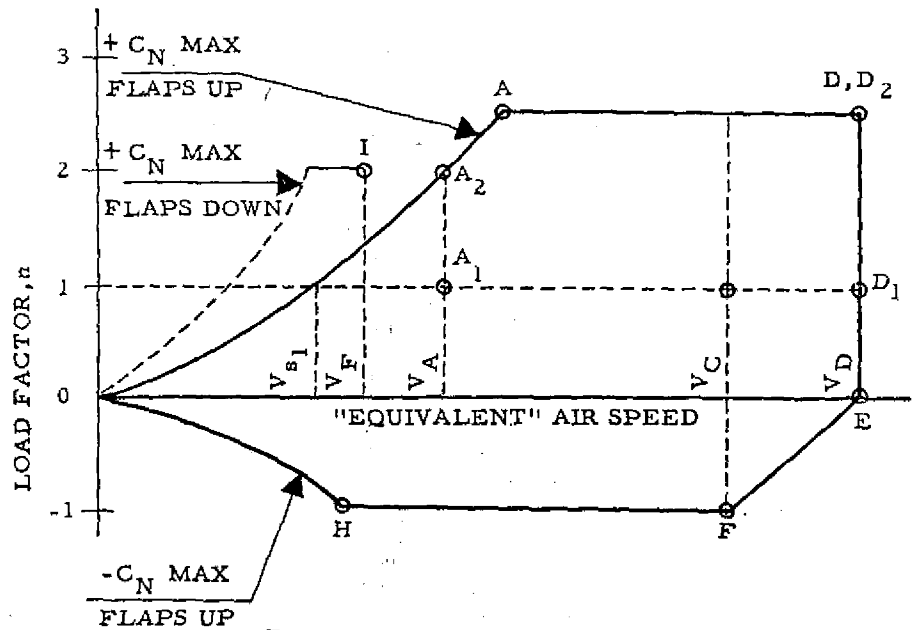


FIGURE 4b–2.—MANEUVERING ENVELOPE.

[Amdt. 4b–11, 24 F.R. 7072, Sept. 1, 1959]

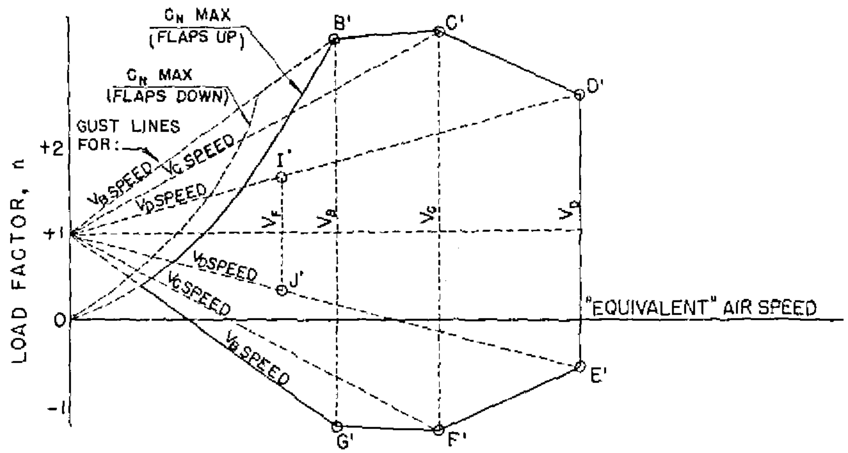


FIGURE 4b–3—Gust envelope.
Load factor vs. velocity, (V–n) diagram.

(5) In the absence of a more rational analysis, the gust load factors shall be computed in accordance with the following formula:

$$n=1+\frac{K_g U_{de} V a}{498(W/S)}$$

where:

$K_g=\frac{0.88\mu_g}{5.3+\mu_g}$=gust alleviation factor;
$\mu_g=\frac{2(W/S)}{\rho\bar{C}ag}$=airplane mass ratio;
U_{de}=derived gust velocities referred to in subparagraphs (1) through (3) of this paragraph (fps);
ρ=density of air (slugs/cu. ft.);
W/S=wing loading (psf);
$\bar{C}$=mean geometric chord (ft.);
g=acceleration due to gravity (ft./sec.^{2});
V=airplane equivalent speed (knots);
a=slope of the airplane normal force coefficient curve C_{N_A} per radian if the gust loads are applied to the wings and horizontal tail surfaces simultaneously by a rational method. It shall be acceptable to use the wing lift curve slope C_L per radian when the gust load is applied to the wings only and the horizontal tail gust loads are treated as a separate condition.

[15 F. R. 3543, June 8, 1950; 15 F. R. 4171, June 29, 1950, as amended by Amdt. 4b–6, 17 F. R. 1089, Feb. 5, 1952; Amdt. 4b–3, 21 F. R. 990, Feb. 11, 1956]

§ 4b.212 Effect of high lift devices.

(a) When flaps are intended for use during takeoff, approach, or landing, the airplane shall be assumed to be subjected to symmetrical maneuvers and gusts within the range determined by the following conditions, at the design flap speeds established for these stages of flight in accordance with § 4b.210(b)(1) and with the flaps in the corresponding positions.

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

(2) Positive and negative 25 fps derived gusts acting normal to the flight path in level flight.

(b) When flaps or similar high lift devices are intended for use in en route conditions (e. g., as speed brakes) the airplane shall be assumed to be subjected to symmetrical maneuvers and gusts, with flaps in the appropriate position at speeds up to the flap design speed chosen for this condition, resulting in limit load factors, within the range determined by the following conditions:

(1) Maneuvering to a positive limit load factor of 2.5.

(2) Positive and negative derived gusts as prescribed in § 4b.211 (b) acting normal to the flight path in level flight.

(c) The airplane shall 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 the following effects as separate conditions:

(1) Propeller slipstream corresponding with maximum continuous power at the design flap speeds V_F, and with take-

off 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).

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b-6, 17 F. R. 1089, Feb. 5, 1952; Amdt. 4b-2, 20 F. R. 5305, July 26, 1955; Amdt. 4b-3, 21 F.R. 991, Feb. 11, 1956; Amdt. 4b-6, 22 F.R. 5564, July 16, 1957; Amdt. 4b-12, 27 F.R. 2990, Mar. 30, 1962]

§ 4b.213 Symmetrical flight conditions.

(a) *Procedure of analysis.* In the analysis of symmetrical flight conditions at least those specified in paragraphs (b), (c), and (d) of this section shall be considered. The following procedure of analysis shall be applicable:

(1) A sufficient number of points on the maneuvering and gust envelopes shall be investigated to insure that the maximum load for each part of the airplane structure is obtained. It shall be acceptable to use a conservative combined envelope for this purpose.

(2) All significant forces acting on the airplane shall be placed in equilibrium in a rational or a conservative manner. The linear inertia forces shall be considered in equilibrium with wing and horizontal tail surface loads, while the angular (pitching) inertia forces shall be considered in equilibrium with wing and fuselage aerodynamic moments and horizontal tail surface loads.

(3) Where sudden displacement of a control is specified, the assumed rate of displacement need not exceed that which actually could be applied by the pilot.

(4) In determining elevator angles and chordwise load distribution in the maneuvering conditions of paragraphs (b) and (c) of this section in turns and pull-ups, account shall be taken of the effect of corresponding pitching velocities.

(b) *Maneuvering balanced conditions.* The maneuvering conditions A through I on the maneuvering envelope (fig. 4b-2) shall be investigated, assuming the airplane to be in equilibrium with zero pitching acceleration.

(c) *Maneuvering pitching conditions.* The following conditions involving pitching acceleration shall be investigated (see figure 4b-2):

(1) *Maximum elevator displacement at speed V_A.* The airplane shall be assumed to be flying in steady level flight (point A_1 on figure 4b-2) and the pitching control suddenly moved to obtain extreme positive pitching (nose up) except as limited by pilot effort in accordance with § 4b.220(a).

(2) *Checked maneuver at speeds between V_A and V_D.* The airplane shall be assumed to be subjected to a checked maneuver from steady level flight (points A_1 *to* D_1 on figure 4b-2) and from the positive load factor (points A_2 to D_2 on figure 4b-2) as follows:

(i) A positive pitching acceleration (nose up), equal to at least the following value, shall be assumed to be attained concurrently with the airplane load factor of unity (points A_1 to D_1 on figure 4b-2) unless it is shown that lesser values could not be exceeded:

$$\frac{39}{V}n\,(n-1.5)\ \text{(radians/sec.}^2\text{)}$$

where n is the positive load factor (see § 4b.211(a)(1)), at the speed under consideration, and V is the airplane equivalent speed, knots.

(ii) A negative pitching acceleration (nose down) equal to at least the following value shall be assumed to be attained concurrently with the airplane positive maneuvering load factor (points A_2 to D_2 on figure 4b-2) unless it is shown that lesser values could not be exceeded:

$$-\frac{26}{V}n\,(n-1.5)\ \text{(radians/sec.}^2\text{)}$$

where n is the positive load factor (see § 4b.211(a)(1)), at the speed under consideration, and V is the airplane equivalent speed, knots.

(3) *Specified control displacement.* In lieu of subparagraph (2) of this paragraph, a checked maneuver based on a rational pitching control motion vs. time profile may be established such that the design limit load factor as defined in § 4b.211(a)(1) will not be exceeded. The airplane response shall result in pitching accelerations not less than those specified in subparagraph (2) unless it is shown that lesser values cannot be exceeded.

(d) *Gust conditions.* The gust conditions B' through J' on figure 4b-3 shall be investigated. The following provisions shall apply:

(1) The air load increment due to a specified gust shall be added to the initial balancing tail load corresponding with steady level flight.

(2) It shall be acceptable to include the alleviating effect of wing down-wash and of the airplane's motion in response to the gust in computing the tail gust load increment.

(3) In lieu of a rational investigation of the airplane response it shall be acceptable to apply the gust factor K_g (see § 4b.211 (b)) to the specified gust intensity for the horizontal tail.

[15 F. R. 3543, June 8, 1950; 15 F. R. 4171, June 29, 1950, as amended by Amdt. 4b-3, 21 F.R. 991, Feb. 11, 1956; Amdt. 4b-11, 24 F.R. 7069, Sept. 1, 1959]

§ 4b.214 Rolling conditions.

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

(a) *Maneuvering.* The following conditions, aileron deflection, and speeds, except as the deflections may be limited by pilot effort (see § 4b.220 (a)), shall be considered in combination with an airplane load factor of zero and of two-thirds of the positive maneuvering factor used in the design of the airplane. In determining the required aileron deflections, the torsional flexibility of the wing shall be taken into account in accordance with § 4b.200 (d).

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

(2) At speed V_A a sudden deflection of the aileron to the stop shall be assumed.

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

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

(b) *Unsymmetrical gusts.* The con dition of unsymmetrical gusts shall b considered by modifying the symmetrica flight conditions B' or C' of figure 4b-3 whichever produces the greater load factor. It shall be assumed that 100 percent of the wing air load acts on one sid of the airplane, and 80 percent acts or the other side.

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b-6, 17 F. R. 1089, Feb. 5, 1952]

§ 4b.215 Yawing conditions.

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

(a) *Maneuvering.* At all speeds from V_{MC} to V_A the following maneuvers shall be considered. In computing the tail loads it shall be acceptable to assume the yawing velocity to be zero.

(1) With the airplane in unaccelerated flight at zero yaw, it shall be assumed that the rudder control is suddenly displaced to the maximum deflection as limited by the control stops or by a 300 lb. rudder pedal force, whichever is critical.

(2) With the rudder deflected as specified in subparagraph (1) of this paragraph it shall be assumed that the airplane yaws to the resulting sideslip angle.

(3) With the airplane yawed to the static sideslip angle corresponding with the rudder deflection specified in subparagraph (1) of this paragraph, it shall be assumed that the rudder is returned to neutral.

(b) *Lateral gusts.* The airplane shall be assumed to encounter derived gusts normal to the plane of symmetry while in unaccelerated flight. The derived gusts and airplane speeds corresponding with conditions B' through J' on Figure 4b-3 as determined by §§ 4b.211 (b) and 4b.212 (a) (2) or § 4b.212 (b) (2) shall be investigated. The shape of the gust shall be as specified in § 4b.211 (b). In the absence of a rational investigation of the airplane's response to a gust, it shall be acceptable to compute the gust loading on the vertical tail surfaces by the following formula:

$$L_t=\frac{K_{g_t}U_{de}Va_tS_t}{498};$$

where:

L_t = vertical tail load (lbs.);

$K_{g_t}=\frac{0.88\mu_{g_t}}{5.3+\mu_{g_t}}$ = gust alleviation factor;

$\mu_{g_t}=\frac{2W}{\rho\overline{C}_t g a_t S_t}\left(\frac{K}{l_t}\right)^2$ = lateral mass ratio;

U_{de} = derived gust velocity (fps);

ρ = air density (*slugs/cu. ft.*);

W = airplane weight (lbs.);

S_t = area of vertical tail (ft.^{2});

$\overline{C}_t$ = mean geometric chord of vertical surface (ft.);

a_t = lift curve slope of vertical tail (per radian);

K = radius of gyration in yaw (ft.);

l_t = distance from airplane C. G. to lift center of vertical surface (ft.);

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

V = airplane equivalent speed (knots).

