U.S. Department of Transportation
Federal Aviation Administration

Advisory Circular

Subject: CERTIFICATION OF TRANSPORT CATEGORY ROTORCRAFT
Date: 9/24/91
Initiated by: ASW-110
AC No: 29-2A
Change: 2

1. PURPOSE: This change revises existing material in 17 paragraphs and adds new material for 41 paragraphs previously shown as "RESERVED."

The change number and date of the changed material are carried at the top of each page. The asterisks (*) in the right and left margins indicate the beginning and end of each change. Rearranged pages having no new material also carry the change number and new date. Pages having no changes retain the same heading information. In paragraphs that are entirely new, asterisks (*) appear only in the margins at the beginning and the end of each new page. In addition, several paragraphs were renumbered for better continuity between AC's 27-1 and 29-2A.

2. PRINCIPAL CHANGES.

a. Paragraphs 72, 80, 141, 206, 245, 268, 304, 373, 498, 621, 637, 656, 726, 748, 775, 776, and 786 are revised.

b. New paragraphs 87, 302, 303, 305, 316, 317, 318, 319, 330, 343, 347, 360, 449, 462, 549, 550, 562, 567, 568, 569, 570, 584, 585, 586, 587, 588, 589, 591, 592, 593, 594, 595, 596, 703, 705, 707, 708, 718, 725, and 785 are added to Chapter 2.

c. New paragraph 788 is added to Chapter 3.

d. Paragraphs listed below have had the paragraph numbers changed.

Existing Numbers	New Numbers
707 (Proposed)	708
708 (Proposed)	707

PAGE CONTROL CHART

Remove Pages	Dated	Insert Pages	Dated
III	4/24/89	III thru XIV	9/24/91
IV	9/16/87		
V thru XII	4/24/89		
XIII	9/16/89		
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135 thru 138	9/16/87	135 thru 137	9/24/91
		138	9/16/87
149 thru 152	9/16/87	149	9/16/87
		150 thru 152	9/24/91
163 and 164 (thru 174)	9/16/87	163	9/16/87
		164 (thru 174)	9/24/91
237 thru 250	9/16/87	237	9/16/87
		238 thru 240 (thru 248)	9/24/91
		249	9/24/91
		250	9/16/87
317 and 318	4/24/89	317	9/24/91
		318	4/24/89
387 and 388 (thru 398)	4/24/89	387	9/24/91
		388 (thru 398)	4/24/89
411 thru 432	9/16/87	411	9/16/87
		412 thru 416 (thru 432)	9/24/91
491	4/24/89	491	4/28/89
492 (thru 498)	9/16/87	492 thru 504 (thru 530)	9/24/91
499 thru 500 (thru 530)	9/16/87		
537 and 538 (thru 540)	4/24/89	537 thru 540	9/24/91
573 and 574 (thru 578)	9/16/87	573	9/24/91
		574 (thru 578)	9/16/87
581 and (thru 596)	9/16/87	581 thru 584 (thru 596)	9/24/91
599 and 600 (thru 602)	9/16/87	599	9/16/87
		600 thru 602-2	9/24/91
607	9/16/87	607 thru 620 (thru 634)	9/24/91
608 (thru 634)	4/24/89		
707	9/16/87	707	9/16/87
708	4/24/89	708 and 708-1 (and 708-2)	9/24/91
751 and 752 (thru 770)	4/24/89	751	4/24/89
		752 thru 754 (thru 770)	9/24/91
789 and 790	9/16/87	789	9/24/91
		790	9/16/87
881 and 882 (thru 888)	4/24/89	881	4/24/89
		882 thru 886 (thru 888)	9/24/91
889 and 890 (thru 900)	4/24/89	889	4/24/89
		890 (thru 900)	9/24/91
903 and 904 (thru 940)	4/24/89	903	4/24/89
		904 thru 906 (thru 920)	9/24/91
		921 thru 930 (thru 940)	9/24/91

where, V_S = vertical speed ft/sec, derived from § 29.725(a)
K_y = pitching radius of gyration - ft. from pitching axis
l_b = distance from most critical c.g. location to the guard or bumper contact point - ft.
W = gross weight less rotor lift from § 29.473(a) - lbs.
g = 32.2 ft./sec.^{2}

(ii) Other, more recent, analytical techniques (most utilizing computer programs) may, of course, be used rather than the ANC-2a means after proper substantiation for applicability and validity.

(iii) The tail rotor guard shall not fail when the limit and ultimate load, which is derived from a combination of the limit kinetic energy and the guard resulting limit deflection required to dissipate the energy, is imposed on the guard and the helicopter tail (see § 29.305).

(3) Substantiation of the guard, skid, or bumper for the design loads derived may be accomplished by test or analysis as stated in § 29.307(a).

(4) Several rotorcraft tail rotor guards are installed solely for the protection of ground personnel from the rotating tail rotor. For guards installed for this purpose, the applicant should use prudent and reasonable design loads and features. Such guards should not present a hazard to the helicopter because of its design features.

* 159. <u>§ 29.413 (through Amendment 29-19) STABILIZING AND CONTROL SURFACES</u>.

a. <u>Explanation</u>. Minimum design loads are specified for stabilizing as well as control surfaces.

(1) Paragraph (a) of the rule requires application of minimum empirical design loads, application of critical maneuvering loads, and application of critical maneuvering loads combined with vertical or horizontal gust loads (30 feet per second per § 29.341).

(2) Paragraph (b) requires load distributions that closely simulate actual pressure distributions. Both spanwise and chordwise distributions are intended.

(3) These surfaces are used for stability and control thereby hopefully extending the c.g. range and increasing the airspeed of modern designs.

(4) To "closely simulate actual pressure condition" on the surfaces, unsymmetrical loads are also required on horizontal surfaces. An arbitrary distribution, if conservative, may be used.

(5) It is noted § 29.571 requires fatigue substantiation of the flight structure which will include control and stabilizing surfaces.

(6) If the surface is controllable, a proof and operation test of the surface control system is required by §§ 29.681 and 29.683. *

b. Procedures. Modern rotorcraft designs have generally employed a fixed or a wholly movable, not split or divided, stabilizing or control surface.

(1) Design Loads.

(i) Limit loads of 15 pounds per square foot will apply up to approximately 90-knot design airspeed. Above 90 knots design airspeed (V_d), the coefficient (C_N = 0.55) imposes higher limit loads on the surface.

(ii) In addition, combined maneuvering and gust loads may impose the highest limit loads on the control surfaces of helicopters. This is attributed to the increase in speed (horizontal gust) and to the change in angle of attack and change in airspeed (vertical gust). Imposing the horizontal gust (30 feet per second or 17.8 knots) on the surface in combination with 130-knot design speed results in a 30 percent increase in the design load. The gust conditions cause a significant increase in design loads due to a change in angle of attack, with a change in resultant airspeed, or due to the increase in airspeed.

(iii) The applicant may choose to derive the limit loads using maximum aerodynamic coefficients for the surface under consideration at the maximum design airspeed combined with a 17.8-knot gust. This would be acceptable provided these design loads exceed the minimum loads derived from a C_N = 0.55 at design airspeed or exceed 15 pounds per square foot load on the surface.

(2) The load distribution on the surface should closely simulate actual pressure distributions.

(i) The spanwise load may be rectangular or other acceptable conservative distributions may be used. The method developed by O. Schrenk in NACA TM 948, 1940, is an acceptable method for approximation of spanwise distribution.

NOTE: The method is valid for aspect ratios of 5 to 12 and for rectangular planforms such as used on rotorcraft, other planforms may be acceptable as prescribed in the TM.

(ii) The chordwise distribution appropriate for the aerodynamic shape should be used.

(iii) The flight load survey conducted under § 29.571 may be used to confirm design parameters and possible load distribution data. On controllable surfaces, the pitching moment (control loads) is measured for fatigue substantiation of the control system. The control stabilizing surfaces are subject to loads measurement and possible fatigue tests for fatigue substantiation also.

(3) Proof of the structure for the required loads is specified in §§ 29.301, 29.303, 29.305, and 29.307. Tests or analysis may be used as prescribed. If analysis is used, fitting factors and other appropriate factors prescribed by the rules of §§ 29.625, 29.621, and 29.623 will be required in the analysis. *

160.-169. RESERVED.

SECTION 10. GROUND LOADS

* 170. § 29.471 (through Amendment 29-19) GENERAL.

a. Explanation. This regulation specifies that limit ground loads must be considered which are:

(1) External loads caused by landing (ground) conditions and by ground taxiing loads as specified in § 29.235.

(2) Loads considering the rotorcraft structure as a rigid body.

(3) Loads in equilibrium with linear and angular inertia loads.

(4) The critical center of gravity "must be selected so that the maximum design loads are obtained in each landing gear element."

b. Procedures.

(1) The standards to be considered are specified in §§ 29.473 through 29.511. These associated standards cover landing gear arrangements, landing conditions, and ground handling conditions.

(2) Drop tests are required for determination of landing load factors. See paragraph 298 of this document.

(3) The application of the design loads derived from the landing load factors will be as specified for each element affected by landing or ground handling loads.

(4) During the applicant's flight test program, the ground, landing, and taxiing load factors may be monitored to assure the design load factors used are adequate. See paragraph 97 of this document for § 29.235 policy.

171. § 29.473 (through Amendment 29-19) GROUND LOADING CONDITIONS AND ASSUMPTIONS.

a. Explanation. The rotorcraft is to be designed for the maximum weight. A rotor lift of two-thirds of the design maximum weight may be used. The minimum limit landing load factor is determined by the drop tests of § 29.725. Provisions are made for supplementary energy absorption devices that have triggering mechanisms.

b. Procedures. Loads for the landing conditions are derived considering mass (equal to the maximum weight) and rotor lift (equal to two-thirds of the maximum weight) acting through the center of gravity throughout the landing impact. Unbalanced external loads resulting from asymmetric loading conditions are reacted as specified in the individual subparagraphs. *

NOTE: If supplementary energy absorption devices are used, neither they nor their triggering devices may fail under the loads established by the limit drop tests or the reserve energy absorption drop tests.

172. § 29.475 (through Amendment 29-19) TIRES AND SHOCK ABSORBERS.

a. Explanation. This section specifies the tire and shock absorber position to be used in ground load derivations.

b. Procedures. Ground loads are to be derived with the tires in static (1g) position and the shock absorbers "in their most critical position." The determination of the "most critical position" for the shock absorbers generally requires a load versus deflection test or analysis of the shock absorber system and a determination of the effect of both load and deflections on the shock absorber, attachment structure, and substructure designed by ground loads.

173. § 29.477 (through Amendment 29-19) LANDING GEAR ARRANGEMENT.

a. Explanation. This section specifies the individual standards to be used for ground load conditions for helicopters having two wheels aft and one or more wheels forward of the center of gravity.

NOTE: § 29.497 gives ground loading conditions for landing gear with tail wheels, and § 29.501 gives ground loading conditions for landing gear with skids.

b. Procedures. The ground loading conditions of §§ 29.235, 29.479 through 29.485, and 29.493 will be used for helicopters having two wheels aft and one or more wheels forward of the center of gravity. This includes forward wheels on separate axles.

174. § 29.479 (through Amendment 29-19) LEVEL LANDING CONDITIONS.

a. Explanation. This section provides explicit level landing load criteria for landing gear with two wheels aft and one or more wheels forward of the center of gravity.

(1) Level landings--

(i) Each wheel contacting the ground simultaneously; and

(ii) Aft wheels contacting the ground with forward wheels just clear of the ground. *

(2) Application of loads--

(i) Maximum design vertical loads applied alone;

(ii) The maximum design vertical loads applied with a drag load of at least 25 percent of the vertical load (applied at the ground contact area); and

(iii) The vertical load at the instant of peak drag load in conjunction with the peak drag load. A ground speed and load application is specified.

(3) A 40 percent/60 percent load distribution between wheels for configurations having two forward wheels including quadricycle. This distribution between wheels on a common axis is to be applied for the conditions of vertical loads only, and for vertical loads combined with drag loads of 25 percent of the vertical loads. Section 29.511 concerns a 60 percent to 40 percent ground load distribution between multiple-wheel units. See paragraph No. 182 of this document for dual wheels on a common axle or axis.

(4) Aircraft pitching moments are to be reacted by the forward landing gear or by the angular inertia forces when the forward landing gear is clear of the ground as specified.

b. Procedures.

(1) The specified loading conditions will be used in load derivations.

(2) The critical center of gravity condition will be used for each gear and gear support structure.

(i) The aft center of gravity condition with the forward gear clear will normally be critical for the aft gear and gear supports.

(ii) The forward center of gravity condition with each gear contacting the ground simultaneously will normally design forward gear elements critical for vertical loads.

(iii) The forward center of gravity condition with the forward gear clear may result in high load factors, angular plus linear, that will greatly affect security of items of significant mass.

(3) The vertical load, at the instant of peak drag load combined with the peak drag component, can be determined from drop tests utilizing wheel spin-up or it can be analytically determined. If analysis is used, it must successfully correlate with the results of a previous well-instrumented test program.

175. § 29.481 (through Amendment 29-19) TAIL-DOWN LANDING CONDITIONS.

a. Explanation. This section provides the criteria for tail-down landing conditions, i.e., "the maximum nose-up attitude allowing ground clearance" with ground loads acting "perpendicular to the ground."

b. Procedures.

(1) The tail-down landing condition will be used to check (by analysis or test) for criticality of landing gear or support structure. This attitude generally creates the highest forward loads on the landing gear in combination with vertical loads.

(2) The tail-down landing condition may be the critical condition for both landing load factor and for energy absorbtion by the main gear. Section 29.725 requires that "each landing gear must be tested in the attitude simulating the landing condition that is most critical." Where questions exist as to the critical attitude, both level landing and tail-down landing attitudes should be used in drop tests required by § 29.725.

176. § 29.483 (through Amendment 29-19) ONE-WHEEL LANDING CONDITIONS.

a. Explanation. This section gives the condition to be used for one-wheel landing conditions. Only the vertical load condition of § 29.479(b)(1) is required.

b. Procedures. The one-wheel landing condition is generally critical for the landing gear-to-fuselage attachments and the landing gear elements between the attachments. Unbalanced external loads are reacted by rotorcraft inertia. Large items of mass located radially from the center of gravity (aircraft centerline may be used) should also be structurally substantiated for the combined rolling (angular) and linear accelerations of this loading condition.

177. § 29.485 (through Amendment 29-19) LATERAL DRIFT LANDING CONDITIONS.

a. Explanation.

(1) This section provides the loading conditions which impose side (and vertical) loads on the landing gear. A level landing attitude is specified. Two main conditions required are--

(i) Only the aft wheels in contact with the ground; and

(ii) All wheels contacting the ground simultaneously.

(2) Loads. The vertical loads to be applied with the side loads are specified as "one-half of the maximum ground reactions of § 29.479(b)(1)." These vertical loads are the level landing loads considering both contact and noncontact with the ground by the forward wheels.

(i) One side load condition is specified as "0.8 times the vertical reaction acting inward on one side and 0.6 times the vertical reaction acting outward on the other side" when only the aft wheels contact the ground.

(ii) The other side load condition (for all wheels contacting the ground) specifies the 80 percent inward/60 percent outward distribution for the aft wheels and 0.8 times (80 percent) the vertical reaction for the forward wheels.

b. Procedures. The loading conditions, as specified, are applied to the landing gear and attaching structure. The loads are applied at the ground contact point, except for full swiveling gear which has the load applied at the center of the axle. In other words, full swiveling gear is considered to have swiveled to a static position under the side load before the design vertical and side loads are achieved. The landing gear backup structure, as well as the landing gear itself, will be substantiated for these side load conditions.

178. § 29.493 (through Amendment 29-19) BRAKED ROLL CONDITIONS.

a. Explanation. This section provides two loading conditions for ground braking operations. Specific vertical loads in conjunction with drag loads (due to braking) are to be considered. The limit vertical load factor is 1.33 for condition of all wheels in contact with the ground, and 1.0 for condition of aft wheels only in contact with the ground and nose wheel clear. The drag load on wheels with brakes is 0.8 times the vertical load or the drag load value based on limiting brake torque, whichever is less.

b. Procedures. The braking loads are calculated from the specified criteria with the shock absorbers in their static (normal) positions and with the drag loads applied at the ground contact point. Structural substantiation of the affected structure may be accomplished by test or analysis. If tests are used, the wheel and tire assembly is commonly replaced with a test fixture so the limit loads and static deflections specified can be more accurately controlled. The test specimen should be complete enough to assure that the landing gear structure and the attach and backup structure are adequately substantiated.

179. § 29.497 (through Amendment 29-19) GROUND LOADING CONDITIONS: LANDING GEAR WITH TAIL WHEELS.

a. Explanation. This section provides the loading conditions for landing gear designs with tail wheels.

(1) Level landings are to consider the following:

(i) All wheels (main and tail) contacting the ground simultaneously, as well as only forward main wheels contacting the ground.

(ii) Maximum design vertical loads applied alone.

(iii) The maximum design vertical loads combined with a drag load of at least 25 percent of the vertical loads for both conditions.

(2) Noseup landings with only the rear wheel or wheels initially contacting the ground must be considered unless shown to be extremely remote.

(3) Level landings on one forward wheel only are to be considered. Drag loads are not required.

(4) Side load conditions are imposed on the main wheels and tail wheels for level landing attitudes. Criteria for full swiveling and locked tail wheels are included in this standard.

(5) Braked roll conditions are specified for the level landing attitudes.

(6) Rear wheel turning loads are also specified for swiveling and locked tail wheels.

(7) Taxiway condition loads for the landing gear and rotorcraft are those that "occur when the rotorcraft is taxied over the roughest ground that may reasonably be expected in normal operation." The aircraft design load factors should not be exceeded during the evaluation. Section 29.235 contains an identical standard that applies to all types of wheel landing gear.

b. Procedures.

(1) The specified loading conditions are to be used in load derivations.

(2) The critical center of gravity condition is used for each gear and gear support structure.

(i) The forward center of gravity condition with the tail gear clear will normally be critical for the forward gear and gear supports. *

* (ii) The aft center of gravity condition with the tail gear clear should be checked for criticality of security of large mass items located forward of the center of gravity Vertical and angular accelerations are additive under this landing condition.

(iii) The aft center of gravity condition with each gear contacting the ground simultaneously will generally design tail gear elements critical for vertical loads. The other conditions are generally less severe but must be proven.

(3) For noseup landing procedures use § 29.481. The reference to "extremely remote" in § 29.497(d)(2) predates current §§ 25.1309, 29.1309, and AC 25.1309.1. This phrase has been used to require consideration of noseup landings unless features of design are present which prevent noseup landings or where such landings are unlikely during the life of the rotorcraft. See paragraph No. 175 of this document.

(4) Use § 29.483 for one-wheel landing procedures, paragraph No. 176 of this document.

(5) Use § 29.485 procedures for side load conditions, paragraph No. 177 of this document.

(6) Use § 29.493 procedures for braked roll conditions, paragraph No. 178 of this document.

(7) For rear wheel turning loads, swiveling of tail landing gears is allowed as in basic side load conditions. The side load is applied at the axle, or if the wheel is locked, the load is applied at ground contact. Rear wheels are loaded with the critical vertical static load in conjunction with an equal side load to substantiate the tail gear.

(8) Since the rotorcraft is to be designed for load factors that will not be exceeded during taxi tests or other conditions, an instrumented taxi test program will be necessary. Use § 29.235, paragraph No. 97, of this document.

180. <u>§ 29.501 (through Amendment 29-19) GROUND LOADING CONDITIONS: LANDING GEAR WITH SKIDS</u>.

a. <u>Explanation</u>. This section provides the ground loading conditions for landing gear with skids. The loading conditions are similar to those for wheeled gear except for the following criteria which are unique to skid gears:

(1) Structural yielding (plastic deformation) of elastic spring members under limit loads is allowed. *

* (2) Design ultimate loads for elastic spring members need not exceed the loads obtained in a drop test with a drop height of 1.5 times the limit drop height. The rotorcraft and the landing gear attachments are subject to the prescribed design ultimate loads.

(3) The gear must be in its most critically deflected position (similar to § 29.475).

(4) Ground reactions are rationally distributed along the bottom of the skid unless otherwise specified. Paragraph (f) concerns specific "concentrated" and arbitrary load conditions.

(5) Drag loads are 50 percent of vertical reactions rather than the 25 percent for wheeled gear.

(6) Side loads are 25 percent of the total vertical reaction rather than the 60-80 percent for wheeled gear.

(7) Side loads are applied to one skid only (inward acting and outward acting) with resulting unbalanced moment resisted by angular acceleration.

(8) A ground reaction load of 1.33 times the maximum weight is to be applied at 45 degrees from the horizontal axis:

(i) Distributed among or between the skids;

(ii) Concentrated at the forward end of the straight portion of the skid tube; and

(iii) Applied only to the forward end of the skid tube and its attachment to the rotorcraft.

(9) A concentrated vertical load equal to one-half of the design limit vertical load is to be applied at a point midway between the skid tube attachments.

b. Procedures.

(1) The specified loading conditions are to be used in load derivations.

(2) The critical center of gravity conditions are to be used for each gear and gear support structure. Asymmetry of the skid tubes, cross tubes, and gear attachments are to be considered in determining the critical center of gravity condition. *

* (3) The rotorcraft and landing gear attachment must be substantiated for ultimate landing loads by either test or analysis utilizing an ultimate load factor of 1.5 in accordance with § 29.303. The elastic spring members may be analyzed or static tested for ultimate loads (and deflections) using either a factor of safety of 1.5 or one associated with an "ultimate" drop height of 1.5 times the limit drop height. Substantiation by "ultimate" drop tests may be used provided all combinations of critical parameters are included in the total substantiation effort. This method will require a series of tests using several test specimens, or a limited number of drop tests plus further substantiations by static tests or analyses for additional critical conditions not covered by the drop test(s).

181. <u>§ 29.505 (through Amendment 29-19) SKI LANDING CONDITIONS</u>.

a. <u>Explanation</u>. This is an optional requirement for ski operations. The regulation specifies vertical loads, side loads, and torque loads (Mz) to be applied to ski installations. The four loading conditions to be applied at the pedestal bearings are:

(1) Simultaneous application of Pn, up load, and Pn/4, horizontal load.

(2) Up load of 1.33 P.

(3) Side load of 0.35 Pn.

(4) Torque load of 1.33 P (in foot-pounds), about vertical axis through centerline of the pedestal bearings.

NOTE: Where P is the maximum static weight on each ski and n is the limit load factor obtained from drop tests. The load factor obtained from wheel or skid landing gear drop tests may be used.

b. <u>Procedures</u>. Structural substantiation may be accomplished by static test or analysis using the specified loads. Skis generally have a limit load rating. The design loads derived for this standard must not exceed the rating. See § 29.737, paragraph No. 305, for information on ski design standards. TSO-c28 concerns, in part, standards for aircraft skis. *

182. § 29.511 (through Amendment 29-19) GROUND LOAD: UNSYMMETRICAL LOADS ON MULTIPLE-WHEEL UNITS.

a. Explanation. Two loading conditions are provided to account for unsymmetrical loads on multiple-wheel units due to landing and normal operations over crowned runways and taxiways and to account for deflated tires. They are:

(1) Sixty percent of total ground reaction applied to one wheel of a dual wheel unit and 40 percent to the other.

(2) Sixty percent of the "specified load for the gear unit" is applied to the wheel with an inflated tire when the other tire is deflated (the 60 percent load may not be less than the 1g static load).

NOTE: The 60:40 distribution also applies to nose wheel units as noted in § 29.479(b)(4).

b. Procedures. Structural substantiation may be accomplished by static test or analysis using the specified load. As provided by the standard, the total load on the gear units may neglect the transverse shift of the load centroid due to unsymmetrical load distribution; i.e., the external load for each gear may be calculated considering the same load centroid as with symmetrical wheel loads, and then the external load for each gear is divided in accordance with the distributions of § 29.511(a) and (b) between the wheels. *

183.-192. RESERVED.

SECTION 11. WATER LOADS

193. § 29.519 HULL TYPE ROTORCRAFT: WATER BASED AMPHIBIAN AND LIMITED AMPHIBIAN. (RESERVED)

194. § 29.521 FLOAT LANDING CONDITIONS. (RESERVED)

195.-204. RESERVED.

SECTION 12. MAIN COMPONENT REQUIREMENTS

* 205. § 29.547 (through Amendment 29-19) MAIN ROTOR STRUCTURE.

a. Explanation. This regulation requires the main rotor structure to be designed to the static load requirements of §§ 29.337 through 29.351 (vertical maneuvering loads, vertical and horizontal gust loads, and yawing maneuver loads). In addition, the main rotor blades, hubs, and flapping hinges are specified to be designed for impact forces of each blade against its stop during ground operation and for specified limit torque at any rotational speed including zero. The torque forces (from the drive system) are distributed to the rotor blades as specified.

b. Procedures.

(1) Substantiation in compliance with this standard is accomplished by application of the flight loads of §§ 29.337 through 29.351 and the torque loads of § 29.361 to the rotor structure by stress analyses and/or static tests. The use of wind tunnel data as well as flight loads survey data may be used to generate and/or check the external load magnitudes and distributions.

(2) Where new materials are used in the main rotor structure, such as composites containing plastics, the effects of temperature and humidity are to be considered in accordance with § 29.603, and the effects of uncertainties in manufacturing processes or inspection methods are to be considered in accordance with § 29.619.

(3) The design impact forces of each blade must be imposed against its stop or stops. Appropriate monitoring of the blades, hubs, flapping hinges, and stops during laboratory tests, ground endurance tests, and flight tests should ensure that the stops are sufficient for ground operation loads. The design torque loads are derived as prescribed.