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b–6, 17 F. R. 1089, Feb. 5, 1952; Amdt. 4b–3, 21 F. R. 991, Feb. 11, 1956; 21 F. R. 1088, Feb. 17, 1956]

§ 4b.216 Supplementary flight conditions.

(a) *Engine torque effects.* Engine mounts and their supporting structures shall be designed for engine torque effects combined with basic flight conditions as described in subparagraphs (1) through (3) of this paragraph. The *limit torque shall be obtained by multiplying* the mean torque by a factor of 1.33 in the case of engines having 5 or more cylinders. For 4, 3, and 2-cylinder engines, the factors shall be 2, 3, and 4, respectively.

(1) The limit torque corresponding with take-off power and propeller speed shall act simultaneously with 75 percent of the limit loads from flight condition A (see fig. 4b–2).

(2) The limit torque corresponding with maximum continuous power and propeller speed shall act simultaneously with the limit loads from flight condition *A* (see fig. 4b–2).

(3) For turbine propeller installations, in addition to the conditions specified in subparagraphs (1) and (2) of this paragraph, the limit torque corresponding with *takeoff power and propeller* speed multiplied by a factor of 1.6 shall be considered to act simultaneously with *1g level flight loads.*

(4) For turbine engine installations, the limit torque load imposed by sudden engine stoppage due to malfunction or structural failure (e. g., compressor jamup) shall be considered in the design of the engine mounts and supporting *structure.*

For turbine propeller installations the limit torque shall be obtained by multiplying the mean torque by *a factor of* 1.25.

(b) *Side load on engine mount.* The limit load factor in a lateral direction for this condition shall be equal to the maximum obtained in the yawing conditions, but shall not be less than either 1.33 or one-third the limit load factor for flight condition *A* (see fig. 4b–2). Engine mounts and their supporting structure shall be designed for this condition which may be assumed independent of other flight conditions.

(c) *Pressurized cabin loads.* When pressurized compartments are provided for the occupants of the airplane, the following requirements shall be met. (See § 4b.373.)

(1) The airplane structure shall have sufficient strength to withstand the flight loads combined with pressure differen*tial loads from zero up to the maximum* relief valve setting. Account shall be taken of the external pressure distribution in flight. Stress concentration and fatigue effects shall be accounted for in the design of pressure cabins (see § 4b.270).

(2) If landings are to be permitted with the cabin pressurized, landing loads shall be combined with pressure differen*tial loads from zero up to the maximum* to be permitted during landing.

(3) The airplane structure shall have sufficient strength to withstand the pressure differential loads corresponding with the maximum relief valve setting multiplied by a factor of 1.33. It shall be acceptable to omit all other loads in this case.

(4) Where a pressurized cabin is separated into two or more compartments by partitions, bulkheads, or floors, the structure supporting the prescribed flight *and ground loads and other structure* the failure of which could interfere with con-

tinued safe flight and landing of the airplane, shall be designed to withstand the effects of sudden release of pressure in any compartment through an opening resulting from the failure or penetration of an external door, window, or windshield panel, or from structural fatigue or penetration of the fuselage in such *compartment* unless it is shown that the probability of failure or penetration is extremely remote. In determining the probability of failure or penetration and probable size of openings, it shall be acceptable to take into account fail-safe features of the design, provided possible *improper* operation of closure devices and inadvertent door openings are also taken into account. It shall be acceptable to take into account pressure relief provided by intercompartment venting. It can be assumed that parts of the airplane, other than the structure specified *in this paragraph, may* be damaged, in which case reasonable design precautions shall be taken to minimize the probability of parts becoming detached which may injure occupants while in their seats.

NOTE: The aforementioned precautions might include, for example, designing internal doors so that they will remain attached to supporting structure even though forced open by differential pressure.

(d) *Unsymmetrical loads due to engine failure.* The airplane shall be designed for the unsymmetrical loads resulting from the failure of the critical engine. Turbopropeller airplanes shall be designed for the conditions prescribed in subparagraphs (1) through (4) of this paragraph in combination with a single malfunction of the propeller drag limiting system (see § 4b.408), taking into account the probable pilot corrective action on the flight controls.

(1) At all speeds between V_{MC} and V_D, the loads resulting from engine power failure due to fuel flow interruption shall be considered as limit loads.

(2) At all speeds between V_{MC} and V_C, the loads resulting from the disconnection of the engine compressor from the turbine or from loss of the turbine blades shall be considered as ultimate loads.

(3) *The time history of the thrust* decay and drag build-up occurring as a result of the prescribed engine failures shall be substantiated by test or other data applicable to the particular engine-propeller combination.

(4) The timing and magnitude of the *probable pilot corrective action* shall be conservatively estimated, considering the characteristics of the particular engine-propeller-airplane combination.

NOTE: It may be assumed that pilot corrective action will be initiated at the time maximum yawing velocity is attained, but not earlier than two seconds after the engine failure. The magnitude of the corrective action may be based on the control forces specified in § 4b.220(a)(1), except that lower *forces may be assumed where it is shown by* analysis or test that such forces will be sufficient to control the yaw and roll resulting from the prescribed engine failure conditions.

(e) *Gyroscopic loads.* The structure supporting the engines shall be designed *for gyroscopic* loads associated with the conditions specified in §§ 4b.213 through 4b.215 with the engines operating at maximum continuous rpm.

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b–2, 20 F. R. 5305, July 26, 1955; Amdt. 4b–3, 21 F.R. 991, Feb. 11, 1956; Amdt. 4b–6, 22 F.R. 5564, July 16, 1957; Amdt. 4b–11, 24 F.R. 7069, Sept. 1, 1959; Amdt. 4b–12, 27 F.R. 2990, Mar. 30, 1962]

§ 4b.217 Speed control devices.

When speed control devices (e.g., spoilers, drag flaps, etc.) are incorporated for use in en route conditions, the following conditions shall apply:

(a) The airplane shall be designed for the symmetrical maneuvers and gusts prescribed in § 4b.211 and the yawing maneuvers and lateral gusts in § 4b.215 with the device extended at all speeds up *to the placard device extended speed.*

(b) When the speed control device incorporates automatic operation or load limiting features, the airplane shall be designed for the maneuver and gust conditions prescribed in paragraph (a) of this section, at the speeds and corresponding device positions which the mechanism permits.

[Amdt. 4b–11, 24 F.R. 7069, Sept. 1, 1959]

CONTROL SURFACE AND SYSTEM LOADS

§ 4b.220 Control surface loads; general.

The control surfaces shall be designed for the limit loads resulting from the flight conditions prescribed in §§ 4b.213 through 4b.215 and the ground gust conditions prescribed in § 4b.226, taking into account the provisions of paragraphs (a) through (e) of this section.

(a) *Effect of pilot effort.* (1) In the control surface flight loading conditions the air loads on the movable surfaces and the corresponding deflections need not exceed those which could be obtained in flight by employing the maximum pilot control forces specified in fig. 4b–5, except that two-thirds of the maximum values specified for the aileron and elevator shall be acceptable when control surface hinge moments are based on reliable data. In applying this criterion, proper consideration shall be given to the effects of servo mechanisms, tabs, and automatic pilot systems in assisting the pilot.

(b) *Effect of trim tabs.* The effect of trim tabs on the main control surface design conditions need be taken into account only in cases where the surface loads are limited by pilot effort in accordance with the provisions of paragraph (a) of this section. In such cases the trim tabs shall be considered to be deflected in the direction which would assist the pilot, and the deflection shall be as follows:

(1) For elevator trim tabs the deflections shall be those required to trim the airplane at any point within the positive portion of the *V-n* diagram (fig. 4b–2), except as limited by the stops.

(2) For aileron and rudder trim tabs the deflections shall be those required to trim the airplane in the critical unsymmetrical power and loading conditions, with appropriate allowance for rigging tolerances.

(c) *Unsymmetrical loads.* Horizontal tail surfaces and the supporting structure shall be designed for unsymmetrical loads arising from yawing and slipstream effects in combination with the prescribed flight conditions.

NOTE: In the absence of more rational data, the following assumptions may be made for airplanes which are conventional in regard to location of propellers, wings, tail surfaces, and fuselage shape: 100 percent of the maximum loading from the symmetrical flight conditions acting on the surface on one side of the plane of symmetry and 80 percent of this loading on the other side. Where the design is not conventional (e. g., where the horizontal tail surfaces have appreciable dihedral or are supported by the vertical tail surfaces), the surfaces and supporting structures may be designed for combined vertical and horizontal surface loads resulting from the prescribed maneuvers.

(d) *Outboard fins.* (1) When outboard fins are carried on the horizontal tail surface, the tail surfaces shall be designed for the maximum horizontal surface load in combination with the corresponding loads induced on the vertical surfaces by end plate effects. Such induced effects need not be combined with other vertical surface loads.

(2) To provide for unsymmetrical loading when outboard fins extend above and below the horizontal surface, the critical vertical surface loading (load per unit area) as determined by the provisions of this section shall also be applied as follows:

(i) 100 percent to the area of the vertical surfaces above (or below) the horizontal surface, and

(ii) 80 percent to the area below (or above) the horizontal surface.

(e) *Loads parallel to hinge line.* Control surfaces and supporting hinge brackets shall be designed for inertia loads acting parallel to the hinge line.

NOTE: In lieu of a more rational analysis the inertia loads may be assumed to be equal to *KW*, where:

K=24 for vertical surfaces,
K=12 for horizontal surfaces,
W=weight of the movable surfaces.

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b–6, 17 F. R. 1089, Feb. 5, 1952]

§ 4b.221 Wing flaps.

Wing flaps and their supporting structure and operating mechanism shall be designed for the critical loads resulting from the conditions prescribed in § 4b.212, taking into account the loads occurring during transition from one flap position and airspeed to another.

[Amdt. 4b–12, 27 F.R. 2991, Mar. 30, 1962]

§ 4b.222 Tabs.

The following shall apply to tabs and their installations:

(a) *Trimming tabs.* Trimming tabs shall be designed to withstand loads arising from all likely combinations of tab setting, primary control position, and airplane speed, obtainable without exceeding the flight load conditions prescribed for the airplane as a whole, when the effect of the tab is being opposed by pilot effort loads up to those specified in § 4b.220 (a).

(b) *Balancing tabs.* Balancing tabs shall be designed for deflections consistent with the primary control surface loading conditions.

(c) *Servo tabs.* Servo tabs shall be designed for all deflections consistent with the primary control surface loading conditions achievable within the pilot maneuvering effort (see § 4b.220 (a)) with due regard to possible opposition from the trim tabs.

[Amdt. 4b–6, 17 F. R. 1090, Feb. 5, 1952]

§ 4b.223 Special devices.

The loading for special devices employing aerodynamic surfaces, such as slots and spoilers, shall be based on test data.

§ 4b.224 Primary flight control systems.

Elevator, aileron, and rudder control systems and their supporting structures shall be designed for loads corresponding with 125 percent of the computed hinge moments of the movable control surface in the conditions prescribed in § 4b.220, subject to the following provisions:

(a) The system limit loads, except the loads resulting from ground gusts (§ 4b.226), need not exceed those which can be produced by the pilot or pilots and by automatic devices operating the controls. Acceptable maximum and minimum pilot loads for elevator, aileron, and rudder controls are shown in figure 4b–5. These pilot loads shall be assumed to act at the appropriate control grips or pads in a manner simulating flight conditions and to be reacted at the attachment of the control system to the control surface horn.

(b) The loads shall in any case be sufficient to provide a rugged system for service use, including considerations of jamming, ground gusts, taxiing tail to wind, control inertia, and friction.

§ 4b.225 Dual primary flight control systems.

(a) When dual controls are provided, the system shall be designed for the pilots operating in opposition, using individual pilot loads equal to 75 percent of those obtained in accordance with § 4b.224, except that the individual pilot loads shall not be less than the minimum loads specified in figure 4b–5.

(b) The control system shall be designed for the pilots acting in conjunction, using individual pilot loads equal to 75 percent of those obtained in accordance with § 4b.224.

§ 4b.226 Ground gust conditions.

The following conditions intended to simulate the loadings on control surfaces due to ground gusts and when taxiing downwind shall be investigated:

(a) The control system between the stops nearest the surfaces and the cockpit controls shall be designed for loads corresponding with the limit hinge moments H of paragraph (b) of this section, except that these loads need not exceed those corresponding with the maxima or figure 4b–5 for each pilot alone, or with 75 percent of these maxima for each pilot when the pilots act in conjunction.

(b) The control system stops nearest the surfaces, the control system locks, and the portions of the systems, if any, between such stops and locks and the control surface horns shall be designed for limit hinge moments H obtained from the following formula:

$H=KcSq$,

where:

H=limit hinge moment (ft. lbs.),
c=mean chord of the control surface aft of the hinge line (ft.),
S=area of the control surface aft of the hinge line (sq. ft.),
q=dynamic pressure (p. s. f.) based on a design speed not less than $10\sqrt{W/S}+10$ (m. p. h.), except that the design speed need not exceed 80 m. p. h.,
K=factor as specified in figure 4b–4.

[15 F. R. 3543, June 8, 1950; 15 F. R. 4171, June 29, 1950, as amended by Amdt. 4b–2, 20 F. R. 5305, July 26, 1955]

§ 4b.227 Secondary control systems.

Secondary controls, such as wheel brake, spoiler, and tab controls, shall be designed for the loads based on the maximum which a pilot is likely to apply to the control in question. The values of figure 4b–6 are considered acceptable.

Surface	K	Position of controls
(a) Aileron	0.75	Control column locked or lashed in mid-position.
(b) Aileron	*±0.50	Ailerons at full throw.
(c), (d) Elevator	*±0.75	(c) Elevator full down. (d) Elevator full up.
(e), (f) Rudder	0.75	(e) Rudder in neutral. (f) Rudder at full throw.

*A positive value of K indicates a moment tending to depress the surface, while a negative value of K indicates a moment tending to raise the surface.

FIGURE 4b-4—Limit hinge moment factor for ground gusts.

[Limit pilot loads (one pilot)]

Control	Maximum load	Minimum load
Aileron:		
Stick	100 lbs.	40 lbs.
Wheel*	80 D in. lbs.**	40 D in. lbs.
Elevator:		
Stick	250 lbs.	100 lbs.
Wheel	300 lbs.	100 lbs.
Rudder	300 lbs.	130 lbs.

*The critical portions of the aileron control system shall be designed for a single tangential force having a limit value equal to 1.25 times the couple force determined from these criteria.

**D=wheel diameter.

FIGURE 4b-5—Pilot control force limits (primary controls).

Control	Limit pilot loads
Miscellaneous: *Crank wheel or lever.	$\frac{1+R}{3}\times 50$ lbs., but not less than 50 lbs. nor more than 150 lbs. (R=radius). (Applicable to any angle within 20° of plane of control.)
Twist	133 in. lbs.
Push-pull	To be chosen by applicant.

**Limited to flap, tab, stabilizer, spoiler, and landing gear operating controls.*

FIGURE 4b-6—Pilot control force limits (secondary *controls*)

[15 F. R. 3543, June 8, 1950; 15 F. R. 4171, June 29, 1950]

GROUND LOADS

§ 4b.230 General.

The limit loads obtained in the conditions specified in §§ 4b.231 through 4b.236 shall be considered as external forces applied to the airplane structure and shall be placed in equilibrium by linear and angular inertia forces in a rational or conservative manner. In applying the specified conditions the provisions of paragraph (a) of this section *shall be complied with.* In addition, for the landing conditions of §§ 4b.231 through 4b.234 the airplane shall be assumed to be subjected to forces and descent velocities prescribed in paragraph (b) of this section. (The basic landing gear dimensional data are given in figure 4b-7.)

(a) *Center of gravity positions.* The critical center of gravity positions within the certification limits shall be selected so that the maximum design loads in each of the landing gear elements are obtained in the landing and the ground handling conditions.

(b) *Load factors, descent velocities, and design weights for landing conditions.* (1) In the landing conditions the *limit vertical inertia load factors at the center of gravity of the airplane shall be* chosen by the applicant, except that they shall not be less than the values which would be obtained in the attitude and subject to the drag loads associated with the particular landing condition, and with the following limit descent velocities and weights:

(i) 10 f. p. s. at the design landing weight, and

(ii) 6 f. p. s. at the design take-off weight.

(2) It shall be acceptable to assume a wing lift not exceeding the airplane weight to exist throughout the landing impact and to act through the center of gravity of the airplane.

(3) The provisions of subparagraphs (1) and (2) of this paragraph shall be predicated on conventional arrangements of main and nose gears, or main and tail gears, and on normal operating techniques. It shall be acceptable to modify the prescribed descent velocities if it is shown that the airplane embodies features of design which make it impos*sible to develop these velocities.* (See § 4b.332 (a) for requirements on energy absorption tests which determine the minimum limit inertia load factors corresponding with the required limit descent velocities.)

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b-3, 21 F. R. 991, Feb. 11, 1956]

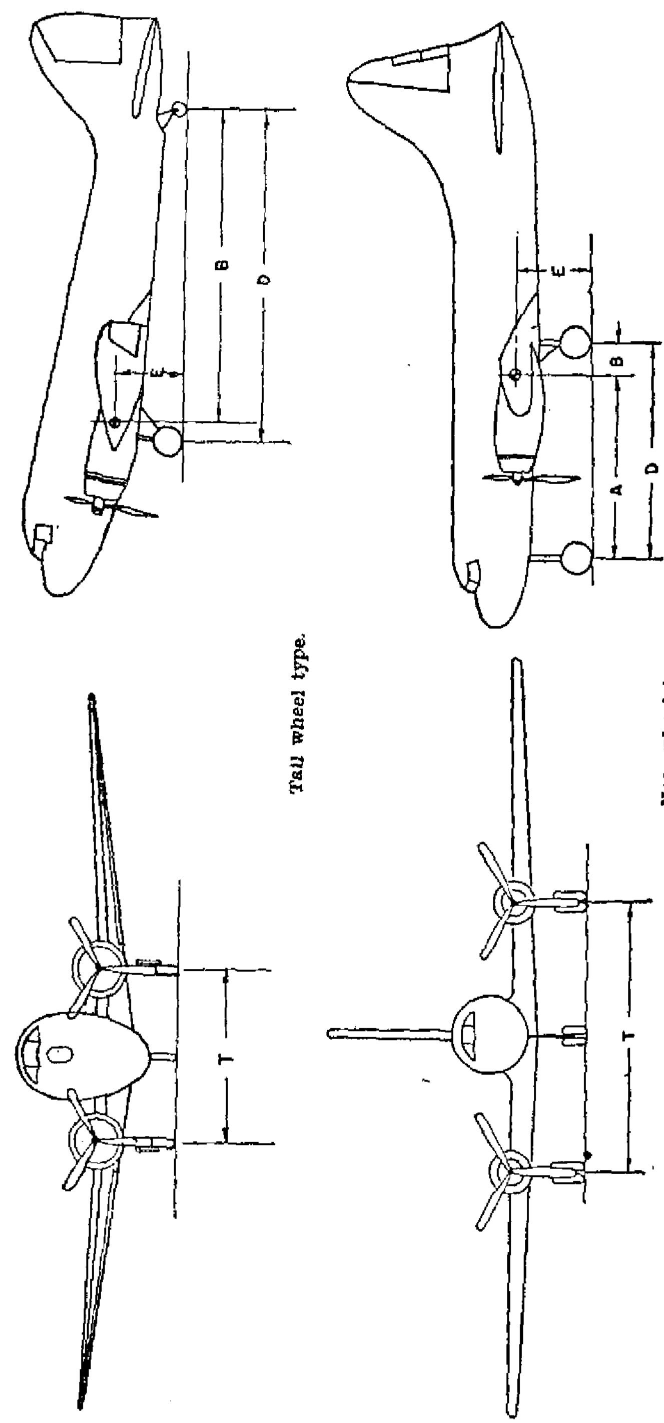


FIGURE 4b-7—Basic landing gear dimension data.

§ 4b.231 Level loading conditions.

(a) *General.* In the level attitude the airplane shall be assumed to contact the ground at forward velocity components parallel to the ground ranging from V_{L_1} to 1.25 V_{L_2} and shall be assumed to be subjected to the load factors prescribed in § 4b.230 (b) (1) where V_{L_1} is equal to V_{s_0} (TAS) at the appropriate landing weight and in standard sea level conditions and where V_{L_2} is equal to V_{s_0} (TAS) at the appropriate landing weight and altitudes in a hot day temperature of 41° F. above standard. When approval of landings downwind exceeding 10 mph is sought, the effect of increased contact speeds shall be investigated. The following three combinations of vertical and drag components shall be considered acting at the axle center line:

(1) *Condition of maximum wheel spin-up load.* Drag components simulating the forces required to accelerate the wheel rolling assembly up to the specified ground speed shall be combined with the vertical ground reactions existing at the instant of peak drag loads. It shall be acceptable to establish the coefficient of friction between the tires and the ground by considering the effects of skidding velocity and tire pressure, except that it need not be greater than 0.8. It shall be acceptable to apply this condition only to the landing gear, directly affected attaching structure, and large mass items (i.e. external fuel tanks, nacelles, etc.).

(2) *Condition of maximum wheel vertical load.* An aft acting drag component not less than 25 percent of the maximum vertical ground reaction shall be combined with the maximum ground reaction of § 4b.230 (b).

(3) *Condition of maximum spring-back load.* Forward-acting horizontal loads resulting from a rapid reduction of the spin-up drag loads shall be combined with the vertical ground reactions at the instant of the peak forward load. It shall be acceptable to apply this condition only to the landing gear, directly affected attaching structure, and large mass items (i. e. external fuel tanks, nacelles, etc.).

(b) *Level landing; tail-wheel type.* The airplane horizontal reference line shall be assumed to be horizontal. The conditions specified in paragraph (a) of this section shall be investigated (See fig. 4b-8.)

(c) *Level landing; nose-wheel type.* The following airplane attitudes shall be considered: (See fig. 4b-8.)

(1) Main wheels shall be assumed to contact the ground with the nose wheel just clear of the ground. The conditions specified in paragraph (a) of this section shall be investigated.

(2) Nose and main wheels shall be assumed to contact the ground simultaneously. Conditions in this attitude need not be investigated if this attitude cannot reasonably be attained at the specified descent and forward velocities. The conditions specified in paragraph (a) of this section shall be investigated, except that in conditions (a) (1) and (a) (3) it shall be acceptable to investigate the nose and main gear separately neglecting the pitching moments due to wheel spin-up and spring-back loads, while in condition (a) (2) the pitching moment shall be assumed to be resisted by the nose gear.

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b-8, 18 F. R. 2214, Apr. 18, 1953; Amdt. 4b-2, 20 F. R. 5306, July 26, 1955; Amdt. 4b-3, 21 F.R. 991, Feb. 11, 1956; Amdt. 4b-11, 24 F.R. 7069, Sept. 1, 1959]

§ 4b.232 Tail-down landing conditions.

In the conditions of paragraphs (a) and (b) of this section the airplane shall be assumed to contact the ground at forward velocity components parallel to the ground, ranging from V_{L_1} to V_{L_2}, where V_{L_1} and V_{L_2} are as indicated in § 4b.231 (a). The load factors prescribed in § 4b.230 (b) (1) shall apply. The combination of vertical and drag components specified in § 4b.231 (a) (1) and (3) shall be considered acting at the main wheel axle centerline.

(a) *Tail-wheel type.* The main and tail wheels shall be assumed to contact the ground simultaneously. (See fig. 4b-9.) Two conditions of ground reaction on the tail wheel shall be assumed to act in the following directions:

(1) Vertical.

(2) Up and aft through the axle at 45° to the ground line.

(b) *Nose-wheel type.* The airplane shall be assumed to be at an attitude corresponding with either the stalling angle or the maximum angle permitting clearance with the ground by all parts of the airplane other than the main wheels, whichever is the lesser. (See fig. 4b-9.)

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b-3, 21 F. R. 991, Feb. 11, 1956]

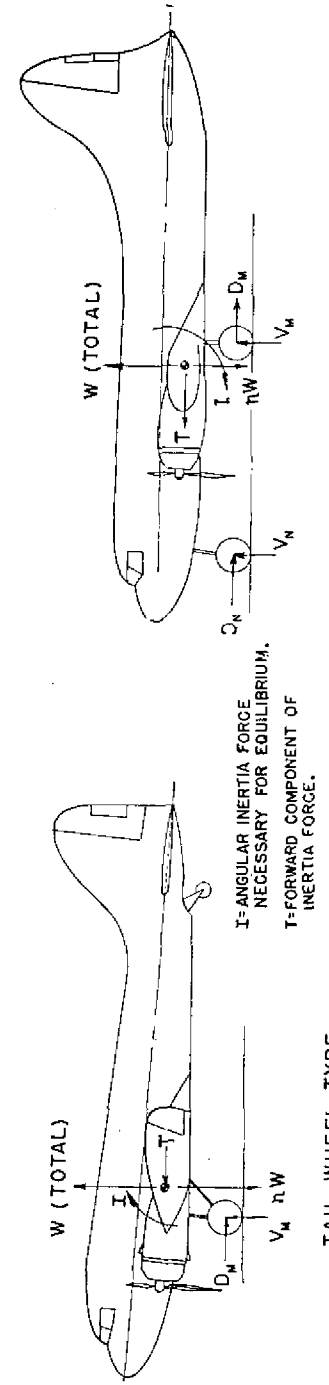


FIGURE 4b-8—Level landing.