206. § 29.549 (through Amendment 29-19) FUSELAGE AND ROTOR PYLON.

a. Explanation. This regulation requires that the fuselage and rotor pylon (including the tail fin, if any) be designed to withstand the flight loads of §§ 29.337 through 29.351, the ground loads of §§ 29.235, 29.471 through 29.497, ski loads of § 29.505, water loads of § 29.521, and rotor loads of § 29.547(d)(1) and (e)(1)(i). The ski and water loads pertain to optional features.

(1) Consideration is also required of--

(i) Auxiliary rotor thrust;

(ii) The torque reaction of each rotor drive system; and

(iii) Balancing air and inertia loads. *

(2) Each engine mount and adjacent fuselage must be substantiated as prescribed. In addition, if 2 1/2-minute power is used, "each engine mount and adjacent structure must be designed to withstand the loads resulting from a limit torque equal to 1.25 times the mean torque for 2 1/2-minute power combined with 1g flight loads."

b. Procedures. Compliance with this standard is accomplished by application of the specified aircraft loads including engine torque to the fuselage and rotor pylon structure by stress analyses and/or static tests. Drive system torque factors to be used are noted in § 29.547 for the main rotor structure as well as in paragraph (e) of this standard.

207. § 29.551 (through Amendment 29-19) AUXILIARY LIFTING SURFACES.

a. Explanation. This regulation specifies that auxiliary lifting surfaces be designed to withstand critical flight and ground loads derived for conditions specified and any "other critical condition expected in normal operation." Stub wings would comply with this standard.

b. Procedures. The surface design loads are derived from the conditions specified. Conservative aerodynamic data, including load distributions, may be used in place of data derived from wind tunnel or instrumented flight testing of the exact aerodynamic shapes involved. Special attention should be placed on concentrated load effects from fuel tanks or other large mass items that may be located in lifting surfaces. These types of load concentrations are to be considered in conjunction with inertia and aerodynamic loads. *

208.-217. RESERVED.

SECTION 13. EMERGENCY LANDING CONDITIONS

218. § 29.561 GENERAL (RESERVED)

219. § 29.563 (through Amendment 29-19) STRUCTURAL DITCHING PROVISION.

a. Explanation. Amendment 29-12 included certification requirements for ditching approvals. The helicopters must be able to sustain an emergency landing in water as prescribed by § 29.801(e).

b. Procedures. Refer to paragraph 337, § 29.801, for procedures.

220.-229. RESERVED.

* 251. § 29.625 (through Amendment 29-19) FITTING FACTORS.

a. Explanation. A 1.15 factor is specified to assure that the calculated load and stress distribution within any fitting is conservative. Application of the factor is excluded or excepted as stated in the rule.

b. Procedures.

(1) The factor may be applied to the calculated load or stress for the fitting.

(2) The structural substantiating data for the rotorcraft, including the rotor system, must include the prescribed fitting factor. The rotor system includes the flight control system rotor head and hubs and rotor blade attachments.

252. § 29.629 (through Amendment 29-19) FLUTTER.

a. Explanation.

(1) The rotorcraft must be free from flutter.

(2) Section 29.251 vibration is an associated flight requirement concerning flight demonstrations. See paragraph 110 of this document for this standard.

(3) Section 29.571(a)(3) concerns in-flight measurement of loads or stresses.

b. Procedures.

(1) Freedom from flutter may be shown by analysis or appropriately instrumented flight flutter tests.

(2) The flight loads survey proposal submitted for compliance with § 29.571 may also contain tests to fulfill compliance with § 29.629. The flight loads survey program encompasses the envelope of design airspeed and rotor r.p.m., and sufficient aerodynamic excitation is generally present to excite any latent flutter modes.

(3) Flight loads survey data or flight flutter test data submitted should be reviewed to assure that excessive oscillatory loads of rotors or surfaces will not be encountered. *

253.-264. RESERVED.

SECTION 16 ROTORS

265. § 29.653 PRESSURE VENTING AND DRAINING OF ROTOR BLADES. (RESERVED)

266. § 29.659 MASS BALANCE. (RESERVED)

267. § 29.661 (through Amendment 29-19) ROTOR BLADE CLEARANCE.

a. Explanation.

(1) The rotors, main and tail, must not strike other parts of the rotorcraft during any operating condition.

(2) Section 29.411 concerns protection of the tail rotor from a ground strike.

b. Procedures.

(1) The applicant should have drawings or sketches of the rotorcraft that show an adequate minimum clearance between the rotors, main and tail, and parts of the rotorcraft. Probable flexing of the rotors should be considered in determining the minimum clearance.

(2) During parts of the FAA-conducted flight test program, frangible devices (wood dowels) or other means of measuring clearance, may be requested to confirm that the clearance shown in the drawings or sketches is adequate in certain operating conditions. Balsa wood dowels may be clamped to the aft part of the fuselage within the rotor arc. If the devices are intact after autorotation landing tests and other tests involving typical abrupt, cyclic, and rudder pedal displacements, the clearance should be satisfactory and compliance obtained. If such measuring devices are used, the type inspection report should contain a record of clearance found during the tests. It is not necessary to precisely determine the clearance but only necessary to determine "enough clearance" as stated in the rule.

268. § 29.663 (through Amendment 29-19) GROUND RESONANCE PREVENTION MEANS.

a. Explanation.

(1) This rule, adopted in Amendment 29-3, requires reliability and damping action investigation for the ground resonance prevention means and requires associated maintenance information in the maintenance manual (§ 29.1529). The probable range of variations in service, not just the allowable range, must be investigated as prescribed. This probable range includes operation on the ground, water, or other appropriate landing surface applicable to the rotorcraft design. Quantitative test data are generally obtained in compliance with this rule. See the preamble to Amendment 29-3 for further information.

(2) Note that the maintenance information is not contained in the approved section of the maintenance manual. *

(3) Paragraph 99 concerns demonstrating freedom from ground resonance during certain applicant and TIA verification evaluations or tests of the rotorcraft. Section 29.241 complements the requirements of § 29.663. As noted in paragraph 99 of this document, the FAA removed from CFR Part 7 a specific requirement for a ground vibration survey. However, § 29.663 was adopted by Amendment 29-3 to investigate possible sources of ground resonance and to assure the reliability of the ground resonance prevention means; i.e., dampers to preclude occurrance of ground resonance. The total rotorcraft system is evaluated under this rule.

(4) Viscous dampers have been used for many years to prevent ground resonance. Modern rotorcraft designs may also use elastomeric dampers and may use elastomeric bearings in the rotor head and rotor pylon attachment to the airframe. The rule also requires investigation of the probable range of variations of these dampers and bearings to preclude ground resonance.

(5) Ground resonance can occur due to flexibility in the rotor pylon restraint system as well as with landing gear flexibilities. This evaluation should include variations in stiffness and damping of the rotor pylon restraints that may occur in service (reference "Ground Vibrations of Helicopters," M. L. Deutsch, JAS, Vol. 13, No. 5, May 1946).

b. Procedures.

(1) The reliability of the means for preventing ground resonance may be substantiated as stated in the rule. An analysis report or a test proposal and subsequent test report may be used to show compliance. The probable ranges of damping restriction are an important part of the assessment. The test may be conducted in conjunction with the testing required by § 29.241. See paragraph 99.

(i) Analysis and tests may be used;

(ii) Reliable service history of identical or closely similar systems may be used. The materials and fluids used, clearance or fits, seals, and physical installation are important items to be evaluated and considered for "closely similar" systems;

(iii) Testing of the complete rotorcraft may be used to prove that malfunction of a single means of the damping system will not cause ground resonance. One method of demonstrating acceptable compliance is by removing all seals from one damper, in conjunction with considering the allowable ranges of damping of the other parts of the rotorcraft damping system and operating the rotorcraft throughout the rotor speed range from start to maximum rotor speed. The rotorcraft cyclic control should be displaced as noted in paragraph 99 of this document to assure that the possible rotorcraft resonance frequencies are excited. If vibrations are damped in all tests, the damping system is satisfactory. Each critical rotor damper and landing gear damper must simulate a malfunction to comply with the rule. The testing discussed, however, could * become very extensive if one were to attempt to test all combinations of all maintenance adjustments of all components which contribute to the prevention of ground resonance, while at the same time rendering each of the pertinent components ineffective in turn and then repeating all of the maintenance tolerance testing each time. Fortunately, rational analytical methods are available which will permit the evaluation of such combinations so that only the combinations with the least amount of margin used are physically tested.

(2) The probable ranges of damping must be investigated as prescribed and noted in subparagraph a(1). An approved test proposal and test results report should be used for complying with § 29.663(b). If wheel landing gear is used on the rotorcraft, the probable ranges of tire pressure or the lowest probable tire pressure should be stated in the test proposal and effects of the tire pressure investigated during the test. See paragraph 99, § 29.241, concerning tests and instrumentation of the test associated with complying with § 29.241. The instrumentation noted in paragraph 99 also applies to § 29.663(b).

(3) If the wheel landing gear is equipped with wheel brakes, the evaluation should include brakes "on" and "off." The nose or tail wheel should be locked and unlocked if it swivels to evaluate any possible adverse effects of this feature. *

269.-278. RESERVED.

SECTION 17. CONTROL SYSTEMS

279. § 29.671 GENERAL. (RESERVED)

280. § 29.675 STOPS. (RESERVED)

281. § 29.679 CONTROL SYSTEM LOCKS. (RESERVED)

282. § 29.681 LIMIT LOAD STATIC TESTS. (RESERVED)

283. § 29.683 OPERATION TESTS. (RESERVED)

284. § 29.685 CONTROL SYSTEM DETAILS. (RESERVED)

285. § 29.687 SPRING DEVICES. (RESERVED)

(3) The required "down-lock" should be checked during the operation test. The design drawing should be reviewed for compliance prior to conducting an operation test.

(4) If an optional "up-lock" is installed, the landing gear should be extended during the operation test after simulation of critical failure mode of the retraction system.

(5) An "operation" test plan or proposal submitted for compliance with § 29.729(d) should include the items noted in the two previous subparagraphs and should include a functional check of the position indicator system. Those tests must be satisfactorily completed before issuing the TIA.

(6) During the official FAA flight tests, compliance with the emergency operation, position indicator, and control aspect of § 29.729(c), (e), and (f), respectively, will be verified or accomplished. In addition, the F and R test program plan (§ 21.35) will specify certain tests or evaluations for the retraction system.

(7) Position Indicator Evaluation.

(i) When evaluating the position indicator system, emphasis should be placed on the switches and their installations, and on the cockpit presentation. Each gear must have its own set of switches to indicate when it is secured in its extreme "up" position and its extreme "down" position. The switches must be located to give a valid indication of the arrival of the gear at its extreme position.

(ii) The reliability and environmental qualifications of the switches to be used should be carefully considered. An example of a condition that has potential for trouble is operation on wet areas. Trouble starts when water is picked up by the tires and deposited on the switches. During winter months the water can freeze, and the resulting ice may prevent the switch from functioning properly.

(iii) An acceptable cockpit presentation consists of two lights for each gear. One light is colored "green" and indicates when its gear is secured in the extreme "down" position. The other light is colored "amber" and indicates when its gear is in transit. When the gear is in either extreme position, the in transit light is "out." For this presentation the indication to the pilot that the gear is in the extreme "up" position is an all-gear lights-out condition.

(iv) Some manufacturers have also included a warning system to alert the crew if the landing gear has not been extended prior to landing. If a warning system is presented, §§ 29.1301 and 29.1309 should be used to evaluate its functional characteristics and the impact of its failure modes.

302. § 29.731 WHEELS. (RESERVED)

303. § 29.733 TIRES. (RESERVED)

304. § 29.735 BRAKES. (RESERVED)

305. § 29.737 SKIS. (RESERVED)

306.-315. RESERVED.

SECTION 19. FLOATS AND HULLS

316. § 29.751 MAIN FLOAT BUOYANCY. (RESERVED)

317. § 29.753 MAIN FLOAT DESIGN. (RESERVED)

318. § 29.755 HULL BUOYANCY. (RESERVED)

319. § 29.757 HULL AND AUXILIARY FLOAT STRENGTH. (RESERVED)

320.-329. RESERVED.

SECTION 20. PERSONNEL AND CARGO ACCOMMODATIONS

330. § 29.771 PILOT COMPARTMENT. (RESERVED)

331. § 29.773 PILOT COMPARTMENT VIEW. (RESERVED)

* 332. § 29.775 (through Amendment 29-19) WINDSHIELD AND WINDOWS.

a. Explanation. Nonsplintering safety glass is required in windshields and windows containing glass to protect crew and passengers in the event that window fracturing occurs.

b. Procedures. Use nonsplintering safety glass in windshield or window applications which contain glass rather than plastic acrylics, polycarbonates, epoxys, etc. The glass selected must meet a specification such as MIL-G-25871, and if new vendors are selected by an airframe manufacturer, test data should be obtained from the vendor to demonstrate the safety glass provided meets an acceptable specification and provides adequate nonsplintering capability. *

333. § 29.777 COCKPIT CONTROLS. (RESERVED)

* 334. § 29.783 (through Amendment 29-20) DOORS.

a. Explanation. This regulation requires at least one door for all closed cabin rotorcraft. Standards for all doors and airstair doors are included. To assure that the doors provide normal entry and egress without causing or contributing to hazardous conditions, even after a minor crash, the following requirements are imposed:

(1) Passenger doors may not be located with respect to any rotor to endanger persons using the doors as instructed.

(2) Means are required for locking crew and external passenger doors to prevent their opening in flight due--

(i) To inadvertent operation; or

(ii) To mechanical failure.

(3) External doors are required to be openable from the inside or outside by simple and obvious means.

(4) Reasonable provisions to prevent jamming of external doors are required as specified and to assure that an "airstair door" is useable.

(5) The following visual indications of external doors being closed and locked are required: *

(i) Direct visual inspection means by crewmembers of the locking mechanism of all external doors.

(ii) Visual means to signal to crewmembers "when normally used external doors are closed and fully locked."

(6) For certain outward opening doors, an auxiliary safety latching device is required "to prevent the door from opening when the primary latching mechanism fails." Suitable operating procedures to prevent this device from being used during takeoff and landing are required if the door cannot be opened from outside the rotorcraft (ref. § 29.783(c)) with the device in place.

b. Procedures.

(1) Passenger doors should be located as far as possible from the auxiliary rotors. The doors may be hinged and door open stops may be provided to separate entering and egressing passengers from the auxiliary rotor blades. If necessary for the design, "appropriate instructions" should be provided for all passenger doors concerning entering and leaving the rotorcraft and safe use of each door relative to all rotors. These instructions should be obvious to a passenger using the door, contain large enough letters to be readily legible, and use letters or background colors associated with danger (i.e. orange or red).

(2) Means to prevent the opening of doors in flight.

(i) Means to prevent the opening of doors in flight due to inadvertent operation may be provided by recessing door handles to prevent their inadvertent operation by the normal movement of passengers about the cabin. If recessing the door handle is impractical, a cover may be provided which will prevent inadvertent operation of the handle, but the cover should be of such design that it does not obscure the door handle nor its operating instructions. It must not unduly interfere with deliberate operation of the door handle by passenger or crew. Transparent or nonsolid covers, easily displaced by deliberate actions, have been used to prevent inadvertent door handle operation. Some rotorcraft designs meet this requirement by requiring that passengers wear their seat belts at all times during flight. This design requires that the "fasten seat belt" sign be on at all times the rotorcraft is in flight (for practical purposes, the "fasten seat belt" light is generally designed to be on when power is applied to the rotorcraft).

(ii) Means to prevent inadvertent door opening in flight due to "mechanical failure" is most efficiently provided by multiple door latches and multiple load path door locking mechanisms so that the door will remain locked after a single failure. Care should be taken in the design of multiple load path latches and mechanisms to assure independence of all failures and to consider the effort of deflections after failures (if a failure allows deflections into the airstream sufficient to increase aerodynamic loads, the increase in loads should be accounted for; if a failure allows significant movement of latching components, the deflections should be accurately accounted for to assure that disengagement of nonfailed latches does not occur). *

* (3) The means to open normally used external doors is required to be simple (such as a rotating handle) and to be accessible from the inside or the outside. To prevent the inadvertent use of emergency exits (separate from normal entry doors) for routine entry and exit with the resulting "wear and tear," the normally used doors for entry and exits should be equipped with operating handles and instructions distinctly different from those of the emergency exits. Obviously, the above does not apply to normally used exits which are also the primary (or only) emergency exits.

(4) Reasonable provisions to prevent jamming of external doors include the following:

(i) Design features of doors which are insensitive to large fuselage deflections for door operation.

(ii) Provision of clearance between door and door frame latching devices sufficient to allow some relative deflection between the door and door frame and still allow door operation. The relative deflections may be determined by static test or by an analysis approved by the FAA.

(iii) Sliding doors are frequently used in transport helicopters for versatility and utility reasons. If sliding doors are used, one of the following features of design may be required to assure that the requirements of § 29.783(d) are met:

(A) The sliding door(s) must be provided with jettison features which allow release of the door(s)from the tracks (to preclude jamming). The emergency release is generally separate and distinct from the normal door handle.

(B) Separate emergency exits of appropriate size and number may be installed in the sliding door(s).

(C) Separate emergency exits of appropriate size and number may be installed in addition to the sliding door(s).

(iv) Whether or not the sliding door is qualified as an emergency exit, it must meet the remaining door design standards.

(5) Direct visual inspection means by crewmembers of the locking mechanism of external doors may provide for visual observation of the door frame and the latching components for engagement or for visual observation of "flag" areas of the locking mechanism. If "flag" areas are used (such as tabs or shoulders which protrude into the crewmember's line of sight when the latches are engaged (locked)), care should be taken to assure that the tab is permanently affixed (or an integral part) to the locking mechanism; and it should not give erroneous readings to the crewmembers under any foreseeable operation or failure of the latching mechanism. "Visual means to signal" to crewmembers "when normally used external doors are closed and fully locked" may be provided by annunciator *

panel lights or equivalent means. The visual indicating system may consist of an indicator for each individual door, or a system connecting all doors in series. If the latter system is used, it need not necessarily show which door is not fully locked. It is not necessary that more than one crewmember be able to ascertain by a visual signal that all external doors normally used by the crew in supplying the rotorcraft, or in loading and unloading passengers and cargo, are fully closed and locked. The visual signal should be located so that it may easily be seen by the appropriate crewmember from his station.

(6) For § 29.783(f), the auxiliary safety latching device to "prevent the door from opening when the primary latching mechanism fails" can be provided by the same multiple load path features which meet the § 29.783(c) requirement for prevention of door opening in flight after a "mechanical failure." If a completely separate "auxiliary safety latching device" is used, it should allow the door to be opened from the inside, or outside, when in place. If the device must be removed to allow use of the door, "suitable operating procedures" (i.e., placards and RFM instructions) will be required for removal of the device during takeoff and landing.

(7) Additional standards for "airstair doors" were added by Amendment 29-20.

(i) An analysis or test may be used to prove compliance with deformation standards in § 29.783(g)(1).

(ii) A sketch, drawing, or demonstration may be used to prove the door is useable for the conditions described in § 29.783(g)(2).

335. <u>§ 29.785 (through Amendment 29-19) SEATS, SAFETY BELTS, AND HARNESSES</u>.

a. <u>Explanation</u>.

(1) This section requires that seats, belts, harnesses, and adjacent parts of the rotorcraft be substantiated for the structural loads resulting from the inertia forces of § 29.561 as well as normal flight and ground inertia forces on a 170-pound occupant. The inertia forces of § 29.561 are ultimate loads and must be multiplied by a factor of 1.33 in determining the "strength of attachment" of each seat to structure and each belt or harness to structure. The seat, belt, etc., are required to sustain applied loads and to protect the occupant from serious injury. The pilot seats must also sustain the effects of the pilot forces of § 29.397.

(2) In addition, the "occupant must be protected from head injury" by the seat belt and one of the following:

(i) A harness to prevent the head from contacting an injurious object. *

* (ii) Elimination of injurious object within striking distance of the head.

(iii) A cushioned rest as specified.

(3) Handholds are required to steady occupants using the aisle in moderately rough air.

(4) Projecting objects which would injure occupants "in normal flight must be padded."

b. Procedures.

(1) Each seat with its belts and harnesses are to be substantiated for the flight, ground, and emergency landing loads of § 29.561 by structural test or stress analysis. Section 29.785(b) states that "each seat must be approved." FAA approval can be gained by Technical Standard Order (TSO) approval or by accomplishing sufficient structural substantiation to gain FAA approval of the seat and its belt(s) as part of the Type Design of the rotorcraft. TSO No. C-39 concerns standards for aircraft seats, including rotorcraft seats. If TSO No. C-39 is used as an approval basis for a specific rotorcraft seat, the seat should be checked to assure it has been substantiated for the vertical (up and down) and side loads imposed by installation in the aircraft. For example, TSO No. C-39 (and NAS 809) specifies an ultimate down load of 4.0g which is in agreement with the 4.0g emergency landing load factor of § 29.561, but it may be less than the design maneuver load factor (which can be as high as 3.5g limit or 5.25g ultimate).

(i) The 1.33 factor is specified for substantiation of attachments of each seat to the structure and each safety belt or harness to the seat or structure for § 29.561 loads, whether analysis or test is used.

(ii) If static testing of seats, belts, and harnesses is used, the body block of NAS 809 may be used. The corners of the NAS 809 body block may be radiused and padded if it is found that the small radii cause premature, unrealistic crippling of thin wall tubing or other structure used in the seat.

(iii) The substantiation of the pilot seats is required to include pilot forces of § 29.397 in conjunction with normal flight and ground loads. For example, the pilot foot force (195 pounds ultimate) must be reacted by the seat.

(2) The following criteria have been found satisfactory for preventing occupant head injuries: *

(i) If a harness is used, it should support the shoulders without applying hazardous loads to the side or front of the neck. It should be easily donned and a single point release with the seat belt is preferred. If separate release is provided, it must be simple, compatible with the seat belt release, and near the seat belt release. The harness should be tested in conjunction with the seat belt using a "body block" similar to that of NAS 809 if possible. If the harness is tested separately from the belt, it should be tested to 50 percent of the forward crash loads for the entire occupant weight of 170 pounds, unless that percentage distribution is found to be unrealistic by a rational analysis.

(ii) Elimination of injurious objects within striking distance of the head and other vital parts can be accomplished by removal of objects with sharp edges or rigid surfaces from within striking distance of vital parts of the occupant. Dimensions and weights for typical occupants are available in U.S. Army USAAULABS Reports 70-22 (August 1969) and 66-39 (June 1966) and NACA Report TN 2991 (August 1953). Because of the range of occupant head striking distance, a combination of "elimination of injurious objects" and "cushioned rests" may be required for some interior configurations.

(iii) An acceptable cushioned rest can be provided by use of a 1-inch thickness of foamed polyvinyl chloride (PVC), or equivalent energy absorbing material. The density of material should be in the 5 to 10 pounds per cubic foot density range. PVC foam has the property of <u>absorbing energy</u> efficiently with negligible rebond effects. PVC foam recovers slowly to the original configuration after deformation. If PVC foam is used, however, care must be taken in its application relative to its flammability characteristics (ref. § 29.853).

(3) Handholds for the occupants are generally provided by seat backs adjacent to the aisle. If the seat backs fold, the amount of support provided by the seat backs before they fold must be evaluated in a furnished interior or mock up. To provide adequate support, the seat back may use an easily disengaged latch or adequate friction in the hinge mechanism to obtain adequate support. Handholds along the aisle are, of course, not needed for rotorcraft with no aisles or where seat belts must be fastened during flight.

(4) Projecting objects which could injure occupants in normal flight should be padded. The amount of padding required depends on the location, size, and minimum radius of the projecting object. In general, this requirement will mean that sharp edges must be padded with one-half inch of PVC foam or equivalent (5 to 10 lbs. density), while objects with radii in excess of 1 inch may meet the requirements of § 29.785(e) with a lesser amount of energy absorbing padding, if it can be contacted only by persons "moving about in the rotorcraft in normal flight." *

* 336. § 29.787 (through Amendment 29-19) CARGO AND BAGGAGE COMPARTMENTS.

a. Explanation.

(1) This section requires that cargo and baggage compartments be designed for normal flight and ground loads and for a 4g ultimate forward load condition. Maximum placarded weights and critical distributions are to be considered.

(2) Means to prevent cargo shifting and contact between any cargo lamp bulb and cargo is to be provided.

b. Procedures. Structure tests or analyses may be used for substantiation for the design loads.

(1) Nets or straps may be used to prevent cargo shifting. The nets or straps are required to be substantiated for the structural loads. They need a means for adjustment to assure proper restraint for different sizes and shapes of cargo.

(2) Cargo lamp bulbs need to be guarded, recessed, or placed in upper inside corners to prevent contact with cargo. *

337. § 29.801 DITCHING. (RESERVED).

338. § 29.803 (through Amendment 29-19) EMERGENCY EVACUATION.

a. Explanation. This regulation specifies that "means for rapid evacuation in a crash landing" be provided considering the landing gear extended or retracted, and "considering the possibility of fire." Any external exits, whether normal entrance doors or service doors, can be considered as emergency exits if the requirements of §§ 29.805 through 29.815 are met. "Limited amphibian rotorcraft" emergency exits are required to be designed for probable maximum local water pressure (or shown to have nonhazardous failure characteristics) and to have a specified number of exits above the water level. Limited amphibian rotorcraft are approved under the provisions of §§ 29.519 and 29.755(b). Sections 29.801 and 29.807(d) refer to similar standards that pertain to "rotorcraft ditching configurations."

b. Procedures. Exits, arrangement, markings, access, and aisle widths as specified in § 29.805 through 29.815 are to be provided. Recent rotorcraft designs have been approved under the "ditching" standards of § 29.801. Previous "limited amphibian rotorcraft" were designed to the same standards.