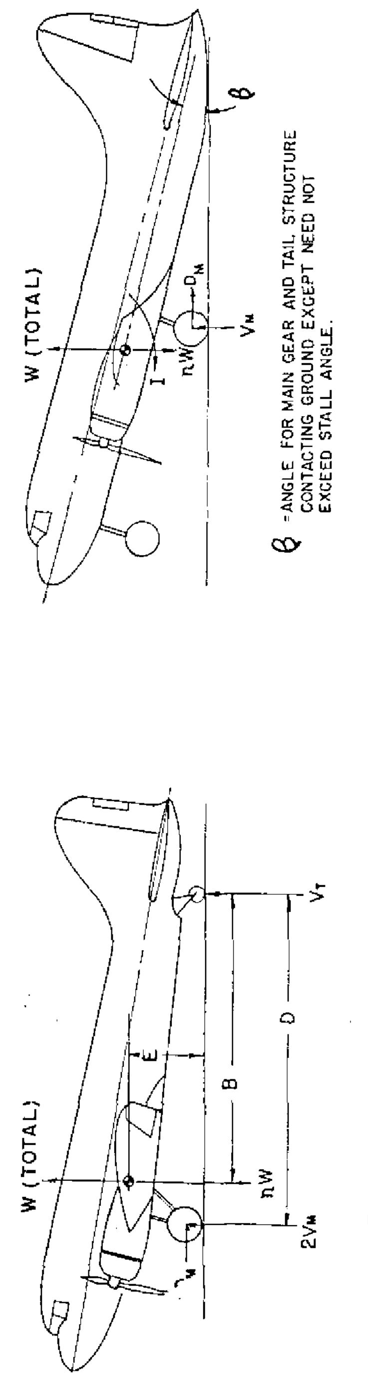
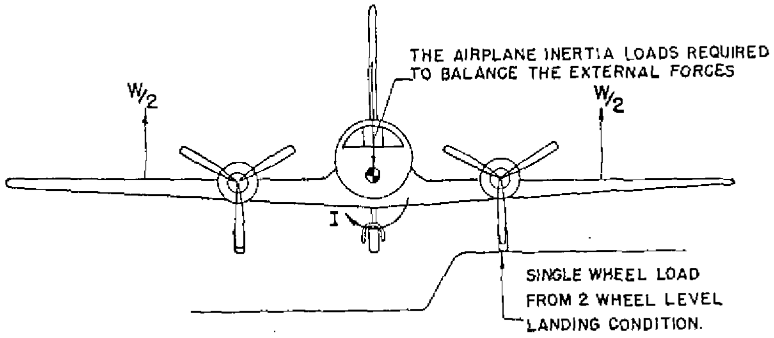


FIGURE 4b-9—Tail down landing.

§ 4b.233 One-wheel landing condition.

The main landing gear on one side of the airplane center line shall be assumed to contact the ground in the level attitude. (See fig. 4b–10.) The ground reactions on this side shall be the same as those obtained in § 4b.231 (a) (2). The unbalanced external loads shall be reacted by inertia of the airplane in a rational or conservative manner.



NOSE OR TAIL WHEEL TYPE

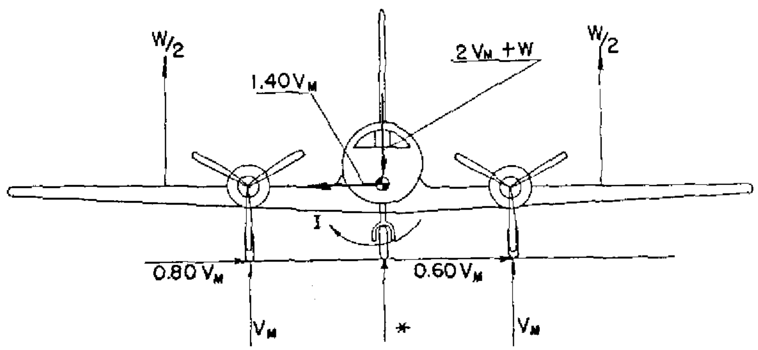
FIGURE 4b–10—One wheel landing.

§ 4b.234 Lateral drift landing condition.

(a) The airplane shall be assumed to be in the level attitude with only the main wheels contacting the ground. (See fig. 4b–11.)

(b) Side loads of 0.8 of the vertical reaction (on one side) acting inward and 0.6 of the vertical reaction (on the other side) acting outward shall be combined with one-half of the maximum vertical ground reactions obtained in the level landing conditions. These loads shall be assumed to be applied at the ground contact point and to be resisted by the inertia of the airplane. It shall be acceptable to assume the drag loads to be zero.

[15 F. R. 3543, June 8, 1950; 15 F. R. 4171, June 29, 1950 as amended by Amdt. 4b–6, 22 F.R. 5564, July 16, 1957]



V_M = one-half the maximum vertical ground reaction obtained at each main gear in the level landing conditions.

* NOSE GEAR GROUND REACTION = 0

NOSE OR TAIL WHEEL TYPE AIRPLANE IN LEVEL ALTITUDE

FIGURE 4b–11—Lateral drift landing.

§ 4b.234a Rebound landing condition.

The landing gear and its supporting structure shall be investigated for the loads occurring during rebound of the airplane from the landing surface. With the landing gear fully extended and not in contact with the ground, a load factor of 20.0 shall act on the unsprung weights of the landing gear. This load factor shall act in the direction of motion of the unsprung weights as they reach their limiting positions in extending with relation to the sprung portions of the landing gear.

[Amdt. 4b–3, 21 F. R. 991, Feb. 11, 1956]

§ 4b.235 *Ground handling conditions.*

The landing gear and airplane structure shall be investigated for the conditions of this section with the airplane at the design take-off weight, unless otherwise prescribed. No wing lift shall be considered. It shall be acceptable to assume the shock absorbers and tires to be deflected to their static position. In the conditions of paragraphs (b) (1) and (2) and (e) (3), it shall be acceptable to use a drag reaction lower than prescribed therein if it is substantiated that an effective drag force of 0.8 times the vertical reaction cannot be attained under any likely loading condition.

(a) *Take-off run.* The landing gear and the airplane structure shall be assumed to be subjected to loads not less than those encountered under conditions described in § 4b.172.

(b) *Braked roll*—(1) *Tail-wheel type.* The airplane shall be assumed to be in the level attitude with all load on the main wheels. The limit vertical load factor shall be 1.2 for the airplane at the design landing weight, and 1.0 for the airplane at the design take-off weight. A drag reaction equal to the vertical reaction multiplied by a coefficient of friction of 0.8 shall be combined with the vertical ground reaction and applied at the ground contact point. (See fig. 4b–12.)

(2) *Nose-wheel type.* The limit vertical load factor shall be 1.2 for the airplane at the design landing weight, and 1.0 for the airplane at the design take-off weight. A drag reaction equal to the vertical reaction multiplied by a coefficient of friction of 0.8 shall be combined with the vertical reaction and applied at the ground contact point of each wheel having brakes. The following two airplane attitudes shall be considered: (See fig. 4b–12.)

(i) The airplane shall be assumed to be in the level attitude with all wheels contacting the ground and the loads distributed between the main and nose gear. Zero pitching acceleration shall be assumed.

(ii) The airplane shall be assumed to be in the level attitude with only the main gear contacting the ground and the pitching moment resisted by angular acceleration.

(c) *Turning.* The airplane in the static position shall be assumed to execute a steady turn by nose gear steering or by application of differential power such that the limit load factors applied at the center of gravity are 1.0 vertically and 0.5 laterally. (See fig. 4b–13.) The side ground reaction of each wheel shall be 0.5 of the vertical reaction.

(d) *Pivoting.* The airplane shall be assumed to pivot about one side of the main gear, the brakes on that side being locked. The limit vertical load factor shall be 1.0 and the coefficient of friction 0.8. The airplane shall be assumed to be in static equilibrium, the loads being applied at the ground contact points. (See fig. 4b–14.)

(e) *Nose-wheel yawing.* (1) A vertical load factor of 1.0 at the airplane center of gravity and a side component at the nose wheel ground contact equal to 0.8 of the vertical ground reaction at that point shall be assumed.

(2) It shall be acceptable to apply the conditions of this subparagraph to the design of only the nose gear, its attaching structure, and the fuselage structure. The airplane shall be assumed to be in static equilibrium with the loads resulting from the application of the brakes on one side of the main gear. The vertical load factor at the center of gravity shall be 1.0. The forward acting load at the airplane center of gravity shall be 0.8 times the vertical load on one main gear. The side and vertical loads at the ground contact point on the nose gear shall be those required for static equilibrium. The side load factor at the airplane center of gravity shall be assumed to be zero. Where this condition results in a nose gear side load in excess of 0.8 times the vertical nose gear load, it shall be acceptable to limit the design nose gear side load to 0.8 times the vertical load with the unbalanced yawing moments assumed to be resisted by aircraft inertia forces.

limit loads shall be equal either to 0.55 times the vertical load at each wheel or to the load developed by 1.2 times the nominal maximum static brake torque, whichever is the lesser. For nosewheel types, the pitching moment shall be balanced by rotational inertia. For tailwheel types, the resultant of the ground reactions shall pass through the center of gravity of the airplane.

(h) *Towing loads.* Towing loads shall be those specified in Figure 4b–26, considering each condition separately. These loads shall be applied at the towing fittings and shall act parallel to the ground. A vertical load factor equal to 1.0 shall be considered acting at the center of gravity. The shock struts and tires shall be in their static positions. The towing load, F_{TOW}, shall be defined as equal to $0.3W_T$ for W_T less than 30,000 pounds, equal to $\frac{6W_T+450{,}000}{70}$ for W_T between 30,000 and 100,000 pounds and equal to $0.15W_T$ for W_T over 100,000 pounds, where W_T is the design maximum take-off weight. For towing points not on the landing gear but located near the plane of symmetry of the airplane, the drag and side tow load components specified for the auxiliary gear shall apply. For tow points located outboard of the main gear, the drag and side tow load components specified for the main gear shall apply. In cases where the specified angle of swivel cannot be obtained, the maximum obtainable angle shall be used.

[15 F. R. 3543, June 8, 1950; 15 F. R. 4171, June 29, 1950, as amended by Amdt. 4b–3, 21 F.R. 992, Feb. 11, 1956; Amdt. 4b–6, 22 F.R. 5564, July 16, 1957; Amdt. 4b–11, 24 F.R. 7069, Sept. 1, 1959; Amdt. 4b–12, 27 F.R. 2991, Mar. 30, 1962]

§ 4b.236 Unsymmetrical loads on multiple-wheel units.

(a) *General.* Multiple-wheel landing gear units shall be assumed to be subjected to the limit ground loads prescribed in this subpart in accordance with the provisions of paragraphs (b) and (c) of this section. A tandem strut gear arrangement shall be considered to be a multiple-wheel unit.

(b) *Distribution of limit loads to wheels; all tires inflated.* The distribution of the limit loads among the wheels of the landing gears shall be established for all landing, taxiing, and ground handling conditions, taking into account the effects of the factors enumerated in subparagraphs (1) through (6) of this paragraph.

(1) Number of wheels and their physical arrangement. In the case of truck type landing gear units, the effects of any see-saw motion of the truck during the landing impact shall be considered in determining the maximum design loads for the fore and aft wheel pairs.

(2) Differentials in tire diameters resulting from a combination of manufacturing tolerances, tire growth, and tire wear. It shall be acceptable to assume a maximum tire-diameter differential equal to ⅔ of the worst combination of diameter variations which is obtained when taking into account manufacturing tolerances, tire growth, and tire wear.

(3) Unequal tire inflation pressure, assuming the maximum variation to be ±5 percent of the nominal tire inflation pressure.

(4) A runway crown of zero and a runway crown having a convex upward shape which may be approximated by a slope of 1½ percent with the horizontal. Runway crown effects shall be considered with the nose gear unit on either slope of the crown.

(5) Airplane attitude.

(6) Structural deflections.

(c) *Deflated tires.* The effect of deflated tires on the structure shall be considered with respect to the loading conditions specified in subparagraphs (1), (2), and (3) of this paragraph taking into account the physical arrangement of the gear components. Consideration shall be given to the deflation of any one tire for all multiple wheel landing gear units and, in addition, to the deflation of any 2 critical tires for landing gear units employing 4 or more wheels per unit. The ground reactions shall be applied to the wheels with inflated tires, except that for multiple-wheel gear units incorporating more than one shock strut, it shall be permissible to use a rational distribution of the ground reactions between the deflated and inflated tires, taking into account the differences in shock strut extensions resulting from a deflated tire.

(1) *Landing conditions.* For one deflated tire and for two deflated tires, the applied load to each gear unit shall be assumed to be 60 percent and 50 percent, respectively, of the limit load applied to each gear for each of the prescribed landing conditions except that, for the drift landing condition of § 4b.234, 100

percent of the vertical load shall be applied.

(2) *Taxiing and ground handling conditions.* For one deflated tire and for two deflated tires, the applied side and/or drag load factor at the center of gravity shall be the most critical value up to 50 percent and 40 percent, respectively, of the limit side and/or drag load factors corresponding with the most severe condition resulting from consideration of the prescribed taxiing and ground handling conditions except that, for the braked roll conditions of § 4b.235 (b) (1) and (2) (ii), the drag loads on each inflated tire shall not be less than those at each tire for the symmetrical load distribution with no deflated tires. For one and two deflated tires the vertical load factor at the center of gravity shall be 60 percent and 50 percent, respectively, of the factor with no deflated tires except that it shall not be less than 1 g. Pivoting need not be considered.

(3) *Towing conditions.* For one deflated tire and for two deflated tires, the towing load, F_{TOW}, shall be 60 percent and 50 percent, respectively, of the load prescribed.

NOTE: In determining the total load on a gear unit with respect to the provisions of paragraphs (b) and (c) of this section, the transverse shift in the load centroid, due to unsymmetrical load distribution on the wheels, is normally neglected.

[Amdt. 4b–6, 22 F.R. 5564, July 16, 1957, as amended by Amdt. 4b–8, 23 F.R. 2590, Apr. 19, 1958; Amdt. 4b–11, 24 F.R. 7069, Sept. 1, 1959]

WATER LOADS

§ 4b.250 General.

The structure of hull and float type seaplanes shall be designed for water loads developed during take-off 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. Unless a more rational analysis of the water loads is performed, the requirements of §§ 4b.251 through 4b.258 shall apply.

[Amdt. 4b–6, 17 F. R. 1090, Feb. 5, 1952]

§ 4b.250–1 Water loads; alternate standards (FAA policies which apply to §§ 4b.10 and 4b.250).

ANC–3 provides a level of safety equivalent to, and may be applied in lieu of, § 4b.250.

[Supp. 16, 16 F. R. 1630, Feb. 16, 1951, as amended by Supp. 20, 17 F. R. 10102, Nov. 7, 1952]

§ 4b.251 Design weights and center of gravity positions.

(a) *Design weights.* The water load requirements shall be complied with at all operating weights up to the design landing weight except that for the take-off condition prescribed in § 4b.255 the design take-off weight shall be used.

(b) *Center of gravity positions.* The critical center of gravity positions within the limits for which certification is sought shall be considered to obtain maximum design loads for each part of the seaplane structure.

[Amdt. 4b–6, 17 F. R. 1090, Feb. 5, 1952]

§ 4b.252 Application of loads.

(a) The seaplane as a whole shall be assumed to be subjected to the loads corresponding with the load factors specified in § 4b.253, except as otherwise prescribed. In applying the loads resulting from the load factors prescribed in § 4b.253, it shall be permissible to distribute the loads over the hull 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 § 4b.256 (b).

(b) For twin float seaplanes, each float shall be treated as an equivalent hull on a fictitious seaplane having a weight equal to one-half the weight of the twin float seaplane.

(c) Except in the take-off condition of § 4b.255, the aerodynamic lift on the seaplane during the impact shall be assumed to be ⅔ of the weight of the seaplane.

[Amdt. 4b–6, 17 F. R. 1090, Feb. 5, 1952]

§ 4b.253 Hull and main float load factors.

Water reaction load factors shall be computed as follows:

For the step landing case:

$$n_W=\frac{C_1 V_{s_0}^{2}}{\tan^{2/3}\beta W^{1/3}};$$

For the bow and stern landing cases:

$$n_W=\frac{C_1 V_{s_0}^{2}}{\tan^{2/3}\beta W^{1/3}}\times\frac{K_1}{(1+r_x^{2})^{2/3}};$$

where:

n_W = water reaction load factor (water reaction divided by the seaplane weight);

C_1 = empirical seaplane operations factor equal to 0.009, except that this factor shall not be less than that necessary to obtain the minimum value of step load factor of 2.33;

V_{s_0}=seaplane stalling speed (mph) with landing flaps extended in the appropriate position and with no slipstream effect;

β=angle of dead rise at the longitudinal station at which the load factor is being determined (see fig. 4b–15a);

W=seaplane design landing weight in pounds;

K_1=empirical hull station weighing factor. (See fig. 4b–15b.) For a twin float seaplane, in recognition of the effect of flexibility of the attachment of the floats to the seaplane, it shall be acceptable to reduce the factor K_1 at the bow and stern to 0.8 of the value shown in figure 4b–15b. This reduction shall not apply to the float design but only to the design of the carry-through and seaplane structure;

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.

[Amdt. 4b–6, 17 F.R. 1090, Feb. 5, 1952]

§ 4b.254 Hull and main float landing conditions.

(a) *Symmetrical step landing.* The limit water reaction load factor shall be in accordance with § 4b.253. The resultant water load shall be applied at the keel through the center of gravity perpendicularly to the keel line.

(b) *Symmetrical bow landing.* The limit water reaction load factor shall be in accordance with § 4b.253. The resultant water load shall be applied at the keel ⅕ of the longitudinal distance from the bow to the step, and shall be directed perpendicularly to the keel line.

(c) *Symmetrical stern landing.* The limit water reaction load factor shall be in accordance with § 4b.253. The resultant water load shall be applied at the keel at a point 85 percent of the longitudinal distance from the step to the stern post, and shall be directed perpendicularly to the keel line.

(d) *Unsymmetrical landing; hull type and single float seaplanes.* Unsymmetrical step, bow, and stern landing conditions shall be investigated. The loading for each condition shall consist 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. (See paragraphs (a), (b), and (c) of this section.) The point of application and direction of the upward component of the load shall be the same as that in the symmetrical condition, and the point of application of the side component shall be at the same longitudinal station as the upward component but directed inward perpendicularly to the plane of symmetry at a point midway between the keel and chine lines.

(e) *Unsymmetrical landing; twin float seaplanes.* The unsymmetrical loading shall consist 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 obtained in accordance with § 4b.253. The side load shall be 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.

[Amdt. 4b–6, 17 F. R. 1091, Feb. 5, 1952]

§ 4b.255 Hull and main float take-off condition.

The provisions of this section shall apply to the design of the wing and its attachment to the hull or main float. The aerodynamic wing lift shall be assumed to be zero. A downward inertia load shall be applied and shall correspond with the following load factor:

$$n=\frac{C_{TO}\,V_{s_1}^{\,2}}{\tan^{2/3}\beta\,W^{1/3}};$$

where:

n= inertia load factor;

C_{TO}= empirical seaplane operations factor equal to 0.003;

V_{s_1}= seaplane stalling speed (mph) at the design take-off weight with the flaps extended in the appropriate take-off position;

β= angle of dead rise at the main step (degrees);

W= seaplane design take-off weight in pounds.

[Amdt. 4b–6, 17 F. R. 1091, Feb. 5, 1952]

§ 4b.256 Hull and main float bottom pressures.

The provisions of this section shall apply to the design of the hull and main float structure, including frames and bulkheads, stringers, and bottom plating. In the absence of more rational data, the pressures and distributions shall be as follows:

(a) *Local pressures.* The following pressure distributions are applicable for the design of the bottom plating and stringers and their attachments to the supporting structure. The area over which these pressures are applied shall be such as to simulate pressures occurring during high localized impacts on the hull or float, and need not extend over an area which would induce critical stresses in the frames or in the overall structure:

(1) *Unflared bottom.* The pressure at the keel (psi) shall be computed as follows:

$$P_k=C_2\frac{K_2\,V_{s_1}^{\,2}}{\tan\beta_k};$$

where:

P_k= pressure at the keel;

C_2= 0.0016;

K_2= hull station weighing factor (see fig. 4b–15b);

V_{s_1}=seaplane stalling speed (mph) at the design take-off weight with flaps extended in the appropriate take-off position;

β_k= angle of dead rise at keel (see fig. 4b–15a).

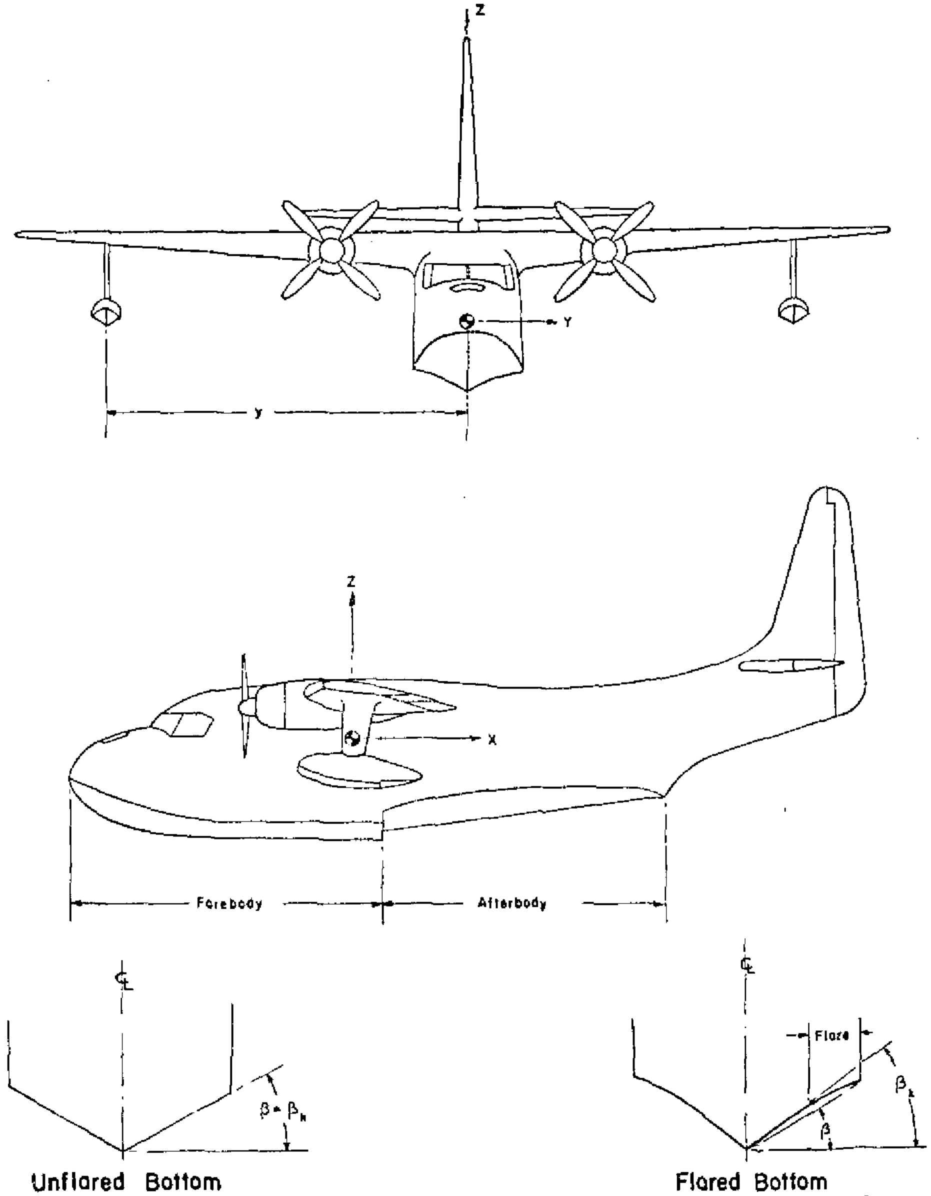


FIGURE 4b–15a—Pictorial definition of angles, dimensions, and directions on a seaplane.

The pressure at the chine shall be 0.75 P_k, and the pressures between the keel and chine shall vary linearly. (See fig. 4b–15c.)

(2) *Flared bottom.* The pressure distribution for a flared bottom shall be that for an unflared bottom prescribed in subparagraph (1) of this paragraph, except that the pressure at the chine shall be computed as follows:

$$P_{ch}=C_3\frac{K_2 V_{s_1}^{2}}{\tan\beta};$$

where:

P_{ch} = pressure at the chine;
C_3 = 0.0012;
K_2 = hull station weighing factor (see fig. 4b–15b);
V_{s_1} = seaplane stalling speed (mph) at the design take-off weight with flaps extended in the appropriate take-off position;
β = angle of dead rise at appropriate station.

The pressure at the beginning of the flare shall be the same as for an unflared bottom, and the pressure between the chine and the beginning of the flare shall vary linearly. (See fig. 4b–15c.)

(b) *Distributed pressures.* The following distributed pressures are applicable for the design of the frames, keel, and chine structure. These pressures shall be uniform and shall be applied simultaneously over the entire hull or main float bottom. The loads so obtained shall 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.

(1) *Symmetrical.* The symmetrical pressures shall be computed as follows:

$$P=C_4\frac{K_2 V_{s_0}^{2}}{\tan\beta};$$

where:

P = pressure;
C_4 = 0.078C_1 (for C_1 see § 4b.253);
K_2 = hull station weighing factor (see fig. 4b–15b);
V_{s_0} = seaplane stalling speed (mph) with landing flaps extended in the appropriate position and with no slipstream effect;
β = angle of dead rise at appropriate station.

(2) *Unsymmetrical.* The unsymmetrical pressure distribution shall consist of the pressures prescribed in subparagraph (1) of this paragraph on one side of the hull or main float center line and one-half of that pressure on the other side of the hull or main float center line. (See fig. 4b–15c.)

[Amdt. 4b–6, 17 F. R. 1091, Feb. 5, 1952]

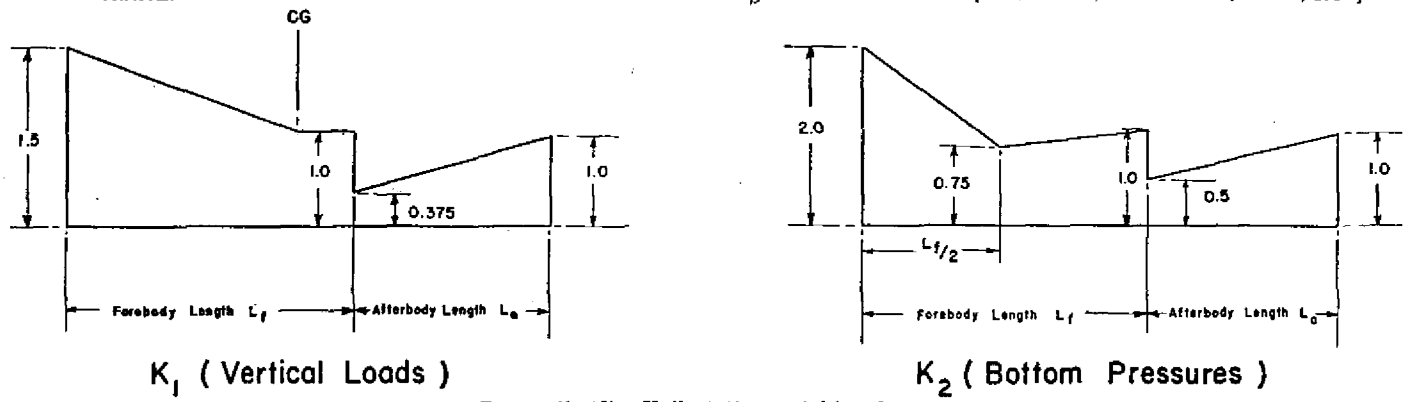


FIGURE 4b–15b—Hull station weighing factor.