339. § 29.805 (through Amendment 29-19) FLIGHTCREW EMERGENCY EXITS.

a. Explanation. Flightcrew emergency exits are required when passenger exits are not convenient. The placement of litters, cargo, or bulkheads may prevent passenger exits from being convenient to the flightcrew. Flightcrew exits, if required, are to be of sufficient size and located on both sides of the rotorcraft (or one top hatch) to "allow rapid evacuation of the flightcrew." A test or tests are required.

b. Procedures. Flightcrew emergency exits, if required, may consist of one overhead hatch or two side exits (one on either side). The size is not explicitly defined except that it be "of sufficient size . . . to allow rapid evacuation of the flightcrew." The ability for "rapid evacuation" should be demonstrated by test. For side exits located immediately adjacent to the crew seat and exceeding Type IV exits (§ 29.807) in size, the test demonstration can be accomplished by normal use and evaluation of the exits by the FAA crew during Type Inspection Authorization (TIA) testing. For any overhead exit or side of fuselage exits not meeting Type IV dimensions, a special demonstration test should be accomplished. This demonstration should show that 2.5 percentile to 97.5 percentile men could egress rapidly through the crew exit(s), i.e., men 5 feet 5 inches to 6 feet 2 inches in height and up to 210 pounds in weight. *

* 340. § 29.807 (through Amendment 29-19) PASSENGER EMERGENCY EXITS.

a. Explanation. The normal passenger exits (type and number in each side of fuselage) are specified as follows:

(1) For overland operations.

Passenger Seating Capacity	Emergency exits (rectangular with corner radii of width/3) for each side of the fuselage			
	Floor level			Step-up -29" Max.
	Type I 24" X 48"	Type II 20" X 44"	Type III 20" X 36"	Type IV 19" X 26"
1 through 10				1
11 through 19			1 or	2
20 through 39		1		1
40 through 59	1			1
60 through 79	1		1 or	2

(2) For overwater operations (related to ditching an optional standard).

Passenger Seating Capacity	Emergency exits (rectangular with corner radii of width/3) for each side of the fuselage	
	Threshold Above Waterline	
	Type III 20" X 36"	Type IV 19" X 26" w/step-up - 29"
1 through 9		1
10 through 35	1*	
Each Additional Unit of 35	1*	

* The passenger seat-to-exit ratio may be increased by using larger exits if proven by analyses or tests.

(3) For crash rollover conditions. Sufficient top, bottom, or ends of fuselage exits are to be provided for evacuation unless the probability of the rotorcraft coming to rest on its side in a crash landing is extremely remote.

(4) Ramp exits to replace Type I or II exits are permitted.

(5) Each emergency exit must be functionally tested.

b. Procedures.

(1) The number and size of overland and overwater operation exits will be as specified. The use of oversize exits is allowed if the threshold is flat and of the specified width. *

(2) The top, bottom, or end fuselage exits should be provided unless features of design are provided which prevent the rotorcraft from coming to rest on its side in a crash landing, and unless sufficient fail-safe and fatigue tests and analyses are conducted of the landing gear and support structure to show it is unlikely that the rotorcraft will come to rest on its side as a result of a single structural failure. An analysis is generally necessary to prove compliance with § 29.807(c).

(3) Ramp exits may be used in place of one Type I or one Type II exit if the required Type I or Type II exit is impractical, and if the § 29.813 exit access requirements are met by the ramp exits.

(4) Each emergency exit is to be opened from the inside and the outside as a functional test. Interior panels and seats should be installed for the exit functional tests to check for interferences and other effects. Section 29.813 pertains to access to the exits.

341. § 29.809 (through Amendment 29-19) EMERGENCY EXIT ARRANGEMENT.

a. Explanation. Emergency exits are to be provided which result in an unobstructed opening to the outside. The following emergency exit requirements are the same as passenger door requirements of § 29.783 and noted for convenience.

(1) Openable from inside or outside.

(2) Simple and obvious means for opening.

(3) Means for locking.

(4) Means to prevent opening in flight inadvertently or as a result of mechanical failure.

(5) Means to minimize jamming in a minor crash landing.

NOTE: In addition the following emergency exit requirements are: (1) the means of opening may not require exceptional effort; and (2) a slide (for floor level exits) or rope must be provided as prescribed for exits whose thresholds are more than 6 feet from the ground (unless located over the wing). Sections 29.1411(c) and 29.1561 contain other standards for the descent devices.

b. Procedures. Subparagraphs 1 through 5 of the above explanation are covered in the procedure for § 29.783, paragraph 334 of this document. *

* (1) The effort required to open the exit can be evaluated when the tests of § 29.807(f) are conducted. If the effort required to open the exit is in the range of 40 to 50 pounds, it is recommended that a person of slight stature, such as a female in the 90 to 110 pound weight range, be used for the exit opening demonstration/test. In any case, the average load required to operate the exit release mechanism and open the exit should not exceed 50 pounds, and the maximum individual load of a test series should not exceed 55 pounds.

(2) If an approved escape slide, or its equivalent, is provided for exits more than 6 feet from the ground with the landing gear extended, it should be located near the door and conspicuously marked. Automatic inflation and deployment under emergency conditions are the preferred means of operation but are not required by § 29.809. If automatic inflation and deployment features are provided, design features should prevent inadvertent deployment if the exit is a door used for normal entry and/or service. If manual deployment methods are used, they must be simple and easily carried out by a person of slight built and strength. The slide should rapidly inflate upon deployment. See § 29.809(f) for standards concerning an escape rope.

342. § 29.811 (through Amendment 29-19) EMERGENCY EXIT MARKING.

a. Explanation. This regulation covers both the marking and illumination by emergency lighting.

(1) Locating and marking signs are specified for each emergency exit with the following features:

(i) Locating signs and marking signs are to--

(A) Be recognizable from a distance equal to the width of the cabin;

(B) Have 1-inch white letters on a 2-inch red background (colors may be reversed); and

(C) Be self- or electrically illuminated to a minimum brightness of 160 microlamberts.

(ii) Locating signs visible to occupants approaching along the main aisle are required for each exit.

(A) The sign is required next to or above the aisle for floor level exits.

(B) Bulkheads or dividers obscuring exits must have exit locating signs except as stated.

(2) Exit operating or release handle instructions are to be-- *

(i) Readable from a distance of 30 inches; and

(ii) Supplemented with a red arrow and sign (for Type I or Type II exits with a handle having rotary motion) with the following features provided:

(A) A red arrow with three-fourths inch shaft, a head of twice the shaft width, and 70° arc at 75 percent of handle length.

(B) The word "open" in red letters 1 inch high near the head of the arrow.

(3) Independent source of light, as prescribed, is to be installed to--

(i) Illuminate marking and locating signs;

(ii) Provide general lighting of 0.05 foot-candles at 40-inch intervals at armrest height along the main aisle; and

(iii) Operate manually and automatically in a crash landing and when the normal electrical power is interrupted.

(4) External exit markings are required which include a 2 inch wide band around the exit, identification, and instructions for opening. The external markings are to have a reflectance difference of 30 percent from the fuselage surface finish.

(5) Emergency exits signs may read simply "EXIT."

(6) Excess exits should meet all of the "EXIT" standards or should not be identified as an exit.

b. Procedures.

(1) Emergency exit locating signs may be located to the side of the aisle for small fuselage heights, rather than over the aisle where it may present a hazard to the occupant's head and possibly impede egress. For small passenger cabins one self-illuminated sign stating "EXIT" may be used as both the locating and marking sign for an individual exit on one side of the cabin (operating instructions will, of course, still be required). If one "EXIT" sign is used to both locate and mark the exit, it should be attached to the fuselage above the exit and not to the exit itself. If it is attached to the exit itself and the exit is discarded from the cabin after opening, the locating function of the exit sign is lost when the exit is removed. That is, there is no sign to locate the exit for passengers other than for the one who discarded the exit. The exit locating sign is a necessity to direct all occupants.

(2) Operating instructions should be provided as specified. They should be kept short but clear; e.g., "rotate handle," "push," "pull," etc. *

* (3) Lighting should be provided as specified to illuminate the cabin for egress paths and to supplement lighting of the exit operating instructions signs.

(4) The reflectance of external exit markings can be checked by appropriate electro-optical instrumentation or by use of photometer card sets. Advisory Circular No. 20-47 provides information for complying with identical standards contained in § 25.811. These are also acceptable for § 29.811. The Munsell Color Company, 2441 North Calvert Street, Baltimore, Maryland 21218, provides a set of cards which includes shades of most commonly used colors.

343. <u>§ 29.813 (through Amendment 29-19) EMERGENCY EXIT ACCESS</u>.

a. <u>Explanation</u>. Paragraph (a) of § 29.813 prescribes design details for passageways, both between passenger compartments and for access to Type I and II emergency exits, should they be provided. Such passageways are not made mandatory by § 29.813 although most larger rotorcraft have used them. Some utility or "wide-body" rotorcraft may have open areas between the crew area (pilots) and passenger area (cabin). These configurations may have lateral seating arrangements providing access to emergency exits of Type I or II size, even though they may not be required by § 29.807(b). These designs may not have a main aisle.

(1) Paragraph (c) of this standard concerns access to Type III and Type IV exits. Although "passageways" with explicit requirements are not required for Type III and Type IV exits, "<u>access</u> from each aisle to each Type III and Type IV exit" is required.

(2) For exits whose thresholds are more than 6 feet above the ground, additional space adjacent to the exit is required to allow room for a crewmember to assist passengers with the descent device such as an escape slide or rope noted in § 29.809(f).

(3) In addition to requiring passageways and crewmember space adjacent to exits over 6 feet above the ground, this standard does not allow obstructions in the projected opening of Type III or Type IV emergency exits for one seat width from the exit, except as noted. For passenger seating configurations of 19 or less, minor obstructions into the projection of the exit are allowed only if "compensating factors to maintain the effectiveness of the exit" are provided.

b. <u>Procedures</u>.

(1) The provision for unobstructed passageways, at least 20 inches wide as specified, is straightforward for medium or large cabins with a main aisle and a typical rectangular floor plan. Care should be taken to assure that seats (with lateral or fore-and-aft movement) or galleys (with doors or drawers) are not installed so that they can encroach upon the required passageway. Design features such as stops in seat tracks, seat back mechanisms, stops in galley door (or drawer) mechanisms may be required to assure that unobstructed passageways are provided. *

(2) The requirement (added by Amendment 29-12) that "access from each aisle to each Type III and Type IV exit" be provided may add design features to the interior of many typical compact interiors of medium-size rotorcraft. Rotorcraft with emergency exits located in either hinged or sliding doors and having passenger area encroachment or protusions by compartments for fuel cells, gear boxes, etc., may require special design features to assure that passengers seated to one side or one area of the cabin have "access" to all Type III or Type IV exits on the same or other side of the rotorcraft. The cabin must not be separated into compartments or partitioned. For example, fold down seat back mechanisms may be required for compact cabin configurations having only lateral aisles rather than longitudinal aisles and having Type III or Type IV exits located on each side of the cabin at the end of the lateral seat row or rows.

(3) The space adjacent to an exit that requires a crewmember to assist passengers with descent devices must be large enough to prevent the crewmember from becoming an obstruction in access to the exit. Twenty inches of access must be maintained.

(4) Minor obstructions are allowed in the projected opening of Type III or Type IV exits (for 19 or less passenger seat configurations) if "compensating factors to maintain the effectiveness of the exit" are provided. Compensating factors may include such design features as larger than required exit opening, additional exits beyond the minimum number required, or steps or other assist features which facilitate egress through the exit with the obstruction. Test or analysis may be required to prove the effectiveness of the compensating feature.

344. <u>§ 29.815 (through Amendment 29-19) MAIN AISLE WIDTH.</u>

a. <u>Explanation.</u> Main aisle widths are specified in the following table:

Passenger seating capacity	Minimum main passenger aisle width: Less than 25 inches from floor	Minimum main passenger aisle width: 25 inches and more from floor
	Inches	Inches
10 or less	12*	15
11 through 19	12	20
20 or more	15	20

*A narrow width not less than 9 inches may be approved when substantiated by tests found necessary by the Administrator.

*

* b. Procedures.

(1) Provide the specified aisle minimum width where a longitudinal main aisle is provided in the type design.

(2) Historically, certain rotorcraft with short, wide cabins were initially designed without a longitudinal main aisle for military and cargo use, but were later fitted and approved for civil passenger configuration. These craft generally have 19 or less passenger seats and have either (1) outboard facing passenger seats, (2) a limited number of lateral rows with fold down seats/seat backs, or (3) a combination of lateral and longitudinal rows with and without main aisles to facilitate entrance and egress. *

345. § 29.831 VENTILATION. (RESERVED)

346. § 29.833 HEATERS. (RESERVED)

347.-356. RESERVED.

SECTION 21. FIRE PROTECTION

357. § 29.851 FIRE EXTINGUISHERS. (RESERVED)

SECTION 27. FUEL SYSTEM COMPONENTS

483. § 29.991 (through Amendment 29-19) FUEL PUMPS.

a. Explanation.

(1) Section 29.991, paragraph (a), provides a definition of the main pump(s) and § 29.991, paragraph (b), requires an "emergency pump(s)." The main pump(s) that is certified as part of the engine does not fall under § 29.991 requirements. The main pump(s) discussed under § 29.991 should therefore be considered "main aircraft pump(s)."

(2) The main aircraft pump(s) consists of whatever pump(s) is required to meet engine or fuel system operation throughout the range of ambient temperature, fuel temperature, fuel pressure, altitude, and fuel types intended for the rotorcraft. If the main aircraft pump(s) is required to meet the above criteria, then an emergency pump(s) is required.

b. Procedures.

(1) Each pump classified as a main aircraft pump, which is also a positive displacement pump, must have provisions for a fuel bypass. An exception is made for fuel injection pumps used on certain reciprocating engines and for the positive displacement, high pressure, fuel pumps routinely used in turbine engines. The bypass may be accomplished via internal spring check valve and fuel passage, or by external plumbing and a check valve. High capacity positive displacement pumps with internal pressure relief and recirculation passages should be checked for overheating if they may be expected to operate continuously at or near 100 percent recirculation.

(2) Section 29.991, paragraph (b) specifies a requirement for "emergency" pumps to provide the necessary fuel after failure of any (one) main aircraft pump. (Injection pumps and high pressure pumps used on turbine engines are exempt.) As stated in this rule, the "emergency" pump must be operated continuously or started automatically to assure continued normal operation of the engine. For some multiengine rotorcraft, another main aircraft pump may possibly be used as the required "emergency" pump. In this case, the dual role of this pump requires it to have capacity to feed two engines at the critical pressure/flow condition. Availability of fuel flow from this backup pump must be automatic and this function should be verified in the preflight check procedure. For Category A rotorcraft, a comprehensive fault analysis of the fuel system is mandatory to assure compliance with § 29.903, paragraph (b).

(3) Section 29.991, paragraphs (c)(1)(i) and (ii) address the situation, usually associated with supercharged reciprocating engines, where fuel pressure must be modulated with respect to carburetor deck pressure. This is accomplished with interconnecting air lines from the carburetor intake (after the supercharger) to the pressure relief connection on the fuel pump(s). A similar connection from the carburetor intake to the vented side of the fuel pressure gauge is needed to obtain correct fuel pressure reading. These systems may require orifices and/or surge chambers to operate correctly.

(4) Section 29.991, paragraphs (c)(2) and (3) requires seal drains which drain safely. A drain impingement test is normally required to verify safe drainage. Use of a colored dye to simulate fuel discharge at the drain line exit or a fluid sensitive coating (Bon Ami) on the aircraft skins will facilitate evaluation of the safety aspects of drain impingement. Pump seal drain requirements would not be applicable for tank immersed pumps.

484. § 29.993 (through Amendment 29-19) FUEL SYSTEM LINES AND FITTINGS.

a. Explanation. This rule outlines design requirements for fuel system lines.

b. Procedures.

(1) Compliance is usually obtained by employing routing and clamping as described in paragraph 709, Chapter 14, Section 2 of AC 43.13-1A and by monitoring the arrangement throughout the developmental and certification test period. Requirements for approved flexible lines may be resolved by utilizing lines listed as TSO C53a approved for installation in either normal or high temperature areas as appropriate.

(2) Verify adequate clearance exists between lines and elements of the rotorcraft control system at extremes of control travel, including control deflections and, for flexible lines (hoses), possible variations in routing.

(3) Flexible lines inside fuel or oil tanks require special evaluation to assure that the external surfaces of these lines are compatible with the fluids involved and that fluid sloshing will not cause line failure. Lines inside tanks should be routed to avoid impingement by fuel or oil filler nozzles.

(4) Good design practice suggests that all flammable fluid lines should be routed to minimize the possibility of rupture in the event of a crash or from engine rotor disc failure.

485. § 29.995 (through Amendment 29-19) FUEL VALVES.

a. Explanation. This regulation requires that fuel valves be supported so that no loads resulting from their operation or from accelerated flight conditions are transmitted to the lines attached to the valve.

b. Procedures. Compliance with this rule is usually accomplished by designing the installation of the fuel valve so that the valve is supported by either primary or secondary airframe structure. *

486. <u>§ 29.997 (through Amendment 29-19) FUEL STRAINER OR FILTER</u>.

a. <u>Explanation</u>. This rule provides for a main in-line fuel filter designed to collect all fuel impurities which could adversely affect fuel system and engine components downstream of the filter. The rule also requires a sediment bowl and drain (or that the bowl be removable for drain purposes) to facilitate separation of contaminations, both solid and liquid, from the fuel.

b. <u>Procedures</u>.

(1) The filter should be mounted in a horizontal segment of the fuel line to facilitate proper action of the sediment bowl. If the filter is located above the fuel tank, it becomes necessary to activitate a fuel boost pump to achieve positive drainage of the filter bowl. Without pump pressure, air may enter the fuel system during the filter draining operation and, for turbine engines, result in transient power surges or engine failure during subsequent engine operation. A flight manual note to require pump(s) to be "on" during filter draining would be appropriate.

(2) Section 29.997(d) sets forth a requirement for filter capacity and for filter mesh. The capacity requirement may be substantiated by showing that the filter, when partially blocked by fuel contaminates (to a degree corresponding to the indicator marking or setting required by §29.1305(a)(17)), does not impair the ability of the fuel system to deliver fuel at pressure and flow values established as minimum limitations for the engine. The filter mesh must be sized to prevent passage of particulate which cannot be tolerated by the engine. FAR Part 33 requires that the degree and type of filtration be established. This information should be the base for selecting the filter mesh. Although a test may be devised and conducted, data from the filter manufacturer usually are acceptable to verify compliance. Note that when the filter capacity is reached, continued flow of contaminated fuel may result in engine failure. A flight manual note regarding precautionary procedures is appropriate.

(3) FAR Part 33 (through Amendment 33-6) has an identical requirement for a fuel filter for engine fuel systems; however, it is not intended that two filters should be required.

* 487. § 29.999 (through Amendment 29-19) FUEL SYSTEM DRAINS.

a. Explanation. This regulation provides for fuel system drains and defines the requirements which the system must meet.

b. Procedures.

(1) The location and function of the fuel system drains are an integral part of any fuel system. There may be several drains required dependent upon the fuel system design. Each fuel tank sump and certain types of fuel strainers or filters require a means to drain (ref. §§ 29.971 and 29.997).

(2) Selection of the location and orientation of the drain discharge in the design phase is important to assure that there is no impingement on any part of the rotorcraft. To show compliance with the requirement may require tests dependent upon whether the applicant has a previously approved design which is similiar, or if the system is a new design for which no previous experience is available.

(3) The location of the drain valve should be selected so that the requirements for accessibility, ease of operation, and protection are met.

(4) Advisory Circular 20-119 provides an acceptable means, but not the only means, of compliance with the requirement for positive locking of fuel drain valves in the closed position.

(5) The fuel drain installation on aircraft with retractable landing gear will be satisfactory if recessed within the outside surface of the aircraft.*

488.-497. RESERVED.

* 501. <u>§ 29.1017 (through Amendment 29-19) OIL LINES AND FITTINGS</u>.

a. <u>Explanation</u>. This regulation outlines the certification requirements for oil lines and fittings.

b. <u>Procedures</u>. The oil system lines and fittings are required to meet the requirements of § 29.993; therefore, the routing and clamping described in Par 709, Chapter 14, Section 2, of AC 43.13-1A may be utilized as guidance for the system design. An evaluation carried out through the development and certification test period will usually surface any problems of interference and/or vibration.

(1) When flexible hoses are used in the lubrication system they must be substantiated. Hoses listed in TSO C53a may be used which would preclude certain substantiation requirements.

(2) *Location of the breather lines and discharge should be carefully* evaluated to determine that the requirements of this paragraph are followed.

(3) The routing of fluid lines should be such that drooping lines and fluid traps which are undrainable are avoided.

502. <u>§ 29.1019 (through Amendment 29-19) OIL STRAINER OR FILTER</u>.

a. <u>Explanation</u>. This regulation defines the requirements for the engine oil system strainer or filter. If a strainer or filter which meets the requirements of this paragraph is incorporated as part of the type certificated engine, an additional airframe filter is not required.

b. <u>Procedures</u>. This paragraph requires an oil strainer or filter through which all of the oil flows for each turbine engine installation. The strainer or filter should be sized to allow oil flow at the flow rates and within the pressure limits as specified in the engine requirements. The effect of oil at the minimum temperature for which certification is sought should be accounted for.

(1) For each oil strainer or filter required by § 29.1019(a) which has a bypass, the bypass should be sized to allow oil flow at the normal rate through the oil system with the filtration means completely blocked.

(2) For each oil strainer or filter installed per this rule, the capacity must be such that the oil flow and pressure are within the operating limits established for the engine. The mesh requirements are determined by the engine specification for the filtration of particle size and density. *

* (3) Section 29.1019(a)(3) requires an indicator that will show when the contaminant level of the filtration system, as specified in § 29.1019(a)(2), has been reached. The indicator should signal a contaminant level which has not caused the filter to go into a bypass condition. Consideration should also be given so that the contaminant level at which the indicator is activated is such that the filter would not bypass during a flight time based on full fuel at a cruise condition with the lubricant contaminated to the degree used to show compliance with § 29.1019(a)(2).

(4) An evaluation of the construction and location of the bypass associated with the strainer or filter should be accomplished. The appropriate installation of the filter based on this evaluation would preclude the release of the collected contaminants in the bypass oil flow.

(5) If an oil strainer or filter installed in compliance with this regulation does not have a bypass, there must be a means to connect it to the warning system required in § 29.1305(a)(18). This warning should indicate to the pilot the contamination before it reaches the capacity established in § 29.1019(a)(2). Section 29.1019(b) covers the blocked oil filter requirements associated with reciprocating engine installations. The lubrication system should be such that the normal oil flow will occur with the filter completely blocked.

503. <u>§ 29.1021 (through Amendment 29-19) OIL SYSTEM DRAINS</u>.

a. <u>Explanation</u>. This regulation requires provisions be provided for safe drainage of the entire oil systems and defines certain requirements for assuring that no inadvertent oil flow occurs from the system provided.

b. <u>Procedures</u>. The design of the oil system must provide a means for safe drainage of the entire oil system. This may require one or more drains dependent upon the design of the system. If a valve is used for this function, it must provide a means for a positive lock in the closed position. The method by which the lock is accomplished may be manual or automatic.

504. <u>§ 29.1023 (through Amendment 29-19) OIL RADIATORS</u>.

a. <u>Explanation</u>. This regulation defines the installation requirements to be considered for oil system radiators.

b. <u>Procedures</u>.

(1) The primary concern with respect to oil radiators is that they are sized to provide the required heat rejection and to provide adequate fluid flow within the prescribed pressure limits. *

* 499. § 29.1013 (through Amendment 29-19) OIL TANKS.

a. Explanation This regulation identifies the requirements that each oil tank must meet. It also specifies that the oil tank installation must meet the installation requirements of § 29.967.

b. Procedures.

(1) The oil tanks usually are constructed of aluminum, aluminum alloy, or stainless steel and are of such a design to permit installation in the aircraft as close to the engine as the design allows. The choice of materials will generally be determined by the selected location of the tank. The tank envelope or outline will generally be determined by the location within the structure of the helicopter.

(2) The design of the tank is required to meet the expansion space requirements as specified in the regulation for the particular installation. This is generally accomplished by locating the filler cap in such a manner that the expansion space cannot be inadvertently filled with the rotorcraft in normal ground attitude.

(3) The tank is required to be properly vented and the vent requirements are identified in the regulation.

(4) Unless alternate means are provided, it is good design practice to locate the oil tank with respect to the engine so that when the helicopter is in its normal ground attitude, a positive head to the oil pump inlet is provided.

(5) Sections of the regulation address specific requirements when Category A certification is requested.

(6) The designer should be aware of the requirements associated with the location of the oil tank outlet and the marking requirements specified in § 29.1557(c)(2).

(7) Flexible oil tank liners may be used; however, they must be approved or shown to be suitable for the particular installation.

(8) An "external oil system" which is defined as being those components, lines, etc., of an oil system which are outside the engine and not supplied as part of a certificated engine. The components of such a system which are within the fire zone and required to be fire resistant. Those outside the fire zone need not be fire resistant. *

500. § 29.1015 OIL TANK TESTS. (RESERVED)

SECTION 28. OIL SYSTEM

* 498. § 29.1011 (through Amendment 29-19) GENERAL.

a. Explanation.

(1) This regulation defines the oil system requirements for engines and the rotor drive systems which require continuous lubrication.

(2) Each engine must have an independent oil system. This lubrication system performs two functions. It provides an adequate oil supply and it incorporates a means of cooling the hot oil discharged from the engine. Acceptable oil quantities must be determined and the system designed to accommodate this quantity.

(3) The adequacy of the oil systems to maintain the oil temperature at or below the specified limits must be shown under the applicable requirements of §§ 29.1041 through 29.1049.

b. Procedures.

(1) To provide engine isolation, meeting the regulations requires completely independent lubrication systems in multiengine rotorcraft. This includes separate oil tanks, oil coolers, oil cooler blowers, and the associated plumbing. Alternate designs providing an equivalent level of independence may be used if substantiated. A single failure of any one system may not--

(i) Prevent continued safe operations of the remaining engine(s); or

(ii) Require immediate action by any crewmember for continued safe operations.

(2) Any rotor drive system which requires continuous lubrication must be sufficiently independent to ensure operation with any engine inoperative. The oil cooling provisions of this system should be sufficient to provide adequate cooling at maximum continuous power of the drive system and critical conditions associated with any single engine failure. The system must also provide for a safe autorotation.

(3) The usable oil quantity for the oil systems may be determined in several ways and the required quantities will vary between piston engine installation and turbine engine installation. Several methods of determining the usable oil capacity are provided in the regulations. However, when oil-fuel ratios other than those prescribed are used, they must be substantiated by data on the oil comsumption of the engine. *

* (3) In showing compliance with the cooling requirements, the applicant should not be required to exceed rotorcraft established limits (gross weight, drive system torque, measured gas temperature, etc.), aircraft power required, or power available. The applicant may elect, however, to exceed these limits in order to minimize test points by conservative testing, or to anticipate future growth (increased gross weight etc.).

(4) The need for a comprehensive cooling test plan prior to certification testing cannot be overemphasized. Highly derated engine installations, the relationship of power required to power available, the use of bleed air devices which would increase the measured gas temperature while aircraft power required remains the same, auxiliary cooling provisions, and the increase in engine temperatures with engine deterioration are factors which could affect the selection of cooling demonstration test points. The following paragraphs will provide some general guidance, but the cooling test plan is the key to a successful program.

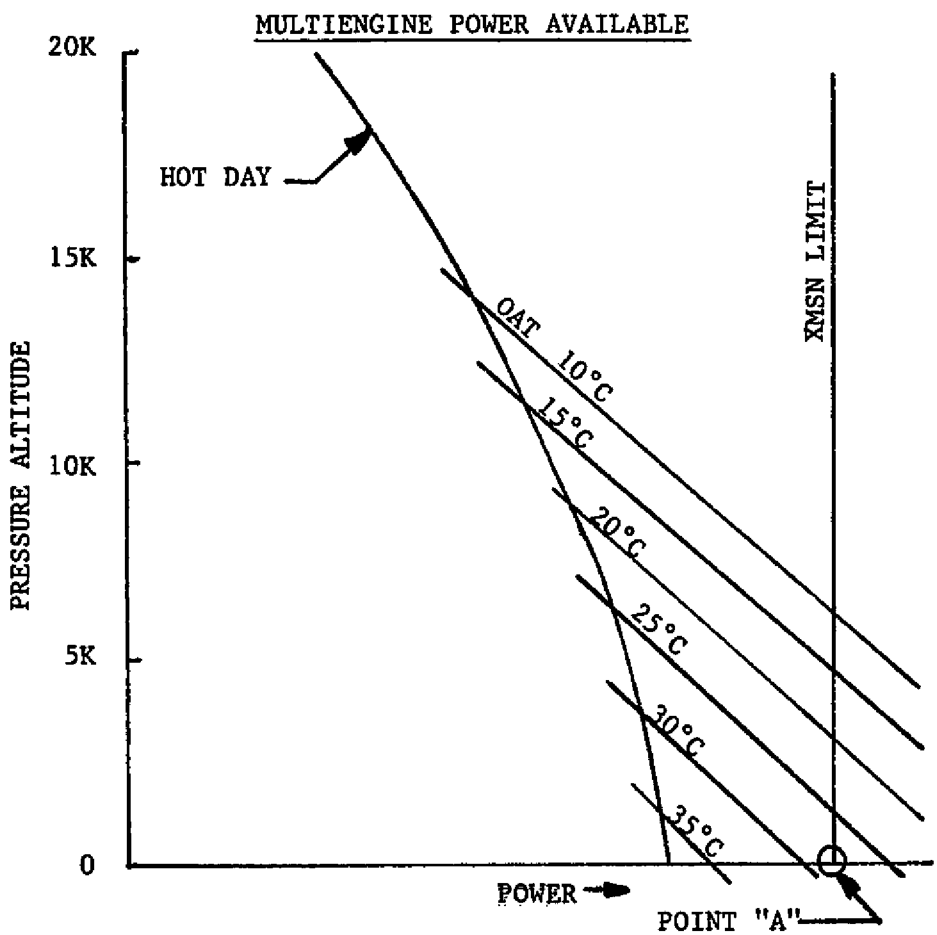

FIGURE 516-1 ADDITIONAL COOLING TEST POINT

*

* 517. § 29.1043 (through Amendment 29-19) COOLING TESTS.

a. Explanation.

(1) Section 29.1043(a) requires that certain ambient temperature correction factors be applied unless testing is accomplished at the maximum ambient atmospheric temperature prescribed.

(2) No corrected temperatures may exceed established limits.

(3) The statement of § 29.1043(a)(4) which requires that test procedures be in accordance with §§ 29.1045 through 29.1049 does not limit testing to the conditions prescribed in those sections. Section 29.1041(a) and (b) provide the basis for examination of other operating and shutdown conditions.

(4) The maximum ambient atmospheric temperature must be at least 100° F at sea level, lapsed to altitude at a rate of 3.6° F per 1,000 feet pressure altitude. The applicant may select a lower maximum ambient atmospheric temperature for winterization installations.

(5) Unless a more rational correction applies, the temperature data (except for cylinder barrels) are to be corrected by adding the difference between the maximum ambient atmospheric temperature and the temperature of the ambient air at the time of the first occurrence of the maximum component or fluid temperature recorded during the cooling test.

(6) Cylinder barrel temperature data are corrected in a similar manner to other components except 0.7 times the difference between the maximum ambient atmospheric temperature and the ambient temperature at the first occurrence of the maximum cylinder barrel temperature is applied.

b. Procedures.

(1) Seldom is testing actually accomplished at the maximum required ambient temperature of at least 100° F at sea level lapsed 3.6° F per 1,000 feet pressure altitude. Component and fluid temperatures must therefore be corrected to derive the item temperature that would have been reached if the test day had matched exactly the maximum ambient temperature day. The applicant may select a higher maximum ambient temperature for cooling certification than the 100° F sea level hot day prescribed. Provisions are also made for selecting a maximum ambient temperature less than the 100° F sea level hot day for winterization installations not intended to function at the hot day conditions.

(2) When cooling test ambient conditions are cooler than the selected or prescribed hot day conditions, the applicant may take advantage of cooling air or fluid flows that would exist at hot day conditions. For example, thermostatically controlled oil cooler flow could be set for hot day conditions. *

SECTION 29. COOLING

* 516. § 29.1041 (through Amendment 29-19) GENERAL.

a. Background.

(1) Few substantive changes have been made to the cooling provision requirements, §§ 29.1041 through 29.1049, since the rules were defined in the Civil Air Regulations, Part 7, effective August 1, 1956. Testing procedures utilized have not precisely followed those rigorously set forth in §§ 29.1045 through 29.1049 as industry and the FAA have recognized the need to vary procedures slightly to accomplish the practical test objectives.

(2) In the paragraphs which follow, the cooling regulations will be explained, and in some instances where the regulations provide specific procedures, "alternative procedures" which have been found acceptable in achieving the rule objectives will be presented. The intent of providing those alternative procedures is not to promulgate new regulations, but rather to provide recognized, accepted procedures for compliance with the objective of the current standards.

b. Explanation.

(1) The rotorcraft design should provide for cooling to maintain the temperatures of all powerplant, auxiliary power unit, and power transmission components and fluids within the limitations established for these items.

(2) Cooling provisions should be adequate for shutdown and for water, ground, and flight operating conditions.

(3) The adequacy of the cooling provisions should be demonstrated by flight testing.

c. Procedures.

(1) Detailed procedures for the demonstration of climb, takeoff and climb, and hover cooling are given in §§ 29.1045 through 29.1049. Other test conditions and procedures necessary to demonstrate adequate cooling for water, ground, flight, and shutdown conditions must be negotiated between the applicant and the FAA certification engineer. A cooling test proposal which defines the agreed test points and procedures should be prepared well in advance of the official certification testing.

(2) The test conditions selected, in addition to those in §§ 29.1045 through 29.1049, would typically include cruise at various airspeeds and altitudes, shutdown after prolonged hover, and sling load cooling if applicable. One test condition which should be examined, particularly with regard to transmission cooling, is the point of highest multiengine mechanical power at the maximum ambient temperature. This is identified as test point "A" in figure 516-1. The selection of test points should be tempered with engineering judgment and based on results from similar aircraft, if such data are available. *

* (3) The component and fluid temperature correction factor to be applied when test ambients do not correspond to the hot day conditions is commonly called the "degree-for-degree correction." It may be possible to justify, and the regulation allows the application of a more rational, less conservative correction factor. A correction factor other than degree-for-degree should be based on engineering test data.

(4) No corrected temperatures may exceed established limits. In order to maintain temperatures within established limits, the applicant may be willing to accept lesser performance than the full capability of a device. For example, a starter/generator capable of cooling under test cell conditions to 200 amperes continuous load may be limited to a lesser value, perhaps to 150 amperes, when installed in the aircraft due to cooling considerations. This continuous load for cooling must be equal to or greater than the allowable continuous load designated on aircraft instruments.

c. Thermal Limit Correction.

(1) An important correction factor which is not discussed in the regulations, but is frequently necessary to show the cooling adequacy required by § 29.1041, is the thermal limit correction factor. This factor is sometimes required if, at test day conditions, the engine measured gas temperature does not correspond to that which would have occurred on a minimum specification engine at hot day conditions.

(2) The correction factor would not apply to those components not affected by changes in measured gas temperature (MGT) at a constant power. Typical items expected to be affected by changes in the MGT at constant power would be engine oil temperature, thermocouple harnesses, or other fluid, component, or ambient temperatures in the vicinity of the engine hot-section or exhaust gases. Other items remote from the hot-section, perhaps the starter-generator or fuel control, would not be expected to be influenced by MGT variations; however, the items affected and the magnitude of the factor to be applied should be established by testing.

(3) There are several acceptable methods for establishing the appropriate thermal limit correction factor during development testing. The general idea is to establish a stabilized flight condition, typically ground-run or IGE hover, and to vary the measured gas temperature at approximately fixed power and OAT conditions. This may be accomplished by utilizing engine anti-ice bleed air, customer bleed air, or by ingesting warmer than ambient air (either an external source or the engine bleed air) into the engine inlet. Care should be used in ingesting warmer than ambient air to assure that the warm air is diffused in order to avoid possible engine surge. *

* (i) If it is not possible to attain a suitable variation in MGT by these methods, an acceptable, but more conservative thermal limit correction may be obtained by allowing both shaft horsepower and MGT to vary at a stabilized flight condition and OAT.

(ii) The component temperature is plotted as a function of MGT, and the thermal limit correction from any test day MGT for any flight condition, to the MGT that would have existed with minimum specification engines on a hot day, is then applied to derive the final measured component temperature.

(4) In certain rare instances, it may not be required that the correction factor be applied to the full thermal limit capability of the engine. Consider the following example for the hot day hover IGE cooling test point at sea level.

	Power (SHP)	Corresponding MGT (°C)
Drive System Limit	900	---
Twin Engine Hot Day Power Available	1,050	750
Hot Day Power Required at Maximum G.W.	850	650
Engine Maximum Allowable MGT (Instrument Marking)	---	765
Test Day (90° F OAT) Parameters	850	600

(i) Notice that the installed hot day power available MGT from the engine performance program, is 15° C cooler than the limit MGT (750° vs. 765° C), thus the engine has 15° C "field margin" which would allow the engine temperature to gradually increase 15° C to maintain a given power as engine life is utilized. Secondly, the measured gas temperature corresponding to hot day power required at maximum gross weight, is less than that corresponding to either the drive system limit or twin engine hot day power available. Thus, the thermal limit correction could be applied from the test day MGT, 600° C, to the power required MGT plus the field margin, 650° C plus 15° C, rather than applying the correction factor to the full thermal capability of the engine, 765° C.

(ii) Care should be used in applying this relieving method, because as the hover altitude changes, the maximum gross weight and power required (and the associated MGT) will vary. The data must be corrected to at least the maximum MGT for a minimum specification engine that can occur in service at the flight condition under investigation. *

518. § 29.1045 (through Amendment 29-19) CLIMB COOLING TEST PROCEDURES.

a. Objective. The objective of the regulation is to verify, for Category A and for Category B rotorcraft described, that cooling provisions are adequate for a one-engine-inoperative (OEI) climb or descent initiated from a multiengine cruise at the critical altitude with stabilized component temperatures. The specific flight conditions and powers are described in the regulation.

b. Explanation.

(1) This regulation specifies climb or descent cooling with OEI for Category A rotorcraft and for Category B rotorcraft with Category A powerplant isolation and fireproof or isolated structure, controls, etc., which are essential for controlled flight and landing. For the Category B machine described, the testing should be accomplished at the steady rate of climb or descent established under § 29.67(b), i.e., at the best OEI rate of climb (or descent) and the remaining engine at maximum continuous power or 30-minute power, whichever is applicable.

(2) The engine whose shutdown has the most adverse effect on the cooling conditions for the remaining engine(s) and powerplant components should be inoperative.

(3) The regulation provides that the climb cooling test may be conducted in conjunction with the takeoff cooling test of § 29.1047. This possible combining of tests applies only to § 29.1047(a), since § 29.1047(b) is a multiengine climb and not related to the OEI climb procedures of § 29.1045.

c. Procedures.

(1) The OEI climb cooling test point begins from a multiengine cruise, with stabilized fluid and component temperatures, 1,000 feet below either the all-engine-critical altitude or the maximum altitude at which the rate of climb is 150 f.p.m., whichever is the lowest altitude. If the minimum altitude derived is less than sea level, the climb should begin from a twin engine cruise with stabilized fluid and component temperatures at the minimum practical altitude.

(i) The all-engine-critical altitude is the maximum altitude at which, for the ambient conditions prescribed, it is possible to maintain the multiengine specified power. For example, if for multiengine operations, the transmission maximum continuous torque can be maintained on the hot day to a maximum altitude of 10,000 feet above which power would have to be reduced because of gas temperature or other limitations, then 10,000 feet is the all-engine-critical altitude. Point "A" in figure 518-1 illustrates the all-engine-critical altitude.

(ii) The 150 f.p.m. climb criteria should be based on multiengine operation at maximum continuous power available at hot day conditions at maximum gross weight. *

* (iii) Fluid and component temperatures are considered stabilized when the rate of change is less than 2° F per minute.

(2) The OEI climb power to be utilized is 30-minute OEI hot day power available (if approval of 30-minute power on the aircraft is requested), followed by maximum continuous hot day power available. If 30-minute OEI power approval is not requested, the power to be utilized would be maximum continuous hot day power available.

(i) Rotorcraft for which approval of a continuous OEI power rating is requested would use the power available on a hot day at the maximum continuous OEI rating following the 30-minute OEI climb phase (or for the entire climb if approval of 30-minute OEI power is not requested).

(ii) If the maximum continuous OEI approval is not requested, then the highest hot day power available approved for continuous usage from the remaining engine(s) under OEI conditions would be used following the 30-minute OEI climb phase (or for the entire climb if approval of 30-minute OEI power is not requested).

(3) In order to achieve representative test results, the helicopter climb rate and airspeed should approximate those which would occur on a hot day. This is accomplished by adjusting helicopter gross weight as required to produce the desired climb rate based on published or predicted climb performance data. The possible adverse effects of climb fuselage attitude on cooling air duct entrances should be considered in the selection of center-of-gravity of the test aircraft.

(4) The OEI climb should be continued for at least 5 minutes after the occurrence of the highest temperature recorded or until the maximum certification altitude is reached. Generally, temperatures would be expected to peak a short time after the climb begins since component and fluid temperatures are stabilized prior to entry to the climb phase.

(5) For Category B rotorcraft, defined in § 29.1045(a)(2) without a positive OEI rate of climb, the descent should begin from a hot day maximum continuous power multiengine cruise, with stabilized fluid and component temperatures, at the all-engine-critical altitude.

(6) The descent should conclude at either the maximum altitude at which level flight can be maintained with one engine inoperative or at the minimum practical altitude, whichever is higher.

(7) The OEI powers available to be utilized during the descent would be the same as those prescribed previously for OEI climb cooling. OEI operation should continue until component and fluid temperatures stabilize. *

SECTION 31. EXHAUST SYSTEM

548. § 29.1121 GENERAL. (RESERVED)

549. § 29.1123 EXHAUST PIPING. (RESERVED)

550. § 29.1125 EXHAUST HEAT EXCHANGERS. (RESERVED)

551.-560. RESERVED.

SECTION 32. POWERPLANT CONTROLS AND ACCESSORIES

561. § 29.1141 POWERPLANT CONTROLS: GENERAL. (RESERVED)

562. § 29.1142 AUXILIARY POWER UNIT CONTROLS. (RESERVED

563. § 29.1143 ENGINE CONTROLS. (RESERVED)

564. § 29.1145 IGNITION SWITCHES. (RESERVED)

565. § 29.1147 MIXTURE CONTROLS. (RESERVED)

566. § 29.1151 ROTOR BRAKE CONTROLS. (RESERVED)

567. § 29.1157 CARBURETOR AIR TEMPERATURE CONTROLS. (RESERVED)

568. § 29.1159 SUPERCHARGER CONTROLS. (RESERVED)

569. § 29.1163 POWERPLANT ACCESSORIES. (RESERVED)

570. § 29.1165 ENGINE IGNITION SYSTEMS. (RESERVED)

571.-583. RESERVED.

SECTION 33. POWERPLANT FIRE PROTECTION

584. § 29.1181 DESIGNATED FIRE ZONES: REGIONS INCLUDED. (RESERVED)

585. § 29.1183 FLAMMABLE FLUID-CARRYING COMPONENTS. (RESERVED)

586. § 29.1185 FLAMMABLE FLUIDS. (RESERVED)

587. § 29.1187 DRAINAGE AND VENTILATION OF FIRE ZONES. (RESERVED)

588. § 29.1189 SHUTOFF MEANS. (RESERVED)

589. § 29.1191 FIREWALLS. (RESERVED)

590. § 29.1193 COWLING AND ENGINE COMPARTMENT COVERING. (RESERVED)

591. § 29.1194 OTHER SURFACES. (RESERVED)

592. § 29.1195 FIRE EXTINGUISHING SYSTEMS. (RESERVED)

593. § 29.1197 FIRE EXTINGUISHING AGENTS. (RESERVED)

594. § 29.1199 EXTINGUISHING AGENT CONTAINERS. (RESERVED)

595. § 29.1201 FIRE EXTINGUISHING SYSTEM MATERIALS. (RESERVED)

596. § 29.1203 FIRE DETECTOR SYSTEMS. (RESERVED)

597.-616. RESERVED.

SECTION 34. EQUIPMENT - GENERAL

617. § 29.1301 FUNCTION AND INSTALLATION. (RESERVED)

618. § 29.1303 FLIGHT AND NAVIGATION INSTRUMENTS. (RESERVED)

619. § 29.1305 POWERPLANT INSTRUMENTS. (RESERVED)

620. § 29.1307 MISCELLANEOUS EQUIPMENT. (RESERVED)

519. <u>§ 29.1047 TAKEOFF COOLING TEST PROCEDURES. (RESERVED)</u>

520. <u>§ 29.1049 HOVERING COOLING TEST PROCEDURES. (RESERVED)</u>

521.-530. <u>RESERVED</u>.

SECTION 30. INDUCTION SYSTEM

531. <u>§ 29.1091 AIR INDUCTION. (RESERVED)</u>

532. <u>§ 29.1093 INDUCTION SYSTEM ICING PROTECTION. (RESERVED)</u>

533. <u>§ 29.1101 CARBURETOR AIR PREHEATER DESIGN. (RESERVED)</u>

534. <u>§ 29.1103 INDUCTION SYSTEM DUCTS. (RESERVED)</u>

535. <u>§ 29.1105 INDUCTION SYSTEM SCREENS. (RESERVED)</u>

536. <u>§ 29.1107 INTER-COOLERS AND AFTER-COOLERS. (RESERVED)</u>

537. <u>§ 29.1109 CARBURETOR AIR COOLING. (RESERVED)</u>

538.-547. <u>RESERVED</u>.

* (8) The airspeeds utilized in the climb and descents should be representative of normal speeds unless cooling provisions are sensitive to rotorcraft airspeed, in which case the airspeeds most critical for cooling should be used. In no case, however, should it be required that the selected airspeeds exceed the speeds established under §§ 29.67(a)(2) and 29.67(b).

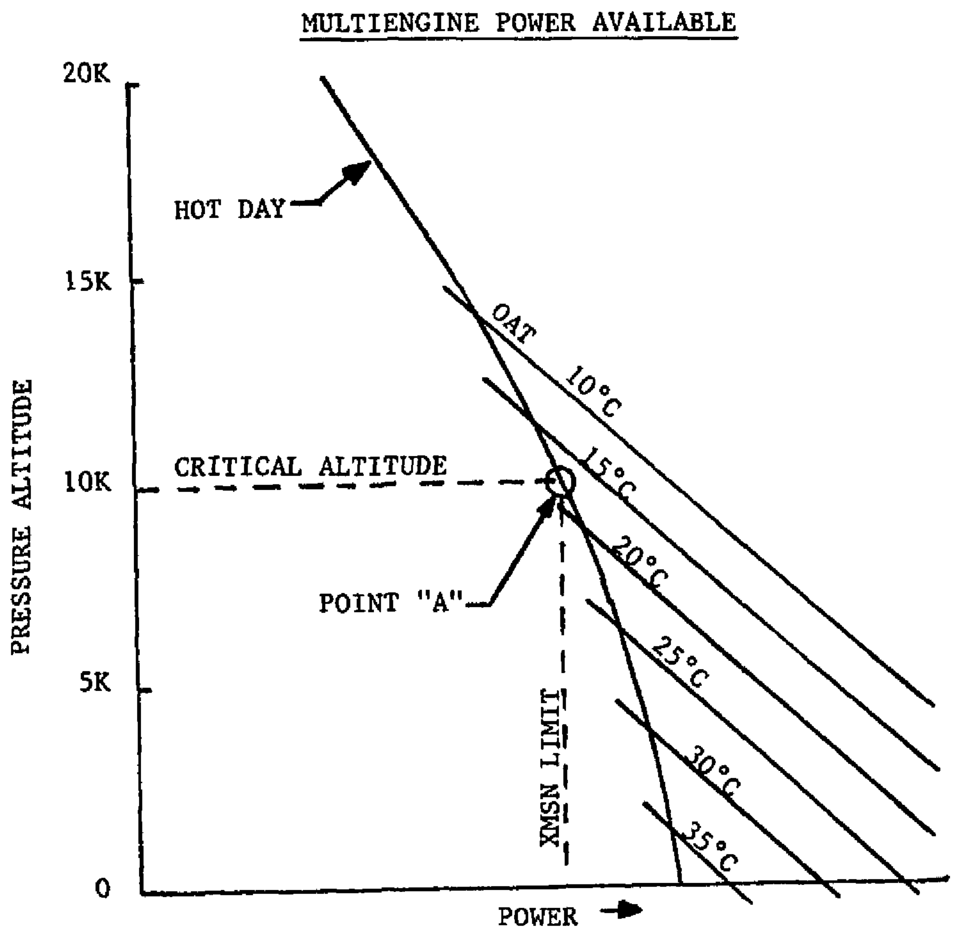

FIGURE 518-1 ALL-ENGINE-CRITICAL ALTITUDE

(3) Green signifies a safe operating condition and more specifically has come to signify landing gear extended and locked. Extensive use of green annunciators throughout the cockpit should generally be avoided due to possible confusion with the special use of green for landing gear. If green annunciators are physically and functionally removed from the landing gear operation, they may be found acceptable for a variety of "safe operating" applications. One such application is "all green for approach," used in autopilot, flight director, and other navigation system displays.

(4) Other colors may be utilized as advisory lights in accordance with § 29.1322(d). Red and amber must not be used as advisory lights due to the possibility of introducing confusion into the cockpit. Obviously, yellow and pink annunciators should be avoided due to their similarity to amber and red. White and blue have been succesfully utilized as advisory segments in past civil designs.

(5) The primary test for designation of color is:

(i) Red - Is immediate action required?

(ii) Amber - Is pilot action (other than immediate) required?

(iii) Green - Is safe operation indicated, and is the indication sufficiently distinct to prevent confusion with the landing gear down indication?

(iv) Other advisory lights - Is the meaning clear and distinct enough to prevent confusion with other annunciations? Do the colors which are utilized differ sufficiently from the colors specified in paragraphs b(1), (2), and (3) above?

(6) Annunciator lights should be visible during bright daylight conditions. This should include visibility in direct sunlight unless lights are located in such a manner that direct sunlight cannot impinge on them.