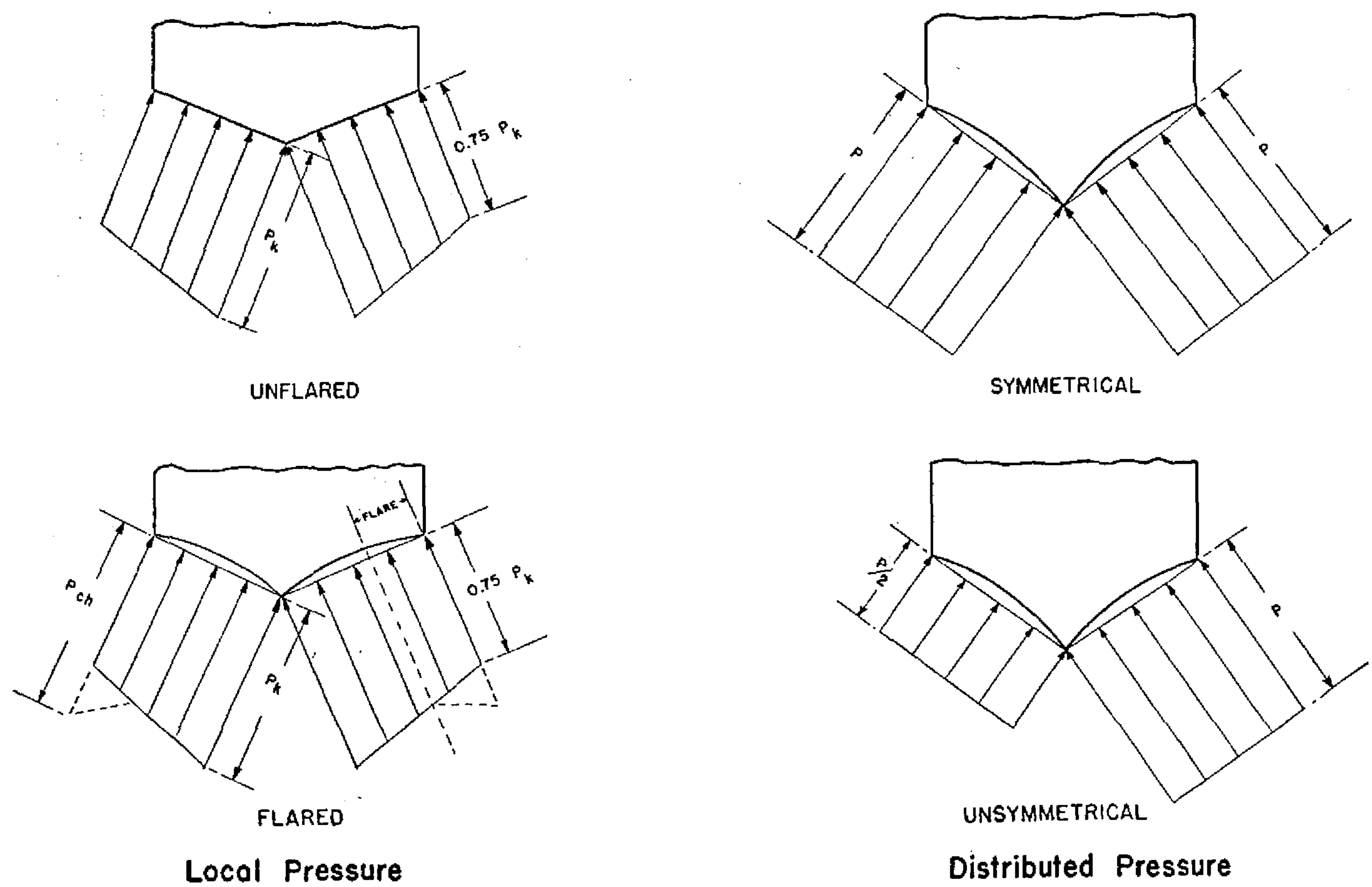


FIGURE 4b–15c—Transverse pressure distributions.

Title 14—Aeronautics and Space § 4b.256

Tow point	Position	Load		
		Magnitude	No.	Direction
Main gear		0.75 F_{TOW} per main gear unit.	1	Forward, parallel to drag axis.
			2	Forward, at 30° to drag axis.
			3	Aft, parallel to drag axis.
			4	Aft, at 30° to drag axis.
Auxiliary gear	Swiveled forward	1.0 F_{TOW}	5	Forward.
			6	Aft.
	Swiveled aft	do	7	Forward.
			8	Aft.
	Swiveled 45° from forward	0.5 F_{TOW}	9	Forward, in plane of wheel.
			10	Aft, in plane of wheel.
	Swiveled 45° from aft	do	11	Forward, in plane of wheel.
			12	Aft, in plane of wheel.

BALANCING FORCES

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

The towing loads at the auxiliary gear and the drag components of the towing loads at the main gear are reacted in each of the following ways:

a. Reaction applied at the axle of the wheel to which load is applied, this reaction having a maximum value equal to the vertical reaction. Airplane inertia is applied as required for equilibrium.

b. The loads reacted by airplane inertia.

FIGURE 4b–26—Towing loads.

§ 4b.257 Auxiliary float loads.

Auxiliary floats, their attachments, and supporting structure shall be designed for the following conditions. In the cases specified in paragraphs (a), (b), (c), and (d) of this section it shall be acceptable to distribute the prescribed water loads over the float bottom to avoid excessive local loads, using bottom pressures not less than those prescribed in paragraph (f) of this section.

(a) *Step loading.* The resultant water load shall 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 shall be perpendicular to the keel. The resultant limit load shall be computed as follows, except that the value of L need not exceed three times the weight of the displaced water when the float is completely submerged:

$$L=\frac{C_5\ V_{s_0}^2\ W^{2/3}}{\tan^{2/3}\beta_s\ (1+r_y^2)^{2/3}};$$

where:

L = limit load;

C_5 = 0.004;

V_{s_0} = seaplane stalling speed (mph) with landing flaps extended in the appropriate position and with no slipstream effect;

W = seaplane design landing weight in pounds;

β_s = angle of dead rise at a station ¾ of the distance from the bow to the step, but need not be less than 15 degrees;

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.

(b) *Bow loading.* The resultant limit load shall 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 shall be perpendicular to the tangent to the keel line at that point. The magnitude of the resultant load shall be that specified in paragraph (a) of this section.

(c) *Unsymmetrical step loading.* The resultant water load shall consist of a component equal to 0.75 times the load specified in paragraph (a) of this section and a side component equal to 0.25 tan β times the load specified in paragraph (a) of this section. The side load shall be applied perpendicularly to the plane of symmetry of the float at a point midway between the keel and the chine.

(d) *Unsymmetrical bow loading.* The resultant water load shall consist of a component equal to 0.75 times the load specified in paragraph (b) of this section and a side component equal to 0.25 tan β times the load specified in paragraph (b) of this section. The side load shall be applied perpendicularly to the plane of symmetry at a point midway between the keel and the chine.

(e) *Immersed float condition.* The resultant load shall 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 shall be as follows:

$$\text{vertical}=\rho g V$$

$$\text{aft}=C_x \rho V^{2/3}\ (KV_{s_0})^2$$

$$\text{side}=C_y\frac{\rho}{2}V^{2/3}\ (KV_{s_0})^2$$

where:

ρ = mass density of water;

V = volume of float;

C_x = coefficient of drag force, equal to 0.10;

C_y = coefficient of side force, equal to 0.08;

K = 0.8, except that lower values shall be acceptable if it is shown that the floats are incapable of submerging at a speed of 0.8 V_{s_0} in normal operations;

V_{s_0} = seaplane stalling speed (mph) with landing flaps extended in the appropriate position and with no slipstream effect.

(f) *Float bottom pressures.* The float bottom pressures shall be established in accordance with § 4b.256 (a) and (b). The angle of dead rise to be used in determining the float bottom pressures shall be as defined in paragraph (a) of this section.

[Amdt. 4b–6, 17 F. R. 1092, Feb. 5, 1952]

§ 4b.258 Seawing loads.

Seawing design loads shall be based on applicable test data.

EMERGENCY LANDING CONDITIONS

§ 4b.260 General.

The following requirements deal with emergency conditions of landing on land or water in which the safety of the occupants shall be considered, although it is accepted that parts of the airplane may be damaged.

(a) The structure shall be designed to give every reasonable probability that all of the occupants, if they make proper use of the seats, belts, and other provisions made in the design (see § 4b.358), will escape serious injury in the event of a minor crash landing (with wheels up if the airplane is equipped with retractable landing gear) in which the occupants experience the following ultimate inertia forces relative to the surrounding structure:

(1) Upward ----- 2.0g (Downward-- 4.5g)
(2) Forward----- 9.0g
(3) Sideward---- 1.5g

(b) The use of a lesser value of the downward inertia force specified in paragraph (a) of this section shall be acceptable if it is shown that the airplane structure can absorb the landing loads corresponding with the design landing weight and an ultimate descent velocity of 5 f. p. s. without exceeding the value chosen.

(c) The inertia forces specified in paragraph (a) of this section shall be applied to all items of mass which would be apt to injure the passengers or crew if such items became loose in the event of a minor crash landing, and the supporting structure shall be designed to restrain these items.

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b–6, 17 F. R. 1093, Feb. 5, 1952]

§ 4b.261 Structural ditching provisions.

(For structural strength considerations of ditching provisions see § 4b.361 (c).)

[Amdt. 4b–8, 18 F. R. 2214, Apr. 18, 1953]

FATIGUE EVALUATION

§ 4b.270 Fatigue evaluation of flight structure.

The strength, detail design, and fabrication of those portions of the airplane's *flight structure* in which fatigue may be critical shall be evaluated in accordance with the provisions of either paragraph (a) or (b) of this section.

(a) *Fatigue strength.* The structure shall be shown by analysis and/or tests to be capable of withstanding the repeated loads of variable magnitude expected in service. The provisions of subparagraphs (1) through (3) of this paragraph shall apply.

(1) Evaluation of fatigue shall involve the following:

(i) Typical loading spectrum expected in service;

(ii) Identification of principal structural elements and detail design points, the fatigue failure of which could cause catastrophic failure of the aircraft; and

(iii) An analysis and/or repeated load tests of principal structural elements and detail design points, identified in subdivision (ii) of this subparagraph;

NOTE: Usually tests of principal structural elements include major fittings, samples of joints, spar cap strips, skin units, and other representative sections of the flight structure.

(2) It shall be acceptable to utilize the service history of airplanes of similar structural design, taking due account of

differences in operating conditions and procedures.

(3) When circumstances require substantiation of the pressure cabin by fatigue tests, the cabin or representative portions of it shall be cycle-pressure tested, utilizing the normal operating pressure together with the effects of external aerodynamic pressure combined with the flight loads. It shall be acceptable to represent the effects of flight loads by an increased cabin pressure, or to omit the flight loads if they are shown to have no significant effect upon fatigue.

(b) *Fail safe strength.* It shall be shown by analysis and/or tests that catastrophic failure or excessive structural deformation, which could adversely affect the flight characteristics of the airplane, are not probable after fatigue failure or obvious partial failure of a single principal structural element. After such failure, the remaining structure shall be capable of withstanding static loads corresponding with the flight loading condition specified in subparagraphs (1) through (4) of this paragraph. These loads shall be multiplied by a factor of 1.15 unless the dynamic effects of failure under static load are otherwise taken into consideration. In the case of a pressure cabin, the normal operating pressures combined with the expected external aerodynamic pressures shall be applied simultaneously with the flight loading conditions specified in this paragraph.

(1) An ultimate maneuver load factor of 2.0 at V_C.

(2) Gust loads as specified in §§ 4b.211 (b) and 4b.215(b), except that these gust loads shall be considered to be ultimate and the gust velocities shall be as follows:

(i) At speed V_B, 49 fps from sea level to 20,000 feet altitude, thereafter decreasing linearly to 28 fps at 50,000 feet altitude.

(ii) At speed V_C, 33 fps from sea level to 20,000 feet altitude, thereafter decreasing linearly to 16.5 fps at 50,000 feet altitude.

(iii) At speed V_D, 15 fps from sea level to 20,000 feet altitude, thereafter decreasing linearly to 6 fps at 50,000 feet altitude.

(3) Eighty percent of the limit loads resulting from the conditions specified in § 4b.220 (c). These loads shall be considered to be ultimate.

(4) Eighty percent of the limit maneuvering loads resulting from the conditions specified in § 4b.215 (a), except that the load need not exceed 100 percent of the critical load obtained in compliance with the provisions of § 4b.215 (a) using a pilot effort of 180 pounds. This load shall be considered to be ultimate.

[Amdt. 4b–3, 21 F.R. 992, Feb. 11, 1956, as amended by Amdt. 4b–6, 22 F.R. 5564, July 16, 1957; Amdt. 4b–8, 23 F.R. 2590, Apr. 19, 1958; Amdt. 4b–12, 27 F.R. 2991, Mar. 30, 1962]

§ 4b.270–1 Flight structure for fatigue evaluation (FAA interpretations which apply to § 4b.270).