(7) If dimming capability is provided, all annunciators, including master warning and caution, may be dimmable so long as the annunciation is clearly discernable for night operation at the lower lighting level. Undimmed annunciations have been found unacceptable for night operation due to disruption of cockpit vision at the high intensity. The dimming circuit should automatically revert to the high intensity setting when power is removed. Automatic dimming/brightening through the use of a photo cell is also acceptable, as are circuits which enable a dimming switch through a position light or other cockpit lighting controls.

(8) The use of flashing lights should be minimized. If a flashing feature is used, it should be controllable through pilot action so that flashing annunciation does not persist indefinitely. The indicator should be so designed that if it is energized and the flasher device fails, the light will illuminate and burn steadily.

(9) Refer to paragraph 779 of this Advisory Circular, Annunciator Panels, for additional design information.

634. <u>§ 29.1323 AIRSPEED INDICATING SYSTEM. (RESERVED)</u>

635. <u>§ 29.1325 STATIC PRESSURE AND PRESSURE ALTIMETER SYSTEMS. (RESERVED)</u>

636. <u>§ 29.1327 MAGNETIC DIRECTION INDICATOR. (RESERVED</u>

637. <u>§ 29.1329 AUTOMATIC PILOT SYSTEM. (RESERVED)</u>

638. <u>§ 29.1331 INSTRUMENTS USING A POWER SUPPLY. (RESERVED)</u>

639. <u>§ 29.1333 DUPLICATE INSTRUMENT SYSTEMS. (RESERVED)</u>

* 640. § 29.1335 (through Amendment 29-19) FLIGHT DIRECTOR SYSTEMS.

a. Explanation. This section prescribes the accepted display criteria for a helicopter three-cue flight director providing command guidance for pitch, roll, and power. Three-cue flight directors for helicopters use the usual pitch and roll command cues with the third cue displayed on the left side of the attitude director indicator (ADI). These instruments can be used in either the two-axes or three-axes modes. In either mode, the lateral command cue controls the roll attitude and the vertical command cue controls the pitch attitude. The helicopter attitude, controlled by the cyclic control, is changed to satisfy the flight director commands. The third cue, when displayed, commands collective pitch position and is used when an airspeed or pitch attitude mode and a vertical mode (altitude hold, glide slope, etc.) are selected.

(1) The general convention for flight director design is that each command bar is a "fly to" command. The motion of the flight director indicator is such to command a corresponding sense of control system motion. This is true of flight director pitch and roll commands and should hold true for additional commands such as collective pitch.

(2) Some consideration should be given to the collective or third cue display. For example, if the collective symbol is selected as the fixed index, the command cue and collective pitch control should move in opposite directions when collective pitch changes are made. This configuration would constitute a conventional "fly to" indicator. If the collective symbol is selected for the movable index, the direction of motion of the collective symbol will coincide with the direction of collective pitch changes. In this case the moving collective symbol does not comply with the "fly to" convention; however, this configuration has been approved by the FAA with special symbology, special background effects, and special color coding, and has performed satisfactorily in service.

b. Procedures. The recommended display for a three-cue flight director incorporates the standard pitch and roll command symbols, either pitch and roll bars or the "V" bar display. The third cue, or collective symbol, should be located on the left side of the ADI. The shape of the moving cue and the background display should be unique to avoid being confused with a glide slope display or angle of attack display. One display uses a third cue, shaped like a small handle, to aid in identifying it as the collective pitch symbol.

(1) The color of the pitch and roll command indicators, the aircraft symbol, the background marking of the third cue, and third cue itself, should be consistent. The optimum color scheme uses the same color for the aircraft symbol and the collective symbol. This is usually fire orange. The command cues including the collective cue also should use the same color, usually yellow. The rationale for the different colors is that the aircraft symbol and the collective symbol (the same color) are moved toward their respective command cues. If the pitch command cue is above the center, the aircraft symbol is raised (nose pulled up) and, if the collective command cue is above the collective symbol, the collective pitch is raised, moving the collective symbol towards the command cue. *

* (2) If the attitude director indicator (ADI) provides a monochromatic display, the collective pitch cue and its background markings must be distinctive to reduce the chance of being confused with the glide slope indicator. This can be accomplished through the use of different shaped cues and background marks. A round cue with a chevron-shaped background marking has been satisfactory. *

(ii) Vibrating pencils, name plates, or permanent inks may be used. However, serial numbers should be applied on each part such that material is upset or displaced on the part, thereby attaining a more permanent number. This is not a requirement however. When material is upset or displaced the least critical or lowest stressed area should be used.

(iii) For small parts, the rule (§ 45.14) allows markings that are equivalent to part and serial numbers. Markings or symbols may be used to enable the identification of a part as one for which a replacement time, inspection interval, or related procedure is specified in the Airworthiness Limitations Section. The FAA stated identification of such small parts is clearly essential for safety and may not be relieved. With adoption of Amendment 29-20, the marking requirements are contained in § 45.14, Amendment 45-12.

(3) The FAA inspector should witness the rigging of the controls of a test rotorcraft. This is imperative for a new helicopter design to assure the practicality and feasibility of the procedures stated in the design data and/or the maintenance manual. The type design data information should be used; then the FAA should assure the manual includes the proper information. Rigging procedures are not included in the airworthiness limitations section.

(4) A draft copy of the manual should be available to the FAA for use during the F&R program if such a program is conducted under § 21.35(b). The manual must be completed and furnished with each aircraft receiving an airworthiness certificate.

730.-739. RESERVED.

SECTION 41. MARKINGS AND PLACARDS

740. § 29.1541 GENERAL. (SEE PARAGRAPH 781)

741. § 29.1543 INSTRUMENT MARKINGS: GENERAL. (SEE PARAGRAPH 781)

742. § 29.1545 AIRSPEED INDICATOR. (SEE PARAGRAPH 781)

* 743. § 29.1547 (through Amendment 29-19) MAGNETIC DIRECTION INDICATOR.

a. Explanation. This regulation identifies the requirement for a calibration placard for the magnetic direction indicator and where it should be located.

b. Procedures. One means of accomplishing the requirements of this regulation is commonly known as swinging the compass. A surveyed compass rose is laid out on an appropriate surface. The compass rose location should be free from the influence of steel structures, underground pipes and cables, reinforced concrete, and other aircraft. The aircraft should be in an attitude which permits an accurate result. Normally the engines are in operation; however, if the helicopter is equipped with an auxiliary power unit which can supply all required electrical power, this can be used in lieu of engine driven generators. Turn the aircraft on successive headings through 360°. It is recommended that the increments be every 30°; however, the increments should not exceed 45°. Prepare a placard to show the correction to be applied at each of the selected headings. When significant errors are introduced by operation of electrical/electronics equipment or systems, the placard should also be marked at each calibration heading showing the correction to be applied when such equipment or systems are turned on or energized. The placard resulting from this calibration should be installed on or near the magnetic direction indicator. *

744. § 29.1549 POWERPLANT INSTRUMENTS. (SEE PARAGRAPH 781)

745. § 29.1551 OIL QUANTITY INDICATOR. (RESERVED)

746. § 29.1553 FUEL QUANTITY INDICATOR. (RESERVED)

747. § 29.1555 CONTROL MARKINGS. (RESERVED)

748. § 29.1557 MISCELLANEOUS MARKINGS AND PLACARDS. (RESERVED)

749. § 29.1559 LIMITATIONS PLACARD. (RESERVED)

750. § 29.1561 SAFETY EQUIPMENT. (RESERVED)

751. § 29.1565 TAIL ROTOR. (RESERVED)

752.-761. RESERVED.

SECTION 42. ROTORCRAFT FLIGHT MANUAL

* 762. § 29.1581 (through Amendment 29-19) GENERAL.

a. Explanation.

(1) The primary purpose of the Rotorcraft Flight Manual (RFM) is to provide an authoritative source of information considered to be necessary for or likely to promote safe operation of the rotorcraft.

(2) Since the flightcrew is most directly concerned with operation of the rotorcraft, the language and presentation of the flight manual shall be directed principally to the needs and convenience of the flightcrew, but should not ignore the needs of other contributors to safe operation. As used with respect to the RFM, safe operation is construed to include, but not be limited to, operation of the rotorcraft in the manner that is mandatory for, or recommended for, compliance with applicable airworthiness requirements, and with the particular provisions of the operating regulations relating to the rotorcraft's approved performance capabilities.

(3) To serve its intended purpose, therefore, the RFM must include the certificate limitations established for the design as a consequence to the type certification evaluation, the performance information necessary to establish the operating limitations imposed in accordance with appropriate operating regulations, and the procedures and other information necessary to enable the flightcrew to safely operate the rotorcraft within the envelope of limitations thus delineated. The outline presented in this circular is directed toward those objectives.

(4) Information and data that are mandatory for an acceptable RFM are prescribed in §§ 29.1581 through 29.1589, and nothing contained in these sections should be construed as amending those requirements. Certain additional elements of flight manuals, however, have been shown by experience to be practical necessities if the document is to serve effectively its intended purpose.

b. Procedures.

(1) The following criteria do not affect the status of RFMs which are presently approved. When such manuals are amended in the future, however, it is recommended that the concepts of this section be incorporated wherever uniformity or clarity will result.

(2) Only the material required by FAR Part 29, or that considered necessary to implement the operating regulation, should be included in the portion of the manual that is approved by the FAA. However, the manufacturer or operator may include other "unapproved" data in a separate and distinctively identified portion within the same document. *

* The RFM is considered necessary for safe operation of the rotorcraft and care should be taken to produce a manual that is consistent with the need for completeness and clarity of the required information. Also, since the RFM is necessary for operation of the rotorcraft in accordance with the certificate limitations, it is considered to be public information.

(3) The page size for the RFM will be left to the discretion of the manufacturer. In this regard, operational compliance with § 91.31 should be considered. A cover should be provided and should indicate the nature of the contents by means of the title, "RFM." Each page of the approved portion should bear the notation "FAA approved," an induction of the approval sequence of that particular page (e.g., a date of approval, a revision number suitably supported by an amendment log which contains the appropriate date, etc.), the helicopter model number as it appears on the type data sheet, and any appropriate document identification number. Pages of the unapproved portion of the flight manual would use the issue date in lieu of the FAA approved date. The material should be bound in semipermanent fashion so that the pages will be protected and retained in proper sequence. In selecting the form of binding, consideration should be given to the necessity for amendment and the ease with which amendments can be accomplished.

(4) Amendments may take the form of revisions or supplements.

(i) A revision is a change to the RFM or its supplement made by the holder of the applicable type certificate (TC) or in the case of supplement prepared as a part of a supplemental type certificate (STC), by the holder of the STC.

(ii) A supplement is an addition to the RFM. If the helicopter manufacturer (holder of the TC) adds optional equipment or specific operations (such as Category "A" vertical operation or IFR operations), then the helicopter manufacturer is responsible for preparing any necessary flight manual material whether he elects it to be a supplement or a revision to the basic manual. If someone other than the helicopter manufacturer applies for an STC to install equipment or modify the helicopter such that a RFM supplement is necessary, then the person who applies for the STC is responsible for the preparation of the RFM supplement.

(5) "Revision" may be incorporated by inserting new pages which embody the amended text and, where applicable, by removing superseded pages. A vertical amendment bar should be inserted in the outer margin, where practicable, to indicate those parts of the text that have been changed. Each amended page should be identified in the same manner as pages of the basic manual, and in addition should carry an identification of its approval sequence.

(6) Supplements are incorporated in the manual by inserting the applicable pages which contain the information associated with the particular change. Each supplemental page should also identify the helicopter type and model flight manual for which the supplement was issued, the name of the issuer, and the FAA approval date. The following statement is an example of a note which*

* would be included on the title page of a flight manual supplement: "For helicopters approved to operate in accordance with the provisions of the helicopter flight manual supplement, the information contained herein supplements the information of the basic flight manual. For limitations, procedures, and performance data not contained in this supplement, consult the basic flight manual."

(7) Supplements should contain as much of the flight manual contents outlined below as considered appropriate for the particular change in type design, including title page and index of contents. It is suggested that these be prepared with a view to insertion in the FAA-approved portion of the flight manual as a complete and self-contained unit.

(8) The RFM should contain as much of the information required in Part 29 as is applicable to the individual type and model. For the purpose of standardization, it is recommended that the sequence of sections and of items within sections, follow the format presented at the end of this paragraph if practicable.

(9) The following information would normally be included in the introduction section of the flight manual.

(i) Title Page. This page should include the manufacturer's name and address and the helicopter model number as it appears on the type certificate data sheet. If desired, include a trade name or trade model number in quotes, provisions for rotorcraft serial number and registration number, approval date of the basic document, and title and signature of the FAA approving official.

(ii) Table of Contents. An index should be located at the front of each section or at the front part of the manual.

(iii) Amendment Log. This log should be in the form of a table with provisions to record, for each amendment, an identifying number, title or description, the page numbers involved, the issue date, the identification of the FAA approving official, and the FAA approval date.

(iv) Separate amendment logs should be provided for each type of amendment issued; i.e., Log of Revisions, Log of Supplements, etc. Amendments issued by other than the holder of the basic type certificate should include a separate amendment log which, in addition to the issue date, should also identify the issuer and the STC number or other approval basis for the associated modification.

(v) List of Current Pages. This table should list, for each approved page of the manual, the issue date and any other appropriate identification necessary to establish that the manual is complete and current.

(10) The following flight manual format would be acceptable. The format recommends a sequence of sections and suggests items which would be included in those sections. *

FLIGHT MANUAL FORMAT

* INTRODUCTION

PART I, FAA APPROVED

Section 1 Limitations

Section 2 Normal Procedures

Section 3 Emergency Malfunction Procedures

Section 4 Performance Data

Section 5 Optional Equipment Supplements

PART II, MANUFACTURER'S DATA

Section 6 Weight and Balance

Section 7 Systems Description

Section 8 Handling, Servicing, and Maintenance

Section 9 Supplemental Performance Information

INTRODUCTION: This section would include any signature pages, list of approved pages, the log of revisions, and any additional introductory information desired. For each section, it is suggested that the following major titles be utilized and that the recommended information listed under each title be incorporated. Each section should include a table of contents and a list of figures applicable to that particular section.

Section 1 - Limitations:

a. Kinds of Operation.

Under this heading, crew requirements, VFR and/or IFR flight authorizations, and any operational restrictions would be presented.

b. Flight Limitations.

This section would include limitations with respect to airspeed, altitude, ambient temperatures, wind, slope, prohibited maneuvers, and any other flight limitations associated with a particular helicopter (i.e., HV limitations for Part 29 Category A helicopters). *

* c. Weight Limitations.

This section would contain all gross weight, center of gravity (both longitudinal and lateral) limitations, and any other weight limitations unique to the helicopter (i.e., crew, passenger and/or cargo loadings, WAT limitations for Part 29 helicopters, etc.)

d. Powerplant Limitations.

This section would include the temperature and pressure limits associated with powerplant operation; i.e., torque, r.p.m., TOT, etc. This section would also include approved fuels and oils and their temperature and pressure limits. Any accessories attached to the powerplant (i.e., starters, generators, etc.), to which limitations in starting or operation are applicable, would be included herein.

e. Rotor Limitations.

This would include the power-on and power-off r.p.m. limits, the effect of altitude on these parameters, and any other limitations associated with the rotor system(s).

f. Drive System Limitations.

This section would include all limitations associated with the drive system (i.e., main transmission, any adapter gear boxes, tail rotor gearbox, and any other drive system component applicable to a particular helicopter).

g. System Limitations.

This section would include any particular system limitations unique to the helicopter (i.e., battery limitations, hydraulic system limitations, and any limitations associated with the various types of stability augmentation and/or automatic flight control systems).

h. Instrument Markings.

All instrument markings would appear in this section. The significance of each limitation and of the color coding would be explained in this paragraph.

i. Placards.

The exact wording and general location of all placards pertaining to flightcrew function or cargo loading would appear in this section. *

* Section 2 - Normal Procedures:

a. Preflight Checks.

This paragraph would include any exterior, interior, and any system checks prior to starting the engine(s).

b. Engine Start.

This paragraph would include any procedures associated with the engine start(s).

c. System Checks.

This paragraph would include any system check procedures such as hydraulic, stability augmentation, electrical, flight control, etc., which should be accomplished prior to takeoff.

d. Takeoff.

This paragraph would include any procedures associated with the takeoff and any procedures unique or applicable to the takeoff profile.

e. Cruise and/or Level Flight.

This paragraph would include any procedures applicable to cruise and/or level flight operation.

f. Approach and Landing.

This paragraph would include any procedures required or recommended for the approach and landing duration of the helicopter operation.

g. Engine/Rotor Shutdown.

This paragraph would include any procedures applicable to the engine and/or rotor shutdown and any procedures applicable upon completion of the helicopter operation.

h. Miscellaneous Procedures.

This section would include procedures for miscellaneous systems or conditions, such as bleed air heater, anti-ice systems, cold weather operations, etc.

Section 3 - Emergency and Malfunction Procedures:

a. Introduction.

This paragraph would include any introductory type information (i.e., definitions of terms used and any other information the manufacturer deemed appropriate). *

* b. Powerplant Failures.

This paragraph would include any information relative to engine, fuel control, or any other powerplant related emergency or malfunction.

c. Drive System Failures.

This paragraph would include recommendations and procedures relative to any drive system failure and/or malfunction.

d. System Failures.

This paragraph would include procedures and recommendations relative to any system failure and/or malfunction (i.e., electrical, hydraulic, and augmented flight control systems).

e. Fire.

This paragraph would include procedures to be followed in the event that engine, cabin, baggage compartment fire or smoke is detected.

f. Emergency Egress.

This paragraph would include emergency evacuation procedures for both the flightcrew and the passengers.

Section 4 - Performance Data:

a. Power Assurance.

This section would include all information relative to the power assurance checks.

b. Hover Information.

This paragraph would include all information relative to hover performance (i.e., hover ceiling IGE and OGE for single and/or multiengine operation). Any relative wind effects would also be included.

c. Takeoff and Landing and Climb Performance.

This paragraph would include information relative to the takeoff and landing profiles (i.e., HV curves, normal climbs, autorotation speeds, takeoff and landing distance over 50-foot obstacles, and any other data applicable to the particular helicopter).

d. Airspeed Calibration.

This paragraph would include the airspeed calibrations required for the particular helicopter. *

* Section 5 - Optional Equipment Supplements:

This section would include all optional equipment supplements. These supplements may modify any of the limitations, procedures (both normal and emergency), and performance characteristics of the basic helicopter.

PART II, Manufacturer's Data (Not FAA Approved)

Section 6 - Weight and Balance:

All supplemental weight and balance information such as crew tables, passenger tables, fuel and oil tables, cargo tables, and any other loading tables applicable to the particular helicopter would appear in this section.

Section 7 - Systems Description:

This section would include all information relative to the various helicopter systems that the manufacturer believes would apply to the particular helicopter.

Section 8 - Handling, Servicing, and Maintenance:

This section would include all information relative to the handling, servicing, and maintenance that the manufacturer would care to present. This section would also include dimensions (i.e., baggage areas, doors, and any internal, external information appropriate to the helicopter).

Section 9 - Supplemental Performance Information:

This section would include any supplemental performance information the manufacturer would wish to provide. This section would also contain the cruise-range information associated with IFR operation. *

* 763. § 29.1583 (through Amendment 29-19) OPERATING LIMITATIONS.

a. Explanation. The purpose of this section is to present the limitations applicable to the rotorcraft type and model as established in the course of the type certification process. The limitations should be presented without explanations other than those explanations prescribed in Part 29. To the maximum practicable extent, the limitations should be presented in "operations" language and format. Since operation of the rotorcraft in accordance with such limitations is required by the operating regulations, the following should be inserted as a note at the beginning of this section: "Operation in compliance with the limitations presented in this section is required by the Federal Aviation Regulations." Section 29.1583 merely states that certain information must be given. The specific information is found during the showing of compliance with other paragraphs in the regulation.

b. Procedures.

(1) Section 29.1545 gives the markings required for the airspeed indicator.

(2) Rotor limits are established during compliance with § 29.33. The markings are specified in § 29.1549.

(3) Powerplant limits are discussed under §§ 29.1549 through 29.1553.

(4) Weight limitations are specified in § 29.25. In the operating limitations section, there should be a statement of the maximum and minimum certificated takeoff and landing weights. For those weight limitations that vary with altitude, temperature, or other variables, the variation in weights may be given in the form of graphs in the performance section of the manual and included as a limitation by specific reference in the limitation section to the appropriate graph or page.

(5) Center of gravity limits are determined in accordance wth § 29.27 and may be presented in the same manner as prescribed for the weight limitations (i.e., a statement under "center of gravity limits" in the limitations section which references graphs or page numbers in the performance section). If landing gear position can measurably effect allowable c.g., this information should be presented together with the moment change due to gear retraction.

(6) The minimum flightcrew is determined under § 29.1523 and is dependent upon the kinds of operation authorized. The established number and identity, by crew position of the minimum flightcrew, must be listed.

(7) Kinds of operations are established under § 29.1525. This section should contain the following preamble: "This rotorcraft is certified in the Transport Category (A and/or B) and is eligible for the following kinds of operation when the appropriate instruments and equipment required by the *

* airworthiness and/or operating rules are installed and approved and are in operable condition." Those of the following, and any others that are applicable, should be listed.

(i) Day and night VFR.

(ii) Approved to operate in known icing conditions.

(iii) IFR.

(iv) Category A vertical operations from ground level or elevated heliports.

(v) Extended overwater operations (ditching).

(vi) External load operations.

(8) Limiting heights and speeds are determined under § 29.1517 and are presented in the form of a height versus velocity diagram in the limitations section for Category A rotorcraft and in the performance section for Category B rotorcraft.

(9) Unusable fuel tests are required by § 29.959. When the amount of unusable fuel has been determined, the manufacturer calibrates his fuel quantity system so that when the fuel quantity in the tank is down to the unusable quantity, his fuel gage will read "zero." A statement should appear in the limitations section to warn the pilot that the quantity of fuel remaining in the tanks when the gage reads zero is not usable in flight.

(10) Often other limitations are included in the limitations section that are not specifically mentioned in the rules but which are necessary for safe operation. Examples are:

(i) Altitude limits.

(ii) Ambient temperature limits.

(iii) Conditions for use of rotor brake.

(iv) Prohibitions against prolonged hover in cross or tail winds to prevent accumulation of noxious fumes in cockpit or cabin.

(v) Prohibitions against acrobatic maneuvers.

(vi) Required placards including text and location.

(vii) Special airworthiness equipment installations such as engine out or low rotor r.p.m. warning systems. *

* 764. § 29.1585 (through Amendment 29-19) OPERATING PROCEDURES.

a. Explanation. The procedures sections of the manual should contain essential information peculiar to the particular type or model, the knowledge of which may be expected to enhance safety in the kinds of operations for which the type or model is approved. Information or procedures not directly related to airworthiness, or not under control of the crew, should not be included, nor should any procedure which is accepted as basic airmanship.

(1) Procedures information should be presented with respect to normal and emergency procedures. Alternatively, information outside the category of normal procedures may be subdivided into categories described as "abnormal" procedures and "emergency" procedures, as described herein.

(2) Notes, cautions, and warnings may be used to emphasize specific instructions or information in general accord with the following.

(i) "Note" should be used with respect to matters not directly related to safety but which are particularly important (e.g., Note: For normal twin-engine operation, maximum permissible torque needle split is 4 percent total).

(ii) "Caution" should be used with respect to safety matters of a secondary order not immediately imminent (e.g., Caution: On engine restart reduce ITT to 750° C on the operating engine).

(iii) "Warning" should be used with respect to safety matters of a primary order or imminent (e.g., Warning: Do not allow rotor r.p.m. to drop below minimum limits).

(3) The operating procedures of this section have been developed with specific regard for the design features and operating characteristics of the rotorcraft and have been approved by FAA for guidance in identifying acceptable procedures for safe operation. Observance of these procedures is not mandatory, and FAA approval of such procedures is not intended to prohibit or discourage development and use of improved or equivalent alternate procedures based on operational experience with the rotorcraft. When alternate procedures are used, full responsibility for compliance with applicable airworthiness safety standards rests with the operator.

b. Procedures. Procedural information should be presented in substantial accord with the categories described below:

(1) Normal Procedures. Normal procedures are concerned with peculiarities of the rotorcraft design and operating features encountered in connection with routine operations, including malfunction cases not considered in the other procedures section (i.e., not considered to degrade safety). Material conforming to the above should be presented for each phase of flight, following in sequence from preflight through engine shutdown, and should include, but not be limited to, systems operation (including fuel system information prescribed in 29.1585(b)), missed approaches, etc. *

* (2) Abnormal Procedures (Optional). Abnormal procedures are concerned with foreseeable situations, usually entailing a failure condition, in which the use of special systems, and/or the alternate use of regular systems, may be expected to maintain an acceptable level of airworthiness. Typical examples of events considered to entail abnormal procedures are minor engine malfunctions and associated conditions for safe flight, stopping and restarting engines in flight, extending landing gear or flaps by alternate means, approach with inoperative engine(s), etc.

(3) Emergency Procedures. Emergency procedures are concerned with foreseeable but unusual situations in which immediate and precise action by the crew, as detailed in the recommended procedures, may be expected to reduce substantially the risk of disaster. Typical examples of incidents considered to be emergencies are fire, ditching, loss of tail rotor thrust, etc.

(4) Ditching Procedures. Amendment 29-12 added ditching standards to Part 29. When ditching approval is requested, appropriate procedures and information will be included in the manual. Scale model tests are generally used to prove autorotation "ditching" characteristics and to prove stability in the water (capsize threshold) of the helicopter type design. Many helicopter designs require emergency float bags that deploy either before water contact or shortly after water contact to provide the flotation and stability necessary to comply with the requirements.

(i) Autorotation altitudes and airspeeds and water contact information, if appropriate, derived from or used during the ditching model tests, should be confirmed during FAA flight tests and should be included in the manual. Information concerning sea states or wave heights to length ratios, investigated and found satisfactory, may be included in the manual if nonsevere sea states are likely to be exceeded.

(ii) Instructions for deploying liferafts may be needed for certain designs. For example, if liferafts are stowed outside the cabin, special instructions may be necessary.

(5) Evacuation Procedures for Helicopter Litter Configurations. Appropriate procedures and minimum crew requirements should be considered and included in the manual or manual supplement, if necessary, to assure timely evacuation.

(6) The use of illustrations to show controls, instruments, explain systems, etc., is encouraged. *

* 765. § 29.1587 (through Amendment 29-19) PERFORMANCE INFORMATION.

a. Explanation.

(1) This section would contain the performance information necessary for operation in compliance with applicable performance requirements of FAR Part 29 and applicable special conditions, together with additional information and data essential for implementing pertinent operational requirements.

(2) Performance information and data may be presented for the range of weight, altitude, temperature, and other operational variables stated as operational performance limitations. It is recommended that performance information and data be presented substantially in accordance with the following paragraphs. Where applicable, reference to the appropriate requirement of the certification or operating regulation should be included.

(i) General. Include all descriptive information necessary to identify the configuration and conditions for which the performance data are applicable. Such information may include the complete model designations of rotorcraft and engines, definition of installed rotorcraft features, and equipment that affects performance together with the operative status thereof. This section should also include definitions or terms used in the performance section (i.e., IAS, CAS, ISA, configuration, CDP, V_{TOSS}, Category A, Category B, LDP, etc.) plus calibration data for airspeed, ambient air temperature, and other information of a general nature.

(ii) Performance Procedures. The procedures, techniques, and other conditions associated with obtainment of the flight manual performance should be included, the procedures may be presented as a performance subsection or in connection with a particular performance graph. In the latter case, a comprehensive listing of the conditions associated with the particular performance may serve the objective of "procedures" if sufficiently complete. Performance figures are based on the installed minimum specification engine unless normally depreciated engine performance is approved.

(iii) Wind Accountability. Wind accountability may be utilized for conventional takeoff field lengths. This accountability shall not be more than 50 percent of the nominal wind component along the takeoff path opposite to the direction of takeoff. In some rotorcraft, it may be necessary to discount the beneficial aid to takeoff performance for winds from 0 to 10 knots. This should be done if it is evident that the winds from 0 to 10 knots have resulted in a significant degradation to the takeoff performance due to the washout of the ground effect cushion.

(iv) The following list is illustrative of the information that may be provided for a transport Category "A" and "B" helicopter.

(A) Density altitude chart for converting from pressure to density altitude.

(B) Temperature conversion chart (°C to °F to °C). *

* (C) Airspeed calibration (calibrated vs. indicated airspeed) for both pilot and copilot systems for level flight, climb, autorotation, and recommended approach rate of descent.

(D) Altimeter correction for pilot and copilot instruments showing the correction factor vs. indicated airspeed at sea level and altitude.

(E) Hover performance charts both in and out-of-ground effect with instructions for their use. The out-of-ground effect hover performance chart is not required but may be useful.

(F) A series of climb performance charts for various weights showing rate of climb vs. pressure altitude for a range of temperatures and showing the variation of best rate of climb speed with pressure altitude. The conditions should appear on each chart (i.e., power, weight, single, or multiengine, etc.).

(G) A chart showing the takeoff flight path for Category A presented in height vs. distance from the hover wheel height to the point at which V_{TOSS} and not less than 35 feet is reached, and the rejected takeoff distance. The chart should identify the critical decision point and V_{TOSS}.

(H) A chart or series of charts to allow calculation of any additional distance which may be required to accelerate to best rate of climb speed from V_{TOSS}.

(I) Landing distance chart for Category A showing the landing distance from a 50-foot height (25-foot for VTOL operations from an elevated heliport) to a stop with one engine inoperative vs. pressure altitude over the range of temperatures being certified. This chart should identify the balked landing decision point (LDP) so the pilot will know how to achieve this performance.

(J) For Category B, a chart or series of charts at various weights showing takeoff distance from hover to 50 feet vs. pressure altitude over the range of temperatures being certified.

(K) For Category B, a landing distance chart similiar to the one for Category A from a 50-foot height to stop with one engine inoperative.

(L) For turbine powered helicopters in all categories, a power assurance check chart.

(M) A statement of the maximum crosswind and downwind components that have been demonstrated as safe for operation near the ground.

(v) Miscellaneous Performance Data. Any performance information or data not covered in items (A) through (L) above, but considered necessary or for use with the operating procedures contained in the rotorcraft flight manual, should be included. *

* (vi) Flightcrew Notes. It is recommended that provisions be made in the manufacturers portion of the Rotorcraft Flight Manual for inclusion of information and data of a type that is useful or desirable for operation of the rotorcraft but is not approved by FAA. (Material in this section should be consistent with material in the approved portion of the manual.)

766. § 29.1589 (through Amendment 29-19) LOADING INFORMATION.

a. Explanation. Control of the rotorcraft weight and balance is an operational function, and is the responsibility of the operator. However, instructions necessary to enable loading of the rotorcraft within the established limits of weight and center of gravity, and to maintain the loading within such limits are required by the operating regulations, and inclusion of such loading instructions in the Rotorcraft Flight Manual is required by § 29.1583(c). Approved loading instructions, therefore, must be presented in the Rotorcraft Flight Manual, and at the option of the applicant, may be included in the approved portion or may be included in the unapproved portion.

b. Procedures.

(1) For the purpose of the flight manual, distinction is made here between the loading instructions required by the certification requirements of Part 29, and the weight and balance data required by the operating requirements. The former prescribed information is applicable to the rotorcraft type, and is subject to FAA approval as flight manual material.

(2) For compliance with the noted requirements, it is necessary for the applicant to develop weight and balance data and loading instructions as necessary to satisfy the needs of both certification and operation. In order to consolidate in one document information on rotorcraft loading, however, it is recommended that the weight and balance data be developed to include appropriate loading instructions, and that both be included in the Rotorcraft Flight Manual as an "unapproved" section entitled, "Weight and Balance." Such a section should include the following statement as a note: "In accordance with FAA procedures, the detail weight and balance data of this section are not subject to FAA approval. The loading instructions of this section, however, have been approved by FAA as satisfying all requirements for instructions on loading of the rotorcraft within approved limits of weight and center of gravity, and on maintaining the loading within such limits."

(3) An actual or specimen weight and balance section should be included in the initial submittal of the manual. Weight and balance data for each particular rotorcraft need not be submitted as flight manual material.

(4) The weight and balance material outlined below is believed to be adequate for rotorcraft with conventional loading and fuel-management techniques. For rotorcraft which necessitate redistribution of fuel (other than normal consumption) to maintain loading within prescribed limits, the material should be amplified as necessary.

(i) Weight Limits. Contained in limitations section of the flight manual. *

* (ii) Center of Gravity Limits. Conatined in the limitations section of the flight manual.

(iii) Dimensions and Datum Line Locations. The dimensions and relative location of rotorcraft features associated with weighing and loading of the rotorcraft and with weight and balance computations should be described and/or illustrated.

(iv) Equipment List. The rotorcraft should be defined or described sufficiently to identify the presence or absence of optional systems, features, or installations that are not readily apparent. In addition, all other items of fixed and removable equipment included in the empty weight should be listed.

(v) Fuel and Other Liquids. Fuel and other liquids, including passenger-service liquids that are included in the empty weight, should be identified and listed together with information necessary to enable ready duplication of the particular condition.

(vi) Weight Computations. Computations of the empty weight and empty-weight c.g., location should be included.

(vii) Empty Weight and Empty-Weight Center of Gravity Location. Statement of these values should be included.

(viii) Loading Schedule. Loading schedule should be included, if appropriate.

(ix) Loading Instructions. Complete instructions relative to the loading procedure, or to use the loading schedule, must be included.

(x) Special Consideration. Consideration should be given to the lateral center-of-gravity loading instructions when various kits such as a side mounted hoist are installed. *

767.-774. RESERVED.

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Jerry M. Kelly
for

James D. Erickson
Manager, Rotorcraft Directorate
Aircraft Certification Service

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(8) This test is the least predictable of all the performance items. Therefore, the expansion and extrapolation of test data are questionable. Weight may not be extrapolated to higher values. In order to extrapolate HV data to higher altitudes, any analytical method must have FAA approval. In lieu of pure analytical methods, simulations have been used successfully, especially for multiengine helicopters. In either case, the maximum allowable extrapolation should be limited to 2,000 feet density altitude (Hd). HV test weights should be consistent with the takeoff and landing WAT (weight, altitude, temperature) limit curve which will be placed in the Rotorcraft Flight Manual. For a given diagram, typical weight reductions that are necessary as altitude is increased can be conservatively estimated by maintaining a constant gross weight divided by density ratio, GW/σ. See Figure 72-2, Part A. If weight is not varied, an enlarged HV diagram is required for safe power-off landing as density altitude is increased. See Figure 72-2, Part B. Another method of presentation is to show varying weights at a constant density altitude. See Figure 72-2, Part C.

(9) Vertical takeoff and landing (VTOL) testing normally does not require separate HV testing. The takeoff and landing tests take on the combined characteristics of takeoff, landing, and HV tests.

(10) Rotorcraft certificated prior to Amendment 29-21 were required to have the resulting height-velocity diagram as an operating limitation. This limitation restricted opportunities when operating large helicopters in various utility applications. Subsequently, Amendment 29-21 allows, under certain conditions, the height-velocity diagram to be placed in the Flight Manual Performance Information Section instead of the Limitations Section. Specifically, the rotorcraft must be: (1) certificated for a maximum gross weight of 20,000 pounds or less; (2) configured with nine passenger seats or less; and (3) certificated in Category B. Testing must be completed with the aircraft at the maximum gross weight at sea level. For altitudes above sea level, the test aircraft must be at a weight no less than the highest weight the rotorcraft can hover out-of-ground-effect (OGE). Rotorcraft certificated prior to Amendment 29-21 can update their certification basis to take advantage of this provision. *

b. Procedures.

(1) Instrumentation.

(i) Ground Station. The ground station must have equipment and instrumentation to determine wind direction and velocity, outside air temperature and (if the test helicopter has reciprocating engines), humidity. Since the tests must be conducted in winds of 2 knots or less, a smoke generator is highly recommended to show both flight crew and ground crew personnel the wind direction and velocity at any given time. Additionally, the location of the ground station should be such that it is free of rotor downwash at all times. Motion picture, phototheodolite, and radio equipment will be necessary to properly conduct the test program. The use of telemetry equipment is desirable if the location of the test site and the magnitude of the test program make it practical.

(ii) Airborne Equipment (Test Helicopter). Necessary installed test equipment may include photopanels and/or recorders for recording engine parameters, control positions, landing gear loads, landing gear deflections, airspeed, altitude, and other variables. An external light attached to the helicopter (or any other means of identifying the engine failure point to the ground camera or phototheodolite) is needed to identify the exact time of engine failure and may also be used to synchronize the ground recorder with the airborne recorded data.

(2) Analytical Prediction. The HV diagram can be estimated by analytical means and this is recommended prior to test. HV, however, is the least predictable of all helicopter performance and because of this, the expansion and extrapolation of test data must be done with great care. Test weight may not be extrapolated. All test points should be approached conservatively with some speed or altitude margin. If the manufacturer has conducted a comprehensive HV flight test program to validate his analytical predictions, much preliminary testing can be eliminated. In any case, the maximum allowable extrapolation from flight test conditions is 2,000 feet density altitude and an approved analytical and/or simulation method must be utilized for extrapolation.

(3) Power.

(i) The appropriate power level before engine failure for the low and high hover points is simply the power required to hover at the prevailing hover conditions. The appropriate power condition prior to failure of the engine for points below the knee is takeoff power or a lower value if approved as an operating limit. For cruise or "fly-in" points above the knee, the appropriate condition is power required for level flight. Rotor speed at execution of the engine failure should be the minimum speed appropriate to the flight condition.

(ii) The applicable power failure conditions are listed in § 29.79(b). Power should be completely cut for normal Category B helicopters. For Category A helicopters, the desired topping power (for the remaining engine(s)) should be set prior to the test. This power value will need adjustment as ambient conditions change. The power can be takeoff power (TOP), 2 1/2-minute power, or some calculated lower power for simulating hot day or higher density altitude conditions. Power is verified and recorded by the pilot by "topping" the engine(s) prior to engine failure tests. Care must be taken to assure that this power value is no more than that which would be delivered by a minimum specification engine under the ambient conditions to be approved.

(4) Test Loadings. Weight extrapolation is not permitted for HV. Therefore, the test weight must be closely controlled. Ballast or fuel should be added frequently to maintain the weight within -1 to +5 percent when testing final points. Ordinarily tests are conducted at a mid center of gravity unless a particular loading is expected to be particularly critical.

(5) Landing Gear Loads.

(i) Instrumented landing gear can be a great help in evaluating test results. This information can be telemetered to a ground station or otherwise recorded and displayed for direct reference following each landing.

(ii) Any landing which results in permanent deformation of aircraft structure or landing gear beyond allowable maintenance limits is considered an unsatisfactory test point.

(6) Piloting Considerations. In verifying the HV diagram, the minimum required instrument panel display and minimum crew should be used in order not to mislead the operational pilot who has no test equipment available and may have no copilot to assist. Three distinctly different flight profiles are utilized in developing the diagram.

(i) High Hover. A stabilized OGE (out of ground effect) hover condition prior to power failure is essential. A minimum 1-second time delay between power failure and initial control actuation is utilized. Following the time delay, the primary concern is to quickly lower collective and to gain sufficient airspeed to allow an effective flare approaching touchdown. While the immediate development of airspeed is necessary, the dive angle must be reasonable and must be representative of that expected in service. While initial aircraft attitude will vary between models and with changing conditions, 10-20° has been previously applied as a maximum allowable nose down pitch attitude. Use of greater attitudes could result in a diagram which is difficult to achieve and unrealistic for operations in service. Initial testing should start relatively high with gradual lowering of height to the final high hover altitude. A stabilized OGE hover condition prior to power failure is essential. If a stabilized high hover condition cannot be achieved prior to the engine cut, then this point should be tested from a minimum level flight speed. This will result in an open-ended HV diagram. A smoke source or balloon on a long cord is highly desirable since the wind can vary significantly from surface observations to typical high hover altitudes. Vertical speed must be very near zero at the throttle chop. Any climb or sink rate can have a significant influence on the success of the test point. Use of a radar altimeter with a cross check to barometric altitude is essential.

(ii) Low Hover. From the low hover position there is no flare capability and little time for collective reaction. No time delay is applied other than normal pilot reaction. For typical designs the collective may not be lowered after power failure. Lowering of the collective is not permitted because it is not a pilot action which could be expected if an engine failed without notice during a hovering condition in service. Initial lowering of collective immediately after power failure can result in very high, unconservative low hover altitudes that are unrealistic for operational conditions. If, however, a design is such that a 1-second pilot delay after power failure could be achieved without any appreciable descent, a slight lowering of collective could be allowed.

(iii) Takeoff Corridor. Normal pilot reaction is applied when the engine is made inoperative. At low speeds collective may be lowered quickly to retain rpm and minimize the time between power failure and ground contact. If airspeed is sufficient for an effective flare, the aircraft is flared to reduce airspeed, retain rotor rpm, and control vertical speed prior to touchdown. Considerable surface area may be needed for a sliding or rolling stop.

(iv) Additional Considerations. The "in-between" points utilize similar techniques. The cruise or "fly-in" points are similar to the high hover point although the steep initial pitch attitudes are not needed as altitude is decreased and airspeed is increased along the curve. The low speed points along the takeoff corridor are similar to the low hover point except that the collective may be quickly lowered and some flare capability may be used as the "knee" is approached. The pilot should be proficient in all normal autorotation landings before conducting height-velocity tests in a single engine helicopter.

(7) Ground Support. Motion picture or theodolite coverage and ground safety equipment are necessary. Communication capability among these elements should be provided. Use of a phototheodolite to compare height/speed with cockpit observations is very desirable.

(8) Verifying the HV Diagram.

(i) A sufficient number of test points must be flown to verify the diagram. The key areas are the knee, high altitude hover, low altitude hover, and high speed touchdown. Test points with excessive gear loads, above average skill requirements, winds above permissible levels, rotor droop below approved minimum transient rpm, damage to the helicopter, excessive power, incorrect time delay, etc., cannot be accepted.

(ii) After the HV diagram is defined, it should be ascertained that the corridor permits takeoffs within ± 5 knots of the recommended takeoff profile.

(9) Flight Manual. The flight manual should list any procedures which may apply to specific points (e.g., high speed points) and test conditions, such as runway surface, wave height for amphibious tests, marginal areas of controllability or landing gear response, etc. The HV curve should be presented in the RFM using actual altitude above ground level and indicated airspeed.

(10) Night Evaluation. If a helicopter is to be certified for night operation, a night evaluation is required. Engine failures should be conducted along the recommended takeoff path. Landings should also be qualitatively evaluated with an engine failed. Engine failures at critical HV conditions are not required. The intent is to show adequate visibility using aircraft and/or runway lights without requiring a duplication of the daytime HV test program. See related discussion under paragraph 64 of this advisory circular.

(11) Water Landings. For amphibious float equipped rotorcraft, day and night water landings should be conducted under critical loading conditions with an engine failed. Engine failures should be conducted along the recommended takeoff path. Engine failures at critical HV conditions are not required. The intent is to show similarity to test results over land without requiring a duplication of the HV test program.

SECTION 4. FLIGHT CHARACTERISTICS

79. § 29.141 (through Amendment 29-19) FLIGHT CHARACTERISTICS - GENERAL.

a. Explanation.

(1) This regulation prescribes the general flight characteristics required for certification of a transport category helicopter. Specifically, it states that the helicopter shall comply with the flight characteristics requirements at all approved operating altitudes, gross weights, center of gravity locations, airspeeds, power, and rotor speed conditions for which certification is requested. While § 29.141(a) does not specifically refer to ambient temperature, the reference to "altitude" in § 29.141(a)(1) is correctly interpreted as "density altitude." Density altitude is, of course, a function of pressure altitude and ambient temperature, hence the need to account for ambient temperature effects. Additional flight characteristics required for instrument flight are contained in paragraph 775 of this advisory circular.

(2) Generally the aircraft structural (load level) survey accounts for takeoff power values at speeds up to and including V_Y. At speeds above V_Y, maximum continuous power is assumed. Stress to rotating components usually increases with airspeed and power. If the takeoff power rating exceeds the maximum continuous power rating, and the structural survey has been conducted under the assumption that takeoff power is not used at speeds above V_Y, the Rotorcraft Flight Manual must limit takeoff power to speeds of V_Y and below. If takeoff power is structurally substantiated throughout the flight envelope, and appropriate portions of the controllability, maneuverability, and trim requirements of §§ 29.141 through 29.161 are met at takeoff power levels, no flight manual entry is needed. Obviously if transmission limits for MC and takeoff power are coincident, no special action is needed.

(3) During the flight characteristics testing, the controls must be rigged in accordance with the approved rigging instructions and tolerances. The control system rigging must be known prior to testing. In addition to the normal rigging procedures, any programmed control surfaces which may be operated by dynamic pressure, electronics, etc., must also be calibrated. During the flight test program, it is frequently necessary to rig a control, such as the swashplate or tail rotor blade angle, to the allowable critical extreme of the tolerance band. For example, it would be necessary to rig the tail rotor to the minimum allowable blade angle if meeting the requirements of § 29.143(c) would be in question. The same consideration must be given to all helicopter controls and moveable aerodynamic surfaces where questionable compliance with the regulations may exist. If the rotor-induced vibration characteristics of the helicopter are significantly affected and require time-consuming rigging for such things as acceptable ride comfort, then the rotor(s) should be rigged to the allowable extreme tolerance limits to determine compliance, for example, with § 29.251.

(4) During the FAA flight test program, the crew should be especially alert for conditions requiring great attentiveness, high skill levels, or exceptional strength. If any of these features appear marginal, it is advisable to obtain another pilot's opinion and to carefully document the results of these evaluations. Section 29.141(b) provides the regulatory basis for these strength and skill requirements. The general requirements for a smooth transition capability between appropriate flight conditions are also included in § 29.141(b). These requirements must also be met during appropriate engine failure conditions for each category of rotorcraft.

(5) For night or IFR approval, § 29.141(c) contains the general regulatory reference which requires additional characteristics for night and IFR flight. The appropriate flight test procedures are included in other portions of this order.

80. § 29.143 (through Amendment 29-19) CONTROLLABILITY AND MANEUVERABILITY.

a. Explanation.

(1) This regulation contains the basic controllability requirements for transport rotorcraft. It also specifies a minimum maneuvering capability for required conditions of flight. The general requirements for control and for maneuverability are summarized in § 29.143(a) which is largely self-explanatory. The hover condition is not specifically addressed in § 29.143(a)(2) so that the general requirement may remain applicable to all rotorcraft types, including those without hover capability. For helicopters, the hover condition clearly applies under "any maneuver appropriate to the type."

(2) Paragraphs (b) through (e), § 29.143, include more specific flight conditions and highlight the typical areas of concern during a flight test program.

(i) Section 29.143(b) specifies flight at V_{NE} with critical weight, center of gravity (c.g.), rotor rpm, and power. Adequate cyclic authority must remain at V_{NE} for nose down pitching of the rotorcraft and for adequate roll control. Nose down pitching capability is needed for control of gust response and to allow necessary flight path changes in a nose down direction. Roll control is needed for gust response and for normal maneuvering of the aircraft. In the past, 10 percent control margin has been applied as an appropriate minimum control standard. The required amount of control power, however, has very little to do with any fixed percentage of remaining control travel. There are foreseeable designs for which 5 percent remaining is adequate and others for which 20 percent may not be enough. The key is, can the remaining longitudinal control travel at V_{NE} generate a clearly positive nose down pitching moment, and will the remaining lateral travel allow at least 30-degree banked turns at reasonable roll rates? Moderate lateral control reversals should be included in this evaluation and since available roll control can diminish with sideslip, reasonable out of trim conditions (directionally) should be investigated. This "control remaining" philosophy must also be applied for other flight conditions specified in this section.

(ii) Section 29.143(c) requires a minimum 17-knot control capability for hover and takeoff in winds from any azimuth. Control capability in wind from zero to at least 17 knots must also be shown for any other appropriate maneuver near the ground such as rolling takeoffs for wheeled rotorcraft. These requirements must be met at all altitudes approved for takeoff and landing. On helicopters incorporating a tail rotor, efficiency of the tail rotor decreases with altitude so that a given sideward flight condition requires more pedal deflection, a higher tail rotor blade angle, and more horsepower. Hence, directional capability in sideward flight (or at critical wind azimuth) is most critical during testing at a high altitude site. Prior to Amendment 29-24, hover controllability, height-velocity, and hover performance were the three regulatory requirements that ordinarily determined the shape of the limiting weight-altitude-temperature (WAT) curve for takeoff and landing. For Category A performance rotorcraft operations, of course, the one-engine-inoperative climb performance requirements may also influence the WAT limit curve. Amendment 29-24 allows, under certain conditions, the deletion of any hover controllability condition determined under Section 29.143(c) from becoming an operating limitation. Section 29.1587 of Amendment 29-24 provides a means wherein Category B certificated rotorcraft (in accordance with the requirements of 29.1, effective with Amendment 29-21) may not be limited by the hover controllability requirements of 29.143(c). Section 29.1583(g) requirements for Category A certificated rotorcraft are unchanged from past regulatory requirements in that if the hover controllability requirements of 29.143(c) result in the most restrictive envelope it will be published as an operating limitation. Section 29.1587(b) provides a means wherein Category B certificated rotorcraft, as defined in FAR 29.1, may not be restricted in its utilization. It allows such rotorcraft to publish the maximum takeoff and landing capabilities of the rotorcraft, provided something other than the 17 knot hover controllability requirement is not limiting. This may be zero wind IGE hover performance or any other performance the applicant elects to use if the maximum safe wind for operations near the ground is provided. Rotorcraft certificated prior to Amendment 29-24 can update their certification basis to take advantage of this provision. If an applicant with a previously type certificated rotorcraft elects to update to this later amendment, caution should be taken to verify that the height-velocity information is done in accordance with Amendment 29-21; that all engine out landing capabilities are satisfactorily accounted for at the new proposed gross weight, altitude, temperature combinations; that takeoff/landing information is provided; and that sufficient information is provided to properly advise the crew of the rotorcraft's capabilities when utilizing this increased performance capabilities.

(iii) Section 29.143(d) requires adequate controllability when an engine fails. This requirement specifies conditions under which engine failure testing must be conducted and includes minimum required delay times.

(A) For rotorcraft which meet the engine isolation requirements of Category A, demonstration of sudden complete single-engine failure is required at critical conditions throughout the flight envelope including hover, takeoff, climb at V_Y, and high speed flight up to V_{NE}. Entry conditions for the first engine failure are engine or transmission limiting maximum continuous power (or takeoff power where appropriate) including reasonable engine torque splits. For multiengine Category A installations (three or more engines) subsequent engine failures should be conducted utilizing the same criteria as that used for first engine failure. The applicant may limit his flight envelope for subsequent failures. Initial or sequential engine failure tests are ordinarily much less severe than the "last" engine failure test required by § 29.75(b)(5). The conditions for last engine failure are maximum continuous power, or 30-minute power if that rating is approved, level flight, and sudden engine failure with the same pilot delay of 1 second or normal pilot reaction time, whichever is greater.