The term "flight structure" as applied to fatigue evaluation is defined as those portions of the airplane's structure failure of which could result in catastrophic failure of the aircraft and includes the wings, fixed and movable control surfaces, fuselage, and their related primary attachments.

[Supp. 38, 23 F. R. 3031, May 7, 1958]

§ 4b.270–2 Fatigue evaluation, general (FAA policies which apply to § 4b.270).

The applicant should submit to the FAA a report outlining the procedures and the substantiating analyses and tests he proposes to follow in showing compliance with the fatigue evaluation requirements of § 4b.270. Typical procedures which may be used as guidance in the fatigue evaluation of the structures are discussed in Appendix H[1] to the Civil Aeronautics Manual 4b.

[Supp. 38, 23 F. R. 3031, May 7, 1958]

§ 4b.271 Fatigue evaluation of landing gear.

The strength, detail design, and fabrication of those portions of the landing gear and its attachment fittings in which fatigue may be critical shall be evaluated in accordance with the provisions of either paragraph (a) or (b) of this section.

(a) The fatigue strength of the structure shall be evaluated and, when indicated by such evaluation, inspection or other procedures shall be established to prevent catastrophic fatigue failure. The evaluation shall include the loading spectrum expected in service and the identification and analysis or repeated load testing of the principal structural elements and detail design points where catastrophic fatigue failure could occur.

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

It shall be acceptable to utilize the service history of airplanes of similar structural design, taking due account of differences in operating conditions and procedures.

(b) It shall be shown by analysis or tests that catastrophic failure is not probable after fatigue failure or obvious partial failure of a single principal structural element. After such failure the remaining structure shall be capable of withstanding static loads corresponding with 80 percent of the limit loads resulting from the conditions prescribed in § 4b.230. These static loads shall be considered ultimate loads.

[Amdt. 4b–12, 27 F.R. 2991, Mar. 30, 1962]

Subpart D—Design and Construction

GENERAL

§ 4b.300 Scope.

The airplane shall not incorporate design features or details which experience has shown to be hazardous or unreliable. The suitability of all questionable design details or parts shall be established by tests.

§ 4b.300–1 Turnbuckle safetying (FAA policies which apply to § 4b.300).

The procedure outlined in § 4b.329–2 should be followed in safetying turnbuckles.

[Supp. 25, 20 F. R. 2278, Apr. 8, 1955]

§ 4b.301 Materials.

The suitability and durability of all materials used in the airplane structure shall be established on the basis of experience or tests. All materials used in the airplane structure shall conform to approved specifications which will insure their having the strength and other properties assumed in the design data.

§ 4b.301–1 Acceptability of materials (FAA policies which apply to § 4b.301).

(a) Materials conforming to established industry or military specifications or to Technical Standard Orders issued by the Administrator are acceptable for use on transport category airplanes. Where new or improved materials are used or where the materials are not covered by specifications sufficient information and data should be submitted to the Administrator to enable him to assess the suitability of the material. In all cases it is the responsibility of the applicant to demonstrate the adequacy of the materials employed.

[Supp. 25, 20 F. R. 2278, Apr. 8, 1955]

§ 4b.302 Fabrication methods.

The methods of fabrication employed in constructing the airplane structure shall be such as to produce a consistently sound structure. When a fabrication process such as gluing, spot welding, or heat treating requires close control to attain this objective, the process shall be performed in accordance with an approved process specification.

§ 4b.303 Standard fastenings.

All bolts, pins, screws, and rivets used in the structure shall be of an approved type. The use of an approved locking device or method is required for all such bolts, pins, and screws. Self-locking nuts shall not be used on bolts which are subject to rotation in operation.

§ 4b.304 Protection.

(a) All members of the structure shall be suitably protected against deterioration or loss of strength in service due to weathering, corrosion, abrasion, or other causes.

(b) Provision for ventilation and drainage of all parts of the structure shall be made where necessary for protection.

(c) In seaplanes, special precautions shall be taken against corrosion from salt water, particularly where parts made from different metals are in close proximity.

§ 4b.305 Inspection provisions.

Means shall be provided to permit the close examination of those parts of the airplane which require periodic inspection, adjustment for proper alignment and functioning, and lubrication of moving parts.

§ 4b.306 Material strength properties and design values.

(a) Material strength properties shall be based on a sufficient number of tests of material conforming to specifications to establish design values on a statistical basis.

(b) The design values shall be so chosen that the probability of any structure being understrength because of material variations is extremely remote. The effects of temperature on allowable stresses used for design in an essential component or structure shall be considered where thermal effects are significant under normal operating conditions.

(c) Values contained in MIL-HDBK-5, MIL-HDBK-17 Part I, ANC-17 Part II, ANC-18, MIL-HDBK-23 Part I, and ANC-23 Part II shall be used unless shown to be inapplicable in a particular case.

NOTE: MIL-HDBK-5, "Strength of Metal *Aircraft Elements*"; MIL-HDBK-17, "Plastics for Flight Vehicles, Part I—Reinforced Plastics"; ANC-17, "Plastics for Aircraft, Part II—Transparent Glazing Materials"; ANC-18, "Design of Wood Aircraft Structures"; MIL-HDBK-23, "Composite Construction for Flight Vehicles, Part I—Fabrication Inspection Durability and Repair"; and ANC-23, "Sandwich Construction for Aircraft, Part II—Material Properties and Design Criteria", are published by the Department of Defense and the Federal Aviation Agency and may be obtained from the Superintendent of Documents, Government Printing Office, Washington 25, D.C.

(d) The strength, detail design, and fabrication of the structure shall be such as to minimize the probability of disastrous fatigue failure. (See also § 4b.270.)

NOTE: Points of stress concentration are one of the main sources of fatigue failure.

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b-6, 17 F. R. 1093, Feb. 6, 1952; Amdt. 4b-8, 18 F. R. 2214, Apr. 18, 1953; Amdt. 4b-3, 21 F.R. 992, Feb. 11, 1956; Amdt. 4b-12, 27 F.R. 2991, Mar. 30, 1962]

§ 4b.306-1 Material strength properties (FAA policies which apply to § 4b.306(c)).

(a) In the case of structures where the applied loads are eventually distributed through a single member within an assembly, the failure of which would result in the loss of the structural integrity of the component involved, the guaranteed minimum design mechanical properties ("A" values) listed in MIL-HDBK-5[96] should be used for design.

(b) Redundant structures wherein failure of individual elements would result in the applied load being safely distributed to other load carrying members, may be designed on the basis of the "90 percent probability" ("B" values).

(c) When strength testing is employed to establish design allowables, such as in the case of sheet-stiffener compression tests, the test results should be reduced through use of a materials correction factor to values which would be met by material having the design allowable material properties for the part under consideration. MIL-HDBK-5 outlines methods of accomplishing this reduction but these are by no means considered as the only methods available.

(d) Use of design values greater than the guaranteed minimums is permissible in applications where only guaranteed minimum values are normally permitted provided that the higher values are substantiated by "premium selection" of the material. These increased design allowables will be acceptable providing that a specimen or specimens of each individual item are tested prior to its use, to assure that the strength properties of the particular item will equal or exceed the properties to be used in design. Such quality control should also be exercised for the manufacture of spare parts.

[Supp. 25, 20 F.R. 2278, Apr. 8, 1955, as amended by Amdt. 4b-12, 27 F.R. 2991, Mar. 30, 1962]

§ 4b.307 Special factors.

Where there is uncertainty concerning the actual strength of a particular part of the structure, or where the strength is likely to deteriorate in service prior to normal replacement of the part, or where the strength is subject to appreciable variability due to uncertainties in manufacturing processes and inspection methods, the factor of safety prescribed in § 4b.200 (a) shall be multiplied by a special factor of a value such as to make the probability of the part being understrength from these causes extremely remote. The following special factors shall be used:

(a) *Casting factors.* For structural castings, the factor of safety prescribed in § 4b.200 shall be multiplied by the casting factors specified in subparagraphs (1) and (2) of this paragraph. The prescribed tests and inspections shall be in addition to those necessary to establish foundry quality control. Castings shall be inspected in accordance with approved specifications.

(1) Each casting, the failure of which would preclude continued safe flight and landing of the airplane or which would result in serious injury to occupants, shall have a casting factor of at least 1.25 and shall receive 100 percent inspection by visual, radiographic, and magnetic particle or penetrant inspection methods or approved equivalent nondestructive inspection methods. Where such castings have a casting factor less than 1.50, three sample castings shall be static tested. The test castings shall comply with the strength requirements of § 4b.201 at an ultimate load corresponding with a casting factor of 1.25 and shall comply with the deformation requirements at a load equal to 1.15 times limit load.

NOTE: Examples of castings to which this subparagraph applies are: structural attachment fittings; parts of flight control systems; control surface hinges and balance weight attachments; seat, berth, safety belt, and fuel and oil tank supports and attachments; cabin pressure valves.

(2) For structural castings other than those specified in subparagraph (1) of this paragraph, the casting factors and inspections shall be in accordance with the following table except that it shall be acceptable to reduce the percentage of castings inspected by nonvisual methods when an approved quality control procedure is established. For castings procured to a specification which guarantees the mechanical properties of the material in the castings and provides for demonstration of these properties by test of coupons cut from castings on a sampling basis, it shall be acceptable to use a casting factor of 1.0. The inspection requirements for such castings shall be in accordance with those specified in the following table for casting factors of 1.25 to 1.50, and the testing requirements shall be in accordance with subparagraph (1) of this paragraph.

Casting factor	Inspections
2.0 or greater	100 percent visual.
Less than 2.0 greater than 1.5.	100 percent visual, and magnetic particle or penetrant or equivalent nondestructive inspection methods.
1.25 to 1.50	100 percent visual, magnetic particle or penetrant, and radiographic, or approved equivalent nondestructive inspection methods.

(3) Castings which are pressure tested as parts of a hydraulic or other fluid system shall not be required to comply with the provisions of this section unless such castings support airplane structural loads.

(4) The casting factor need not exceed 1.25 with regard to bearing stresses regardless of the method of inspection employed. A casting factor need not be employed with respect to the bearing surface of a part if the bearing factor used (see paragraph (b) of this section) is greater than the casting factor.

(b) *Bearing factors.* (1) Bearing factors shall be used of sufficient magnitude to provide for the effects of normal relative motion between parts and in joints with clearance (free fit) which are subject to pounding or vibration. (Bearing factor values for control surface and system joints are specified in §§ 4b.313(a) and 4b.329(b).)

(2) A bearing factor need not be employed on a part if another special factor prescribed in this section is of greater magnitude than the bearing factor.

(c) *Fitting factors.* (1) A fitting factor of at least 1.15 shall be used on all fittings the strength of which is not proven by limit and ultimate load tests in which the actual stress conditions are simulated in the fitting and the surrounding structure. This factor shall apply to all portions of the fitting, the means of attachment, and the bearing on the members joined.

(2) In the case of integral fittings the part shall be treated as a fitting up to the point where the section properties become typical of the member.

(3) The fitting factor need not be employed where a type of joint made in accordance with approved practices is based on comprehensive test data, e. g., continuous joints in metal plating, welded joints, and scarf joints in wood.

(4) A fitting factor need not be employed with respect to the bearing surface of a part if the bearing factor used (see paragraph (b) of this section) is of greater magnitude than the fitting factor.

[15 F.R. 3543, June 8, 1950, as amended by Amdt. 4b-12, 27 F.R. 2991, Mar. 30, 1962]

§ 4b.308 Flutter, deformation, and vibration

Compliance with the following provisions shall be shown by such calculations, resonance tests, or other tests as are found necessary by the Administrator.

[96] MIL-HDBK-5 "Strength of Metal Aircraft Elements" specifies "A" and "B" values for allowable design properties. The "A" values are those which the material producer has indicated to be the minimum he expects for the given material. The only values considered guaranteed values are the tensile ultimate and tensile yield "A" values which have been published by the material producer for the grain direction accepted for commercial guarantees. The "B" values represent design properties which the materials producers have indicated will be met or exceeded by 90 percent of the material supplied by them. More detailed information on the derivation of related design mechanical properties can be obtained by referring to § 3.1.1. "Material Properties" of MIL-HDBK-5.

flight with the engines on one side of the plane of symmetry inoperative and the remaining engines at take-off power. For airplanes with flaps which are not subjected to slipstream conditions, the structure shall be designed for the loads imposed when the wing flaps on one side are carrying the most severe load occurring in the prescribed symmetrical conditions and those on the other side are carrying not more than 80 percent of that load.

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b–6, 17 F. R. 1093, Feb. 5, 1952; Amdt. 4b–3, 21 F. R. 992, Feb. 11, 1956]

§ 4b.324–1 Procedure for demonstrating wing flaps that are not interconnected (FAA policies which apply to § 4b.324(a)).

If the wing flaps are not mechanically interconnected, tests should be conducted to simulate flap malfunctioning (to the extent of the flaps being retracted on one side and extended on the other) during take-offs, approaches, and landings to demonstrate that the airplane is safe under these conditions.

[Supp. 24, 19 F. R. 4461, July 20, 1954]

§ 4b.325 Control system stops.

(a) All control systems shall be provided with stops which positively limit the range of motion of the control surfaces.

(b) Control system stops shall be so located in the system that wear, slackness, or take-up adjustments will not affect adversely the control characteristics of the airplane because of a change in the range of surface travel.