* (B) For Category B powerplant installation rotorcraft, demonstration of sudden complete power failure is required at critical conditions throughout the flight envelope. This includes speeds from zero to V_{NE} (power-on) and conditions of hover, takeoff and climb at V_Y. Maximum continuous power is specified prior to the failure for the cruise condition. Power levels appropriate to the maneuver should be used for other conditions. The corrective action time delay for the cruise failure should be 1 second or normal pilot reaction time (whichever is greater). Cyclic and directional control motions which are part of the pilot task of flight path control are normally not subject to the 1-second restriction; however, the delay is always applied to the collective control for the cruise failure. If the aircraft flying qualities and cyclic trim configuration would encourage routine release of the cyclic control to complete other cockpit tasks during cruise flight, consideration should be given to also holding cyclic fixed for the 1-second delay. Although the same philosophy could be extended to the directional controls, the likelihood of the pilot having his feet away from the pedals is much lower, unless the aircraft has a heading hold feature. Rotor speed at execution of the cruise condition power failure should be the minimum power-on value. The term "cruise" also includes cruise climb and cruise descent conditions. Normal pilot reaction times are used elsewhere. Although this requirement specifies maximum continuous (MC) power, it does not limit engine failure testing to MC power. If a takeoff power rating is authorized for hover or takeoff, engine failure testing must also be accomplished for those conditions in order to comply with § 29.63(c). Following power failure, rotor speed, flapping, and aircraft dynamic characteristics must stay within structurally approved limits.

(iv) Section 29.143(e) addresses the special case in which a V_{NE} (power-off) is established at an airspeed value less than V_{NE} (power-on). For this case, engine failure tests are still required at speeds up to and including V_{NE} (power-on), and the rotorcraft must be capable of being slowed to V_{NE}.

86. <u>§ 29.177 (through Amendment 29-24) STATIC DIRECTIONAL STABILITY</u>.

a. <u>Explanation</u>. This rule requires that positive static directional stability be demonstrated at the <u>trim</u> airspeeds defined in § 29.175. The trim speed for climb is V_Y and for cruise is $0.9V_H$ or $0.9V_{NE}$ (whichever is less). For autorotation that airspeed defined by the midpoint of the speed range specified in § 29.175(c) may be used.

b. <u>Procedure</u>.

(1) Tests for static directional stability require instrumentation for pedal position and sideslip angle. Lateral cyclic control position instrumentation should be provided for IFR certification tests. To obtain accurate sideslip angle and airspeed information, a "yaw boom" is usually installed for the purpose of mounting a sideslip vane and swiveling airspeed pitot head outside the main rotor downwash region of influence. Special care should be taken to ensure that the yaw boom installation has been verified to be structurally adequate and free of dynamic instabilities for all combinations of airspeed and rotor speed likely to be experienced during the static directional evaluation. For some installations, the instrumentation yaw boom may influence the flying qualities of the rotorcraft itself. Thus, it is advisable to correlate yaw string displacement or slip indicator ball widths of skid with yaw boom sideslip angle, and then repeat a few critical points with the yaw boom removed.

(2) For some rotor system designs, the main and tail rotor flapping angle may be a critical instrumentation requirement for static directional testing. Both main and tail rotor flapping may increase dramatically at high airspeeds with increasing sideslip angle. Therefore, for rotor systems exhibiting this characteristic, flapping should be monitored carefully during the sideslip maneuver to avoid exceeding limitations. Static directional stability is normally defined in terms of pedal displacement required to maintain a straight flight path sideslip. A single-rotor helicopter flying in coordinated flight will exhibit a small inherent sideslip due to tail rotor thrust and fuselage/main rotor sideforces. This condition is normally taken as trim with the inherent sideslip angle noted. Airspeeds should be the trim values described above. A generally accepted technique follows:

(i) Stabilize at the trim point, and note indicated airspeed.

(ii) Record trim conditions including inherent sideslip. Maintain fixed collective and throttle for the remainder of the maneuver.

(iii) Smoothly yaw the aircraft with directional control and coordinate with lateral control to establish the desired sideslip angle. A steady heading can best be ensured by maintaining a track over a straight landmark on the ground such as a section line or straight segment of powerline or highway.

(iv) Note airspeed immediately upon completion of the yaw maneuver. There may be a small change from the trim airspeed. Fly the new airspeed while maintaining a constant heading, and record indicated airspeed, control positions (directional at a minimum), sideslip angle, rotor speed, rate of descent, amount of ball deflection, and bank angle. The pilot should note the physical sideforce feel experienced. A minimum of two sideslip data points on each side of the trim point should be obtained to adequately define the slope of the pedal displacement versus sideslip angle relationship.

(v) Smoothly return the aircraft to the inherent sideslip angle. Static directional stability plots can be expected to differ slightly on either side of the inherent sideslip angle. Positive static directional stability is indicated by increased left pedal displacement for a larger right sideslip and, conversely, increased right pedal for a larger left sideslip angle.

87. § 29.181 (through Amendment 29-24) DYNAMIC STABILITY: CATEGORY A ROTORCRAFT

a. Explanation. This section requires that Transport Category A rotorcraft, certificated under Amendment 24 of FAR 29, demonstrate positive damping for short-period oscillations (5 seconds or less) at forward speeds from V_Y to V_{NE} with primary controls free and fixed. This requirement would prevent persistent or divergent short-period oscillations and thus alleviate the pilot workload to actively dampen oscillatory motions for all types of operations.

b. Procedures.

(1) Tests for short period dynamic stability are carried out in the same manner as for IFR (Ref. Para 775 of AC 29-2A) except the oscillation need not be damped as heavily (i.e., to 1/2 amplitude in not more than one cycle). Similarly pulses and doublets may be used to generate an upset condition that would be expected to be encountered in moderate turbulence for that particular rotorcraft.

(2) Tests should be conducted at the critical gross weight, altitude, center of gravity, rotor rpm, and power conditions during routine climb, cruise, and descent condition for speeds from V_Y to V_{NE}. This test must be conducted with the minimum amount of stability augmentation approved for continued safe flight. Consideration should be given to optional equipment that are to be mounted externally.

(3) This requirement is not applicable to transport category rotorcraft certificated as Category B only. The requirements for this situation are unchanged. *

88.-95. RESERVED.

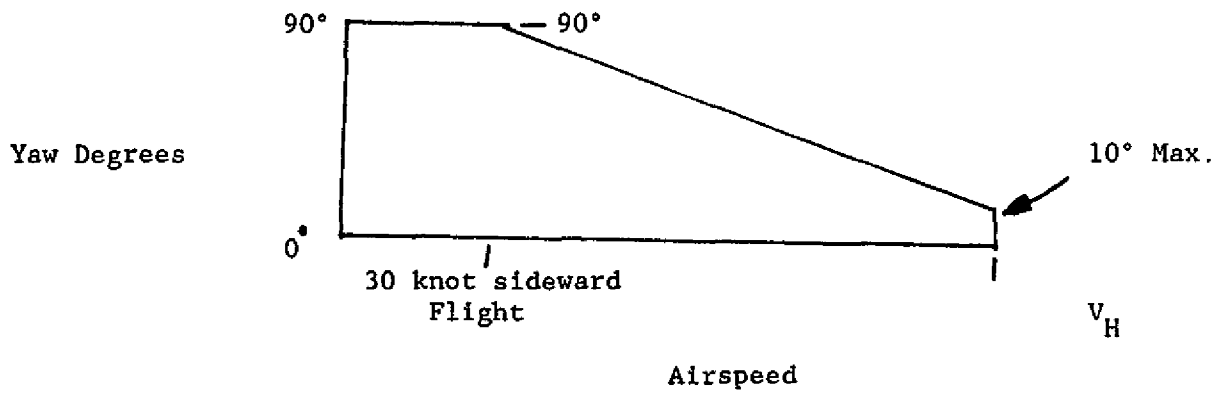

FIGURE 140-1

(viii) During flight test evaluations, yaw angles have been measured using a yaw angle probe (swiveling vane type) on a nose boom. Both a visual readout for the pilot and a record, such as an oscillograph trace, have been used. This test may be conducted in the flight test program or in the flight load survey program. This record should confirm the yaw angle used in design as conservative with respect to operational and actual flight characteristics. This test is not a requirement however.

141. § 29.361 (through Amendment 29-19) ENGINE TORQUE.

a. Explanation.

(1) The rotorcraft should be designed for limit engine torque values, as prescribed by the rule, to account for maximum engine torque, including certain transients and torsional oscillations. The rule recognized that reciprocating (piston) engines generate higher torque oscillations than turbine engines.

(i) A factor of 1.25 applies to maximum continuous power for turbine engines. Section 29.923 refers to torque output and Section 29.927(b) refers to other torque output conditions for use in an "endurance test."

(ii) Torque factors are also specified for reciprocating engines having two or more cylinders in § 29.361(a)(2) or § 29.361(b) of Amendment 29-26. The appropriate torque factor applies to takeoff power torque as well as maximum continuous power and other power conditions.

(2) Amendment 29-26 introduced additional turbine engine installation considerations for the following:

(i) Engine torque loads associated with emergency operation of governor-controlled turboshaft engines.

(ii) Torque reaction loads from sudden turbine engine stoppage which is applied to the engine and the engine suspension and restraint system.

(3) Paragraph 206 of this document concerns § 29.549(c) and (e) that contains design standards for engine mounts and adjacent structure for flight and landing and also flight with 2 1/2-minute OEI power rating. Amendment 29-26 added OEI power to the standard.

(4) Section 29.547(e)(1)(ii) concerns the application of limit engine torque to design of the main rotor structure. *

b. Procedures.

(1) The engine torque associated with the maximum continuous power condition should be multiplied by the appropriate torque factor to obtain the engine torque value used for structural substantiation purposes of the rotorcraft.

(2) The torque values associated with the minimum power-on rpm limit should be used. Maximum power-on speed limit will result in a lower torque value when calculating torque from design horsepower values. However, due to piston engine power output characteristics, an engine may produce a higher torque at higher engine speeds contrary to the previous statement. The torque factor should account for this characteristic.

(3) For turbine engines limit torque values are determined for the four cases cited. Two cases are related to "endurance" test standards.

(4) For sudden stoppage of turbine engines the engine manufacturer can reasonably provide engine rotating inertia and deceleration time expected in the event of sudden engine stoppage which generates these critical loads in the engine mounting and restraint system. These manufacturers' data should be acceptable for use in complying with this part of the design standard. *

142.-151. RESERVED.

SECTION 9. CONTROL SURFACE AND SYSTEM LOADS

152. § 29.391 (through Amendment 29-19) GENERAL.

a. Explanation. This general rule concerns requirements for design loads of tail rotors, control or stabilizing surfaces, and their control system.

b. Procedures. The design criteria and/or the design loads report must contain the loads dictated by the referenced rules. See paragraphs 153, 154, 155, 156, 157, 158, and 159 of this document.

153. § 29.395 (through Amendment 29-19) CONTROL SYSTEM.

a. Explanation. Control system design loads and the application of these loads are contained in this rule.

(1) Paragraph (a) of the rule specifies the way or means of reacting the design loads specified in §§ 29.397 and 29.399 (for dual control systems). The design loads must be imposed on any locks and stops and irreversible mechanisms in the control system. Both rotor blade horns and control surface horns must react, without failure, the specified loads while the controls are in critical positions.

(2) Paragraph (b) of the rule specifies application of limit pilot forces or of the maximum loads that can be obtained in normal operation, including any single power boost system failure, whichever is greater. However, minimum limit pilot force 0.60 of the loads specified in §§ 29.397 and 29.399, may be used, as specified, in parts of the primary control system that are not stiff enough to react to the loads specified in the first part of paragraph (b) of the rule. Note the objective for a rugged control system.

(3) Control system design feature and test requirements are found in §§ 29.671 through 29.695. Bearing factors and fitting factors are specified in §§ 29.623 and 29.625, respectively.

b. Procedures.

(1) The design criteria and/or a design loads report that includes the primary control system design loads should be submitted for FAA approval.

(2) The rotorcraft control system may be tested to ultimate design loads or may be analyzed for the ultimate design loads. See paragraph 124 of this document.

(i) It is advisable that the applicant prepare a proposal describing the procedures and techniques to be used in the static testing of the control system which reflects compliance with the condition specified. It is further advisable that the FAA concur that the tests proposed achieve that objective. Omission of these steps may result in the need for retesting. The test results should be documented.

(ii) If tests are not conducted, a structural analysis of the control system is required. Appropriate factors from §§ 29.685(e), 29.623, and 29.625 must be used as specified. A structural analysis report should be used to document compliance with § 29.685(d)(1) and (4), and § 29.685(f).

(3) If a part of the control system is not stiff or rigid enough to react the design loads specified in § 29.397, that part of the system may be substantiated for lower loads as prescribed.

(i) The limit design loads are those loads specified in § 29.397;

(ii) The limit design loads are the maximum that can be obtained in normal operation, including any single power boost system failure, except for objectives stated for a rugged system; and

(iii) In lieu of a rational analysis, the limit design loads may be 0.60 of the loads specified in § 29.397.

(iv) For example, if a control surface servo tab or a small elevator is a part of the helicopter design, the control system for this part must be stiff enough to react the control surface loads without failure and to provide enough surface deflection to control the helicopter. These limit loads may be 60 pounds fore and aft and 40 pounds laterally on the cyclic control stick in lieu of a rational analysis and may be the maximum loads that can be obtained in normal operation.

(v) If a hydraulic power actuation or boost system is part of the helicopter design, the design limit load for the affected parts of the control system will be the maximum output force of the boost at normal operating pressure added to the limit design loads resulting from the loads specified in § 29.397. If a single failure in the power portion of the hydraulic system results in actuator forces that exceed the maximum output force at normal operating pressure, the highest output loads must be used as noted in subparagraph (3)(ii). This hydraulic system failure standard is specified in § 29.695(a)(1) as well.

(4) Controls proof and operation test is required by §§ 29.307(b), 29.681, and 29.683. This test is conducted using the design limit loads approved under § 29.395(b). See paragraphs 282 and 283 of this document.

154. <u>§ 29.397 (through Amendment 29-19) LIMIT PILOT FORCES AND TORQUES</u>.

a. <u>Explanation</u>. Design forces are contained in the rule.

(1) Primary controls, pilot and copilot, must be designed for the limit pilot forces specified in paragraph (a) of the rule.

(2) For other operating controls, such as flap, tab, stabilizer, rotor brake, and landing gear, design limit forces are specified in paragraph (b).

206. <u>§ 29.549 (through Amendment 29-19) FUSELAGE AND ROTOR PYLON</u>.

a. <u>Explanation</u>. This regulation requires that the fuselage and rotor pylon (including the tail fin, if any) be designed to withstand the flight loads of §§ 29.337 through 29.351, the ground loads of §§ 29.235, 29.471 through 29.497, ski loads of § 29.505, water loads of § 29.521, and rotor loads of § 29.547(d)(1) and (e)(1)(i). The ski and water loads pertain to optional features.

(1) Consideration is also required of--

(i) Auxiliary rotor thrust;

(ii) The torque reaction of each rotor drive system; and

(iii) Balancing air and inertia loads.

(2) Each engine mount and adjacent fuselage must be substantiated as prescribed. In addition, if 2 1/2-minute power is used, "each engine mount and adjacent structure must be designed to withstand the loads resulting from a limit torque equal to 1.25 times the mean torque for 2 1/2-minute power combined with 1g flight loads." Amendment 29-26 extended paragraph (e) of the standard to 2 1/2 minute "OEI power." *

b. <u>Procedures</u>. Compliance with this standard is accomplished by application of the specified aircraft loads including engine torque to the fuselage and rotor pylon structure by stress analyses and/or static tests. Drive system torque factors to be used are noted in § 29.547 for the main rotor structure as well as in paragraph (e) of this standard.

207. <u>§ 29.551 (through Amendment 29-19) AUXILIARY LIFTING SURFACES</u>.

a. <u>Explanation</u>. This regulation specifies that auxiliary lifting surfaces be designed to withstand critical flight and ground loads derived for conditions specified and any "other critical condition expected in normal operation." Stub wings would comply with this standard.

b. <u>Procedures</u>. The surface design loads are derived from the conditions specified. Conservative aerodynamic data, including load distributions, may be used in place of data derived from wind tunnel or instrumented flight testing of the exact aerodynamic shapes involved. Special attention should be placed on concentrated load effects from fuel tanks or other large mass items that may be located in lifting surfaces. These types of load concentrations are to be considered in conjunction with inertia and aerodynamic loads.

208.-217. <u>RESERVED</u>.

SECTION 13. EMERGENCY LANDING CONDITIONS

218. § 29.561 (through Amendment 29-24) GENERAL.

a. Explanation.

(1) The occupants should be protected as prescribed from serious injury during an emergency/minor crash landing on water or land for the conditions prescribed in the standard. The standard states that each occupant should be given every reasonable chance of escaping serious injury in a minor crash landing.

(2) Section 29.561(b)(3) specifies certain ultimate inertial load factors but allows a lesser downward vertical load factor by virtue of a 5 fps ultimate rate of descent at maximum design weight.

(3) In addition, the occupants must be protected from items of mass inside the cabin as well as outside the cabin. For example, a cabin fire extinguisher must be restrained for the load factors prescribed in this section. A transmission or engine must be restrained to the load factors in § 29.561(b)(3) if located adjacent to, above, or behind the occupants.

(4) For aircraft equipped with retractable landing gear, the landing gear must be retracted for compliance.

(5) Fuel tank protection.

(i) Underfloor fuel tanks are specifically addressed in § 29.561(d). The fuselage structure must be designed to resist crash impact loads prescribed in § 29.561(b)(3) and to also protect the fuel tank from rupture as prescribed. The landing gear must be retracted if the rotorcraft is equipped with retractable gears.

(ii) Section 29.963(b), a general fuel tank design standard, also refers to § 29.561. This standard specifies that each tank and its installation must be designed or protected to retain fuel without leakage under the emergency landing conditions in § 29.561. Paragraph 454 of this AC relates to this standard.

(6) The minor crash conditions contained in § 29.561(b)(3) must also be considered in designing doors and exits (§ 29.783(d) and (g), and § 29.809(e)).

b. Procedures.

(1) The design criteria report or another similar report of the rotorcraft structural limits should contain the (ultimate) minor crash condition load factors.

(2) Section 29.785 (paragraph 336 of this AC) concerns application of this design standard to seats (berths, litters), belts, and harnesses.

(3) The ultimate design landing and maneuvering load factors may exceed the minor crash condition load factors. The highest load factor derived must be used.

(iii) Ball or roller bearings may be used to conduct the current at rotating joints. However, increased friction or possible seizure of the bearing may occur. The potential for this should be evaluated. Inspection and replacement criteria for possible damage should be addressed in the manual for continued airworthiness. Bearings are especially susceptible to pitting and internal arcing.

(iv) Report DOT/FAA/CT-86/8, April 1987, Determination of Electrical Properties of Grounding, Bonding, and Fastening Techniques for Composite Materials, may assist the applicant.

(4) Fuel Systems. Refer to Report DOT/FAA/CT-83/3 referenced in paragraph 245a. For additional information on the lightning protection requirements for fuel systems for rotorcraft with a certification basis which includes Amendment 29-26 refer to paragraph 449 of this AC. *

246. <u>§ 29.611 (through Amendment 29-19) INSPECTION PROVISIONS</u>.

a. <u>Explanation</u>. The rotorcraft must have access panels, or openings, that will allow for proper maintenance and/or adjustment of the rotorcraft systems.

(1) The rule states: There must be means to allow close examination of each part that requires recurring inspection, adjustment for proper alignment and functioning, or lubrication.

(2) "Structural" or load-carrying access panels may be used to comply with the rule. Structural panels should have stencils or permanent labels (§ 29.1541(a)(2)) stating the panels must be installed prior to ground or flight operation.

(3) Holes or "nonstructural" access panels should be used whenever possible.

b. <u>Procedures</u>.

(1) The determination of compliance can be accomplished in conjunction with the following activities:

(i) Reviewing type design drawings.

(ii) Conformity inspections accomplished during certification testing.

(iii) Be evaluated during the control system proof and operation tests (§§ 29.681 and 29.683).

(iv) During type inspection tests and functioning and reliability testing.

(2) Equipment requiring frequent inspections (at less than 25-hour intervals), lubrication, or adjustments should be accessible through "nonstructural" doors. Areas or items requiring daily attention should be accessible through "nonstructural" doors since properly rated maintenance personnel are required to "open and close," or reinstall structural panels and special design features, such as multiple pins and latches, are generally necessary for structural doors.

(3) If the rotorcraft is subject to an FAA Maintenance Review Board Approval Program, further review of the rotorcraft inspection provisions will be obtained.

251. <u>§ 29.625 (through Amendment 29-19) FITTING FACTORS</u>.

a. <u>Explanation</u>. A 1.15 factor is specified to assure that the calculated load and stress distribution within any fitting is conservative. Application of the factor is excluded or excepted as stated in the rule.

b. <u>Procedures</u>.

(1) The factor may be applied to the calculated load or stress for the fitting.

(2) The structural substantiating data for the rotorcraft, including the rotor system, must include the prescribed fitting factor. The rotor system includes the flight control system rotor head and hubs and rotor blade attachments.

252. <u>§ 29.629 (through Amendment 29-19) FLUTTER</u>.

a. <u>Explanation</u>.

(1) The rotorcraft must be free from flutter.

(2) Section 29.251 vibration is an associated flight requirement concerning flight demonstrations. See paragraph 110 of this document for this standard.

(3) Section 29.571(a)(3) concerns in-flight measurement of loads or stresses.

b. <u>Procedures</u>.

(1) Freedom from flutter may be shown by analysis or appropriately instrumented flight flutter tests.

(2) The flight loads survey proposal submitted for compliance with § 29.571 may also contain tests to fulfill compliance with § 29.629. The flight loads survey program encompasses the envelope of design airspeed and rotor r.p.m., and sufficient aerodynamic excitation is generally present to excite any latent flutter modes.

(3) Flight loads survey data or flight flutter test data submitted should be reviewed to assure that excessive oscillatory loads of rotors or surfaces will not be encountered.

253.-264. <u>RESERVED</u>.

SECTION 16. ROTORS

265. § 29.653 (through Amendment 29-22) PRESSURE VENTING AND DRAINAGE OF ROTOR BLADES.

a. Explanation. The rule requires each rotor blade to be provided with venting and drainage means (i.e., holes, etc.) or the blade must be sealed and designed to withstand internal pressure.

b. Procedures. Although the rule provides for venting and drainage features, recently certificated blades have been designed to be sealed and to sustain the "maximum pressure differentials expected in service." For modern blade designs, the internal pressure buildup due to environmental effects and centrifugal acceleration effects (near the tip) can be readily sustained with moisture sealing accomplished. The use of sealed blades is highly advantageous and recommended because of the possibility for severe corrosion damage resulting from trapped moisture and because of the difficulty in finding internal corrosion damage by use of field level inspections.

266. § 29.659 (through Amendment 29-22) MASS BALANCE.

a. Explanation. The rule requires that mass balancing of rotors and blades be provided, as necessary, to prevent excessive vibration and flutter. Further, the rule requires structural substantiation of the mass balance installation.

b. Procedures.

(1) The weight, geometry, and location of rotor and blade mass balance devices are determined as the requirements of §§ 29.571 and 29.629 are met.

(2) The structural substantiation should show static strength to meet the maneuver and gust loads of §§ 29.337, 29.339, and 29.341. In addition, the main rotor loads of § 29.547(c) should be substantiated. The fatigue strength of the mass balance devices (including structural supports) should meet the requirements of § 29.571.

(3) In addition to the appropriate strength requirements, some recent designs have included features which trap the balance weight inside a limited area even if the primary attachment means (adhesive, bolts, etc.) fail. This type of design feature is recommended because of the severe loading environment to which balance devices are subjected.

267. § 29.661 (through Amendment 29-19) ROTOR BLADE CLEARANCE.

a. Explanation.

(1) The rotors, main and tail, must not strike other parts of the rotorcraft during any operating condition.

(2) Section 29.411 concerns protection of the tail rotor from a ground strike.

b. Procedures.

(1) The applicant should have drawings or sketches of the rotorcraft that show an adequate minimum clearance between the rotors, main and tail, and parts of the rotorcraft. Probable flexing of the rotors should be considered in determining the minimum clearance.

(2) During parts of the FAA-conducted flight test program, frangible devices (wood dowels) or other means of measuring clearance, may be requested to confirm that the clearance shown in the drawings or sketches is adequate in certain operating conditions. Balsa wood dowels may be clamped to the aft part of the fuselage within the rotor arc. If the devices are intact after autorotation landing tests and other tests involving typical abrupt, cyclic, and rudder pedal displacements, the clearance should be satisfactory and compliance obtained. If such measuring devices are used, the type inspection report should contain a record of clearance found during the tests. It is not necessary to precisely determine the clearance but only necessary to determine "enough clearance" as stated in the rule.

* 268. § 29.663 (through Amendment 29-30) GROUND RESONANCE PREVENTION MEANS.

a. Explanation.

(1) This section, adopted in Amendment 29-3, and amended by Amendment 29-30, requires reliability and damping action investigation for the ground resonance prevention means which typically includes the shock struts. Section 29.1529 requires associated maintenance information in the maintenance manual. The probable range of variations in service, not just the allowable range, should be established and investigated as prescribed. This probable range includes operation on the ground, or other appropriate landing surface applicable to the rotorcraft design. Quantitative test data are generally obtained in compliance with this rule although analysis or tests may be employed. The preamble to Amendment 29-3 contains additional information.

(2) Note that the maintenance information is not contained in the approved mandatory section of the maintenance manual.