(c) Control system stops shall be capable of withstanding the loads corresponding with the design conditions for the control system.

§ 4b.326 Control system locks.

Provision shall be made to prevent damage to the control surfaces (including tabs) and the control system which might result from gusts striking the airplane while it is on the ground or water (see also § 4b.226). If a device provided for this purpose, when engaged, prevents normal operation of the control surfaces by the pilot, it shall comply with the following provisions.

(a) The device shall either automatically disengage when the pilot operates the primary flight controls in a normal manner, or it shall limit the operation of the airplane in such a manner that the pilot receives unmistakable warning at the start of take-off.

(b) Means shall be provided to preclude the possibility of the device becoming inadvertently engaged in flight.

[Amdt. 4b–6, 17 F. R. 1093, Feb. 5, 1952]

§ 4b.327 Static tests.

Tests shall be conducted on control systems to show compliance with limit load requirements in accordance with the following provisions.

(a) The direction of the test loads shall be such as to produce the most severe loading in the control system.

(b) The tests shall include all fittings, pulleys, and brackets used in attaching the control system to the main structure.

(c) Analyses or individual load tests shall be conducted to demonstrate compliance with the special factor requirements for control system joints subjected to angular motion. (See §§ 4b.307 and 4b.329 (b).)

§ 4b.328 Operation tests.

An operation test shall be conducted for each control system by operating the controls from the pilot compartment with the entire system loaded to correspond with 80 percent of the limit load specified for the control system. In this test there shall be no jamming, excessive friction, or excessive deflection.

§ 4b.329 Control system details; general.

All details of control systems shall be designed and installed to prevent jamming, chafing, and interference from cargo, passengers, and loose objects. Precautionary means shall be provided in the cockpit to prevent the entry of foreign objects into places where they would jam the control systems. Provisions shall be made to prevent the slapping of cables or tubes against other parts of the airplane. The following detail requirements shall be applicable with respect to cable systems and joints.

(a) *Cable systems.* (1) Cables, cable fittings, turnbuckles, splices, and pulleys shall be of an approved type.

(2) Cables smaller than ⅛-inch diameter shall not be used in the aileron, elevator, or rudder systems.

(3) The design of cable systems shall be such that there will be no hazardous change in cable tension throughout the range of travel under operating conditions and temperature variations.

(4) Pulley types and sizes shall correspond with the cables used.

(5) All pulleys and sprockets shall be provided with closely fitted guards to prevent the cables and chains being displaced or fouled.

(6) Pulleys shall lie in the plane passing through the cable within such limits that the cable does not rub against the pulley flange.

(7) Fairleads shall be so installed that they do not cause a change in cable direction of more than 3°.

(8) Clevis pins (excluding those not subject to load or motion) retained only by cotter pins shall not be used in the control system.

(9) Turnbuckles attached to parts having angular motion shall be installed to prevent positively any binding throughout the range of travel.

(10) Provision for visual inspection shall be made at all fairleads, pulleys, terminals, and turnbuckles.

(b) *Joints.* (1) Control system joints subjected to angular motion in push-pull systems, excepting ball and roller bearing systems, shall incorporate a special factor of not less than 3.33 with respect to the ultimate bearing strength of the softest material used as a bearing.

(2) It shall be acceptable to reduce the factor specified in subparagraph (1) of this paragraph to a value of 2.0 for joints in cable control systems.

(3) The approved rating of ball and roller bearings shall not be exceeded.

[15 F. R. 3543, June 8, 1950, as amended by Amdt. 4b–6, 17 F. R. 1093, Feb. 5, 1952]

§ 4b.329–1 Installation of turnbuckles (FAA policies which apply to § 4b.329(a)).

Fork ends of turnbuckles should not be attached directly to control surface horns or to bellcrank arms unless a positive means (such as the use of shackles, links, universal joints, spacer bushings, ball bearings, etc.) is used to prevent binding of turnbuckles relative to the horns or bellcrank arms or unless it can be shown that turnbuckles have adequate strength assuming one end fixed to the horn or arm and the design cable loads pulling off the other end at 5° to the turnbuckle axis. There should be no interference between the horns or bellcrank arms and the fork ends of turnbuckles throughout the range of motion of the control surfaces.

[Supp. 25, 20 F. R. 2278, Apr. 8, 1955]

§ 4b.329–2 Safetying of turnbuckles (FAA policies which apply to § 4b.329).

Section 4b.300 requires in part that there be no design features or details which experience has shown to be hazardous or unreliable. Experience has shown that the reliability of turnbuckles should be insured by safetying with wire as shown in figure 5. After safetying the turnbuckle, no more than three threads should be exposed on either side of the turnbuckle barrel and the ends of each safety wire should be securely fastened by at least four wraps. A turnbuckle safetying guide is given in table 1.

[Supp. 25, 20 F. R. 2278, Apr. 8, 1955]

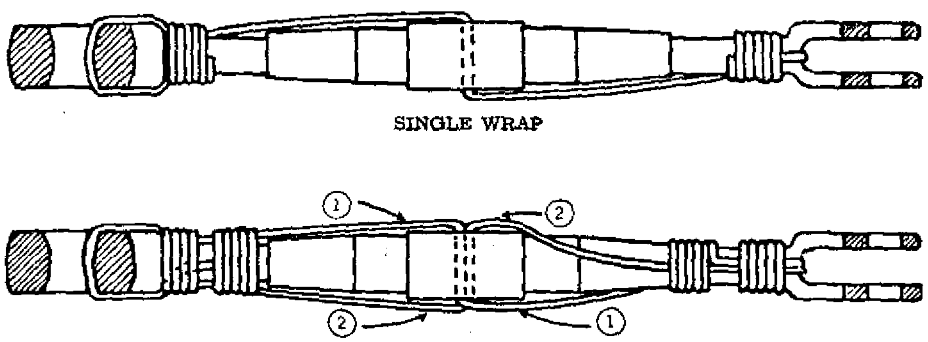
TABLE 1—TURNBUCKLE SAFETYING GUIDE

Cable size (inch)	Minimum breaking strength (pounds)				Type of wrap	Diameter of safety wire (inch)	Material (annealed condition)
	MIL–C–1511, steel (carbon) flexible, preformed		MIL–W–5424, steel (corrosion-resisting) flexible, preformed				
	7 x 7	7 x 19	7 x 7	7 x 19			
1/16	480		480		Single	0.040	Copper, brass, galvanized or tinned steel, soft iron, or monel.
3/32	920		920		do	.040	Do.
1/8		2,000		1,760	do	.040	Stainless steel.
1/8		2,000		1,760	Double	.040	Copper, brass, galvanized or tinned steel, soft iron, or monel.
5/32 and greater		2,800		2,400	do	.040	Galvanized or tinned steel, soft iron, stainless steel, or monel.
5/32 and greater		2,800		2,400	do	.051	Copper, brass.

NOTES

1. The swaged and unswaged turnbuckle assemblies are covered by AN standard drawings.
2. Certain of the AN standard swaged terminal parts specify a safety wire hole size of 0.047 inch. This hole may be reamed sufficiently to accommodate the 0.040 and 0.051 diameter wires.
3. The double wrap procedure given in Navy Specification PO–42A, Amendment No. 1, or the safetying procedure described by Military Standard Drawing MS33591 (ASG), may be used in lieu of the method shown in figure 5.

[Supp. 25, 20 F.R. 2278, Apr. 8, 1955, as amended by Amdt. 4b–12, 27 F.R. 2992, Mar. 30, 1962]



1. Wire 1 is passed through the turnbuckle hole as shown, the two wire ends are passed through the right and left hand ends of the turnbuckle and are then bent back along the barrel of the turnbuckle.

2. Wire 2 is installed and wrapped (these wraps are next to the ends of the turnbuckle).

3. The two loose ends of wire 1 are then wrapped.

FIGURE 5.

[Supp. 25, 20 F. R. 2278, Apr. 8, 1955]

§ 4b.329-3 Approval of control system components (FAA policies which apply to § 4b.329(a)).

The Administrator does not issue specific approvals as such for cables, cable fittings, turnbuckles, splices, pulleys, etc., for general use on aircraft. Approval is limited to its use as part of a specific airplane design. Conformance with established industry or military specifications or adequate substantiation of the manufacturer's own design, are the procedures utilized in complying with the "approved type" requirement.

[Supp. 25, 20 F. R. 2279, Apr. 8, 1955]

§ 4b.329-4 Cable terminals (FAA policies which apply to § 4b.329(a)).

The selection of cable terminal locations and their proximities should minimize the possibility of interferences with structure, fairleads, other terminals, etc., and the possibility of pairing wrong cables during maintenance or overhaul.

[Supp. 25, 20 F. R. 2279, Apr. 8, 1955]

§ 4b.329-5 Bellcrank and idler installation (FAA policies which apply to § 4b.329(b)).

The design of such items as bellcrank arms, tab drums, idlers, etc., should minimize the possibility of inadvertent installation in the reversed direction, or, as an alternative, to preclude the possibility of jamming or interference that might result from such reversed installation.

[Supp. 25, 20 F. R. 2279, Apr. 8, 1955]

§ 4b.329-6 Ball and roller bearings (FAA policies which apply to § 4b.329(b)(3)).

The "approved ratings" of ball and roller bearings referred to are the ratings established in MIL-HDBK-5, "Strength of Metal Aircraft Elements".

[Supp. 25, 20 F.R. 2279, Apr. 8, 1955, as amended by Amdt. 4b-12, 27 F.R. 2992, Mar. 30, 1962]

LANDING GEAR

§ 4b.330 General.

The requirements of §§ 4b.331 through 4b.338 shall apply to the complete landing gear.

§ 4b.331 Shock absorbers.

(a) The shock absorbing elements for the main, nose, and tail wheel units shall be substantiated by the tests specified in § 4b.332.

(b) The shock absorbing ability of the landing gear in taxiing shall be demonstrated by the tests prescribed in § 4b.172.

§ 4b.332 Landing gear tests.

The landing gear shall withstand the following tests:

(a) *Shock absorption tests.* (1) It shall be demonstrated by energy absorption tests that the limit load factors selected for design in accordance with § 4b.230 (b) for take-off and landing weights, respectively, will not be exceeded.

(2) In addition to the provisions of subparagraph (1) of this paragraph, a reserve of energy absorption shall be demonstrated by a test simulating an airplane descent velocity of 12 f. p. s. at design landing weight, assuming wing lift not greater than the airplane weight acting during the landing impact. In this test the landing gear shall not fail. (See paragraph (c) of this section.)

(b) *Limit drop tests.* (1) If compliance with the limit landing conditions specified in paragraph (a) (1) of this section is demonstrated by free drop tests, these shall be conducted on the complete airplane, or on units consisting of wheel, tire, and shock absorber in their proper relation. The free drop heights shall not be less than the following:

(i) 18.7 inches for the design landing weight conditions,

(ii) 6.7 inches for the design take-off weight conditions.

(2) If wing lift is simulated in free drop tests the landing gear shall be dropped with an effective mass equal to:

$$W_e = W\left(\frac{h+(1-L)d}{h+d}\right);$$

where:

W_e = the effective weight to be used in the drop test (lbs.).
h = specified free drop height (inches).
d = deflection under impact of the tire (at the approved inflation pressure) plus the vertical component of the axle travel relative to the drop mass (inches).
$W = W_M$ for main gear units (lbs.), equal to the static weight on the particular unit with the airplane in the level attitude (with the nose wheel clear in the case of nose wheel type airplanes),
$W = W_T$ for tail gear units (lbs.), equal to the static weight on the tail unit with the airplane in the tail-down attitude,
$W = W_N$ for nose wheel units (lbs.), equal to the vertical component of the static reaction which would exist at the nose wheel, assuming the mass of the airplane acting at the center of gravity and exerting a force of 1.0g downward and 0.25g forward.
L = the ratio of the assumed wing lift to the airplane weight, not in excess of 1.0.

(3) The attitude in which a landing gear unit is drop tested shall simulate the airplane landing condition critical for the unit.

(4) The value of d used in the computation of W_e in subparagraph (2) of this paragraph shall not exceed the value actually obtained in the drop test.

(c) *Reserve energy absorption drop tests.* (1) If compliance with the reserve energy absorption condition specified in paragraph (a) (2) of this section is demonstrated by free drop tests, the landing gear units shall be dropped from a free drop height of not less than 27 inches.

(2) If wing lift equal to the airplane weight is simulated, the units shall be dropped with an effective mass equal to:

$$W_e = W\left(\frac{h}{h+d}\right);$$

where the symbols and other details are the same as in paragraph (b) of this section.

[15 F. R. 3543, June 8, 1950; 15 F. R. 4171, June 29, 1950, as amended by Amdt. 4b-3, 21 F. R. 992, Feb. 11, 1956]

§ 4b.333 Limit load factor determination.

(a) In determining the airplane inertia limit load factor n from the free drop tests specified in § 4b.332, the following formula shall be used:

$$n = n_j \frac{W_e}{W} + L;$$

where:

n_j = the load factor during impact developed on the mass used in the drop test (i. e., the acceleration dv/dt in g's recorded in the drop test plus 1.0). (See § 4b.332 (b) (2) for explanation of W_e, W, and L).

(b) The value of n determined in paragraph (a) of this section shall not be greater than the limit load factor used for the landing conditions. (See § 4b.230 (b).)