(3) Paragraph 99 concerns demonstrating freedom from ground resonance during certain applicant and TIA verification evaluations or tests of the rotorcraft. Section 29.241 complements the requirements of § 29.663. As noted in paragraph 99 of this document, the FAA removed from CAR Part 7 a specific requirement for a ground vibration survey. However, § 29.663 was adopted by Amendment 29-3 to investigate possible sources of ground resonance and to assure that the reliability of the ground resonance prevention means; i.e., dampers, shock struts, etc., would preclude the occurrence of ground resonance. The total rotorcraft system, including landing gear, struts, tires, etc., is evaluated under this standard.

(4) Viscous dampers in the rotor head have been used for many years to prevent ground resonance. Modern rotorcraft designs may also use elastomeric dampers and may use elastomeric bearings in the rotor head and rotor pylon attachment to the airframe. The standard applies to viscous and elastomeric dampers. The "probable" range in damping shall be investigated. The standard also requires investigation of the probable range of variations of these dampers, whether viscous or elastomeric, and elastomeric bearings to preclude ground resonance.

(5) Ground resonance can occur due to flexibility in the rotor pylon restraint system as well as with landing gear flexibilities and/or shock struts. See Paragraph b(2) for an explanation. An analysis may be done to show the effect of the rotor pylon mount stiffness on ground resonance stability. If the analysis shows that rotor pylon mount stiffness could affect ground resonance, the evaluation should include variations in stiffness and damping of the rotor pylon restraints that may occur in service (reference "Ground Vibrations of Helicopters, "M. L. Deutsch, JAS, Vol 13, No. 5, May 1946).

(3) The required "down-lock" should be checked during the operation test. The design drawing should be reviewed for compliance prior to conducting an operation test. The "down-lock" system should be evaluated for § 29.1309 function and reliability requirements.

(4) If an optional "up-lock" is installed (including hydraulic locking), the landing gear should be extended during the operation test after simulation of critical failure mode of the retraction system (ref. § 29.1309).

(5) An "operation" test plan or proposal submitted for compliance with § 29.729(d) should include the items noted in the two previous subparagraphs and should include a functional check of the position indicator system. Those tests must be satisfactorily completed before issuing the TIA.

(6) During the official FAA flight tests, compliance with the emergency operation, position indicator, and control aspect of § 29.729(c), (e), and (f), respectively, will be verified or accomplished. In addition, the F and R test program plan (§ 21.35) will specify certain tests or evaluations for the retraction system.

(7) Position Indicator Evaluation.

(i) When evaluating the position indicator system, emphasis should be placed on the switches and their installations, and on the cockpit presentation. Each gear must have its own set of switches to indicate when it is secured in its extreme "up" position and its extreme "down" position. The switches must be located to give a valid indication of the arrival of the gear at its extreme position.

(ii) The reliability and environmental qualifications of the switches to be used should be carefully considered. An example of a condition that has potential for trouble is operation on wet areas. Trouble starts when water is picked up by the tires and deposited on the switches. During winter months the water can freeze, and the resulting ice may prevent the switch from functioning properly.

(iii) An acceptable cockpit presentation consists of two lights for each gear. One light is colored "green" and indicates when its gear is secured in the extreme "down" position. The other light is colored "amber" or "red" and indicates when its gear is in transit. When the gear is in either extreme position, the in transit light is "out." For this presentation, the indication to the pilot that the gear is in the extreme "up" position is an all-gear, lights-out condition.

(iv) Some manufacturers have also included a warning system to alert the crew if the landing gear has not been extended prior to landing. If a warning system is presented, §§ 29.1301 and 29.1309 should be used to evaluate its functional characteristics and the impact of its failure modes.

302. § 29.731 WHEELS.

a. Explanation. This standard requires use of approved wheels, either approved under TSO-C26 or a later revision or approved under the type certificate for the aircraft. Wheels must satisfy both a design static (1g) load and design limit landing or taxiing load determined under the applicable ground load requirements. Standards for a tire installed on a wheel are contained in § 29.733.

b. Procedures.

(1) The structural design loads data shall contain both a static load and a landing and taxiing load for each wheel. These loads are determined by virtue of compliance with the standards of § 29.731(b) and (c). The ratings of the wheel shall not be exceeded. TSO-C26c contains minimum performance standards for TSO approval of aircraft wheels and wheel-brake assemblies. Ratings are assigned in accordance with this performance standard.

(2) If a wheel selected for an aircraft design has TSO approval, the wheel manufacturer will supply the rating to the aircraft manufacturer. Each wheel shall be marked as prescribed which includes a listing of the TSO number. Even though a wheel is TSO approved, the application on the aircraft (loads imposed on the wheel) requires proof that the rating is not exceeded.

(3) If a wheel selected for an aircraft design is not approved under a TSO, the necessary data, both detail design and assembly drawings and qualification tests and test report data, will be required to comply with the standards contained in Part 29. Design control and inspections will be accomplished as a part of the aircraft type design. Structural substantiation and any appropriate qualification tests shall be accomplished. See §§ 29.471 through 29.497 and § 29.511 for the ground load conditions.

(4) The Tire and Rim Association, Inc., generally issues a yearbook listing tire and rim sizes and ratings. The dimensions and contours for aircraft wheel rims are contained in Section 9 of this yearbook.

303. § 29.733 (through Amendment 29-12) TIRES.

a. Explanation.

(1) This standard specifies both design and performance criteria for tires. The tire must fit the wheel rim. The maximum static ground reaction for the condition specified must not exceed the maximum static load rating of each tire. In addition, any tire of retractable gear systems must have adequate clearance from surrounding structure and systems as specified.

(2) Main, nose, and tail wheel tires must comply.

(3) Rotorcraft design maximum weight shall be used. Static and "dynamic" conditions are specified for rotorcraft tires.

(4) Tire performance standards are contained in TSO-C62c.

b. Procedures.

(1) The aircraft structural design loads should contain a maximum static load imposed on the tires. The load is derived for a static ground reaction assuming the design (maximum) weight and the critical center of gravity for each tire of the landing gear. The wheel loads are determined under § 29.731(b). Reduced weight but forward c.g. conditions may result in the highest static load on a nose wheel tire. Thus, combinations of weight and c.g. locations require investigation for the maximum tire load of each main, nose, and tail wheel tire. Nose wheel tires are subject to a specific dynamic condition.

(2) The maximum possible size of the tires considering appropriate temperatures, aging, and pressure should be obtained to check wheel well and cover clearances. Tire dimensions (for clearances) may be found in the yearbook noted in paragraph 303b(4). If the tire clearance is questionable, objects may be taped to the tire to simulate tire growth or oversize dimensions expected and the wheel retracted and rotated by hand to check for possible interferences. Minimum clearance, such as one-half inch, may be adequate as a design objective. The design drawings should be reviewed for information of correct systems installations and landing gear rigging within the wheel wells and wheel covers, if installed. If necessary to control tire sizes, specific manufacturer's tires should be used as "required equipment" and the tire manufacturer and the part number should be specified in the design data and on the type certificate data sheet as "required equipment."

(3) As specified in § 29.729(d), an operation test of any retractable landing gear should be performed. During this operation test, the tire clearances should be determined and recorded for the maximum tire size expected in service. Only the least or minimal clearance found, if adequate, should be recorded. *

(4) The Tire and Rim Association, Inc., generally issues a yearbook listing tire and wheel rim sizes and ratings. This information is advisory as stated in the yearbook. Section 9 concerns aircraft tires and rims. Table AP-5 in Section 9 of the yearbook concerns tires used on helicopters. The tire may be selected initially from the yearbook, but qualification data for the specific tires used shall be furnished with the type design data in compliance with the standards. Section 9 also contains tire size and tire growth dimensions.

(5) Minimum performance standards for aircraft tires, excluding tail wheel tires are found in TSO-C62c, Aircraft Tires. Tires meeting the TSO are marked as prescribed in the standards. The load rating (ref. § 29.733) is marked on the tire. TSO tires are not required but should be used whenever possible. The manufacturer's information, such as load rating, should be included in the aircraft type design structural substantiation data.

304. <u>§ 29.735 (through Amendment 29-24) BRAKES</u>.

a. <u>Explanation</u>.

(1) Brakes are required for wheel landing gear aircraft. Minimum performance standards are contained in this section. During the course of the FAA flight test program and of any F&R program conducted under § 21.35, the brakes shall be used and evaluated.

(2) Design criteria are contained in this standard.

(i) The braking device must be controllable by the pilot. It is optional for the second pilot station except as may be specified under the provisions of § 29.771.

(ii) The braking device must be usable during power-off landings.

(3) Performance criteria are also contained in this standard.

(i) The brakes must be adequate to counteract any normal unbalanced torque when starting or stopping the rotor or rotors.

(ii) The brakes must be adequate to hold the rotorcraft parked on a 10 degree slope on dry, smooth pavement.

(4) In §§ 29.493(b)(2) and 29.497(g)(2)(ii), limiting brake torque is one ground load standard for design of the landing gear.

(5) Although not specifically noted in a standard, the position of the brake on the wheel is important. The brake should be positioned to avoid ground contact whenever the tire is deflated.

(6) TSO-C26c contains minimum performance standards for aircraft landing wheels and wheel-brake assemblies. For rotorcraft, a wheel-brake assembly design rating is established by the manufacturer. The TSO standard for rotorcraft brakes specifies a 20 degree slope standard (rather than a 10 degree slope) for an over-pressure hydraulic brake test. *

(7) The brake application device at the pilot station is subject to other structure strength standards in this Part, such as the limit pilot forces or torque specified in § 29.397.

b. Procedures.

(1) Wheel-brake assemblies approved under TSO-C26 or a later revision will have various (rotorcraft) ratings as specified in the standard. One rating of TSO standard for a rotorcraft wheel-brake assembly is the kinetic energy capacity in foot-pounds at the design landing rate of absorption. The design takeoff and landing weight and rotorcraft speed in knots for brake application are a part of the equation. The brake manufacturer should furnish this rating and the two noted parameters for the selected design or designs. The ratings of selected brakes should be included in a structural design data report such as a design criteria report. The use or application of each brake design on the particular rotorcraft design should not exceed capacity of the brake or the ratings established under the TSO. If appropriate, the part number and manufacturer of each brake may be listed in the structural data reports as well as listed in the type design drawings.

(2) The limiting brake torque obtained from the brake manufacturer should be used in complying with §§ 29.493(b)(2) and 29.497(g)(2)(ii).

(3) Compliance with the brake standards should be confirmed, demonstrated, and recorded as a part of the flight test type inspection report. This applies to TSO brakes and to brakes approved as a part of the aircraft type design.

(4) If found necessary under the provisions of § 29.771, the second pilot station should have brake control devices. The brake control devices should be listed with the other required equipment that defines the equipment necessary for a second pilot station.

(5) A brake assembly may be evaluated and approved under Part 29 as a part of the aircraft type design. TSO-approved brakes are not specifically required but are recommended. For non-TSO-approved brakes, all detail and assembly drawings, required test proposals, and test results reports may be submitted and processed as a unique part of the particular aircraft type design.

(6) During an inspection of the landing gear, such as an engineering compliance inspection, the brake location should be checked to ensure the brake does not contact the ground when the tire is deflated. Type design drawings should control the proper location of the brake on the landing gear. *

305. § 29.737 SKIS.

a. Explanation. This standard is, in part, derived from small airplane standards. Aircraft skis approved under TSO-C28 may be used on rotorcraft. TSO-C28 for aircraft skis refers to Sections 4 and 5 of National Aircraft Standards Specification 808, dated December 15, 1951, for strength and performance standards. The standard also addresses flight/aerodynamic loads.

(1) A maximum limit load rating is assigned to each ski approved under TSO-C28.

(2) This limit load rating must not be exceeded by the maximum limit ground load determined under the standards of § 29.505, Ski landing conditions.

(3) The ski installation is also subject to the maximum aerodynamic and inertia loads and to the ground rotation or torque load per § 29.505(c).

(4) Ski mounting or installation parts used in the particular application are subject to substantiation as any landing gear member is subject to substantiation.

(5) Ski installations are also subject to flight and ground operation evaluations.

(6) Pads or "bear paws" on skid or wheel landing gears for use in snow or soft soils are unique to rotorcraft. These shall be approved also. For new type certificate applications after November 27, 1989, § 29.571, Amdt. 29-28 requires fatigue substantiation of the landing gear. The effect of pads, etc., shall be evaluated in compliance with the standard.

b. Procedures.

(1) The limit load rating for the ski selected shall be obtained from the ski manufacturer. This information shall be included in the design criteria and/or structural substantiation reports. The type design drawings will include the appropriate part number for the TSO-approved product and the necessary installation information.

(2) The design limit loads derived in compliance with § 29.505 shall not exceed the ski limit load rating. The skis shall be substantiated for the torque load in § 29.505(c) since the TSO standard does not contain a similar requirement.

(3) Skis that are not TSO approved may be approved as a part of the aircraft type design by complying with the strength and performance standards contained in TSO-C28 (NAS 808).

(4) The aerodynamic loads shall be based on a limit load design speed of 1.11 V_{NE}. The maximum V_{NE} used in design may be reduced only for a "ski configuration" airspeed limitation.

(5) Pads or "bear paws" installed on skid or wheel landing gear to facilitate operations in snow conditions or marsh lands may be approved as a part of or as an alteration to the aircraft type design. Rational flight and landing design loads applicable to the particular pad design must be developed and strength substantiating data submitted proving compliance with the strength and performance standards contained in Part 29. In addition, skid landing gear may be subject to excessive vibratory loads while in flight whenever the weight and mass distribution is altered by adding "bear paws." The effect of additional weight should be investigated over the flight operating regimes, including the approved range of rotor speeds. Resonant vibratory conditions should be avoided or highly damped, thus avoiding a potential change in service life. In compliance with § 29.571, Amdt. 29-28, stress measurement, etc., may be necessary, if the standard is applicable. *

306.-315. RESERVED.

SECTION 19. FLOATS AND HULLS.

316. § 29.751 (through Amendment 29-3) MAIN FLOAT BUOYANCY.

a. Explanation.

(1) This section specifies standards for single and multiple float buoyancy in fresh water. The standard does not apply to ditching/emergency flotation devices, but to amphibian rotorcraft devices.

(2) It is a design and a performance standard. Rigid or inflatable floats may be used. Enough water tight compartments (per Amendment 29-3) rather than a specific number are required to minimize the probability of capsizing when one compartment is flooded or deflated.

b. Procedures.

(1) Excess buoyancy. A minimum of 50 or 60 percent in excess of the maximum certificated weight of the rotorcraft is required for single or multiple floats, respectively. The weight of fresh water (density 62.42 pounds per cu. ft.) displaced by fully submerged float or floats (total volume of each float at operating pressure is used) should be a minimum of 50 or 60 percent greater than the maximum certificated weight of the helicopter.

(2) Capsizing.

(i) Each float should have enough sealed, separate and approximately equal volume compartments to minimize the probability of capsizing when the critical compartment is flooded or deflated. Five or more compartments in each float are usually necessary to meet the standard. Ten compartments per float have been employed in certain designs.

(ii) An analysis or test or combination thereof may be used, if necessary, to prove a positive margin of stability with the most "critical" compartment in one float flooded or deflated.

(iii) The location of the floats, and the most critical compartment, the helicopter weight, mass moment of inertia, and center of gravity location are also important considerations for capsize stability.

317. § 29.753 MAIN FLOAT DESIGN.

a. Explanation.

(1) Strength or design load standards are encompassed in the standard for inflatable bag and rigid floats. Bag pressure loads are included. The standard applies to an amphibious rotorcraft.

(2) The float landing loads are derived from the drop test of the float landing gear, or the load may be derived from tests of the wheel (or skid) landing gear (reference § 29.521). Bag type floats are not subject to the side loads according to the standard. Rigid floats, whether single or dual, are subject to the side load in each direction.

(3) Inflatable bag type floats should also be designed for the maximum pressure differential developed for the maximum operating altitude difference requested. That is, the resulting pressure difference between an operational altitude and a take-off site elevation should be established, and proven and may become an operating limitation.

(4) Landing loads suffice for the aerodynamic loads for typical rotorcraft float designs. Nonetheless, design and/or support of the forward part of bag type floats should be evaluated for maximum design speeds to prevent collapse or significant distortion of the bag while in flight.

(5) Resistance to puncture and abrasion at attach/wear points is not in the standard but is an important design consideration. "Girt" or attachment design loads shall be sufficient to withstand the loads imposed by the standards.

(6) The water or sea conditions (wave heights) evaluated in §§ 29.231 and 29.239 tests are not limitations but should be noted in the procedures section of the flight manual.

(7) The standard does not apply to ditching/emergency floatation devices.

* b. Procedures.

(1) Landing load factor.

(i) A drop test of the float landing gear may be conducted to obtain the limit landing load factor (reference § 29.725). Level landing attitude should be used for the float assembly.

(ii) The limit load factor for wheel or skid landing gear may be used (reference § 29.521) for the floats.

(iii) The float design ultimate load factor is 1.5 multiplied by the limit load factor.

(2) Flight aerodynamic loads--bag type floats.

(i) Evaluate collapse or significant distortion of bag type floats for speeds up to V_d (1.11 V_{NE}) with the minimum operating bag pressure.

(ii) External tubes to support the bag may be employed.
NOTE: Design landing loads may exceed the flight loads.

(3) Altitude differential loads.

(i) Bag type floats should not rupture due to the change in absolute pressure from take-off to the operating altitude. The applicant should select and prove the maximum operating altitude differential desired. A 5,000 to 8,000 feet operating differential may be a sufficient limitation. That is the helicopter with bags properly inflated could not operate more than 5,000 to 8,000 feet above the take-off site elevation. (See (3)(iii) for pressure relief values.)

(ii) A proof and ultimate pressure test should be conducted for the design. If operating or inflation pressure is 2.62 psi (including a tolerance) and 5,000 feet (pressure) differential is desired (use sea level to 5,000 feet pressures), the proof or limit pressure should be 2.62 + 2.47 = 5.09 psi. The pressure relief valves may be set at this value also. The change in size during inflation should be recorded. Significant changes may adversely affect flight characteristics and should be evaluated. The ultimate or burst free pressure should be proof pressure (5.09 psi) multiplied by 1.5 or 7.635 psi. A video or photographic record may be used as a reference of the change in size or shape for this test.

(iii) Each compartment should be equipped with a pressure relief valve to further protect the bag from excessive internal pressure .

(iv) At least one float should be subjected to a burst pressure test. Record the gauge pressure of burst.

(4) Landing loads.

(i) Rigid float vertical and a combined vertical and aft load conditions. A vertical or up-load only and a vertical combined with an aft load component for a resulting vector angle of 14.03 degrees from the vertical axis of the helicopter shall be used. Reference § 29.521(a). The resulting design load is the same load in both cases.

(ii) Rigid float side and vertical load condition. For each rigid float, whether single or dual, a vertical load combined with a side load resulting in a vector angle of 26.6 degrees from the vertical axis of the helicopter shall be used. The side load is applied to each float individually. Both inward and outward acting side load conditions shall be substantiated separately for the design of dual floats.

(iii) Load distribution on rigid floats shall be appropriate for the critical conditions. ANC-3 or § 25.533 and FAR Part 25, Appendix B may be useful.

(iv) Bag type float. The loads and the distribution of the loads are rather simple according to the standards. Only vertical loads and vertical with aft (drag) component are specified in the standard. These shall be distributed along the length over 75 percent of the projected area of the bag. Side loads are not required.

(5) Operating limitations.

(i) V_{NE} with floats installed is typically lower than the V_{NE} for wheel or skid landing gear helicopter configurations.

(ii) Bag inflation pressure shall be placarded or stenciled near inflation fittings.

(iii) The operating attitude differential proven for bag floats shall be an operating limitation. In addition, the flight manual should caution pilots about the effect of a significant decrease in altitude from the take off level which causes or reduces pressure in the bag. Placards may be employed as well.

(iv) Flight test results may dictate a further reduction in V_{NE} or changes in other operating limitations. *

* 318. § 29.755 (through Amendment 29-3) HULL BUOYANCY.

a. Explanation.

(1) This section contains performance standards for an integral fuselage hull and auxiliary (such as outrigger) floats. Water-based, amphibian and limited amphibian rotorcraft were encompassed in the standard.

(2) Amendment 29-3 added but Amendment 29-30 removed paragraph (b) which concerned Limited Amphibian Rotorcraft. Rotorcraft of that type used a "boat type hull" which is not desirable now and are certificated to the standards of § 29.801, Ditching, and § 29.563, Structural ditching provisions. (Limited amphibian rotorcraft were converted to the ditching configuration.)

(3) The worst combination of wave height and surface winds selected by the applicant shall be used in compliance with the standard.

b. Procedures.

(1) Capsizing.

(i) The hull and auxiliary floats shall have enough sealed compartments to allow failure of the critical, single, compartment in either the hull or auxiliary float and minimize the probability of capsizing.

(ii) Location of the most critical compartment (whether hull, sponson, or auxiliary), helicopter weight, mass moment of inertia, and c.g. location are also important considerations to prove stability or not capsizing.

(iii) The lightweight helicopter configuration and wind and wave condition should be considered, as well as the heavy weight configuration.

(iv) The sea state (worst combination of wave height and surface winds) is selected by the applicant. The condition proven is included in the procedures or information section of the flight manual. (It is not an operating limitation.)

(2) Buoyancy.

(i) Excess buoyancy is necessary to comply with the standard but the amount is dependent on several factors, such as number, size, and location of the sealed, watertight, compartments.

(ii) Wheel tires may be used for buoyancy if appropriate to the design.

(iii) Fuel tanks, if properly located and protected from potential rupture and if the aircraft has a system to rapidly empty the tanks, may be used also for buoyancy.

(iv) Buoyancy may be determined using the displacement of fresh water, with 62.42 pounds per cubic ft. density.

(3) Tests.

(i) If necessary, scale models may be used to prove the stability of the helicopter design for the sea state and wind conditions selected by the applicant.

(ii) The rotorcraft is subject to water tests per § 29.231. Compliance with part of this standard may be demonstrated or proven for the sea state or wave height, and wind conditions selected in conjunction with the TIA flight test program. This information is not an operating limitation.

(iii) Proposals should be submitted for evaluation. *

* 319. § 29.757 (through Amendment 29-3) HULL AND AUXILIARY FLOAT STRENGTH.

a. Explanation. The standard is an objective or performance strength standard. The water loads in § 29.519 shall be imposed for the hull and auxiliary floats in a conservative manner. The hull and float are "rigid" conventional amphibian or water-based aircraft structures.

b. Procedures.

(1) The water loads and conditions specified in § 29.519 shall be used. The pressures or load distributions should be appropriate to the design. ANC-3 and §§ 25.523 through 25.535 and Appendix B to FAR Part 25 may be of use.

(2) The water loads and applications of the loads are objective standards. A proposal and early discussions in the life of a project should be used to agree on an appropriate avenue or means of compliance. Tests or analysis supported by tests may be appropriate.

320.-326. RESERVED

329. <u>§ 29.775 (through Amendment 29-24) WINDSHIELD AND WINDOWS</u>.

a. <u>Explanation</u>. Nonsplintering safety glass is specified in windshields and windows containing glass to protect crew and passengers if window fracturing occurs. In any case, windshields and windows are to be made of transparent materials which will not break into dangerous fragments.

b. <u>Procedures</u>.

(1) Use nonsplintering safety glass in windshield or window applications which contain glass rather than plastic acrylics, polycarbonates, epoxys, etc. The glass selected should meet a specification such as MIL-G-25871, and if new vendors are selected by an airframe manufacturer, test data should be obtained from the vendor to demonstrate the safety glass provided meets an acceptable specification and provides adequate nonsplintering capability.

(2) Windshields and windows should be designed so that either--

(i) They are made of material which will not cause a serious reduction in the field of view by becoming suddenly opaque; or

(ii) Any one panel becoming opaque will not cause a serious reduction in the field of view (ref. § 29.773).

(3) In the event of any reasonably probable failure, a transparency heating system must be incapable of raising the temperature of any windshield or window to a point where there would be a danger of fire or structural failure (ref. § 29.1309).

* 330. § 29.777 COCKPIT CONTROLS.

a. Explanation. This section defines the general cockpit control requirements. Cockpit control location and arrangement, with respect to the pilot's seat, must be designed to accommodate pilots from 5'2" to 6'0" in height.

b. Procedures.

(1) The applicant should have a cockpit design report which documents the anthropometric suitability of the cockpit. Subsequent cockpit evaluations of control movement and location should be conducted with adjustable seats and/or controls positioned in a flight position for the subject pilot. Essential controls should be evaluated with the shoulder harness locked in the retracted position. Evaluation pilots should be aware of their individual anthropometric measurements and temper their assessments based on this information. Ideally, a new design should include evaluations by a range of different sized subject pilots. Control considerations for a second pilot position are the same as for the pilot station. Paragraph 327 discusses current philosophy concerning duplication of controls.

(2) As background, the following are examples of cockpit control issues which should be avoided:

(i) Collective control blocking the lateral movement of a pilot's leg, which in turn restricts the left lateral cyclic displacement.

(ii) Seat or seat cushion impeding the aft cyclic movement.

(iii) Inadequate space for large feet equipped with large flight boots.

(iv) Control/seat relationship which requires unusual pilot contortions at extreme control displacements.

(v) Control/seat relationship or control system geometry which will not permit adequate mechanical advantage with unboosted controls or in a boost OFF situation.

(vi) Addition of control panels or equipment to instrument panels or consoles which restrict full control throw.

(vii) Brake pedal geometry which results in inadvertent brake application upon displacement of the directional controls.

(viii) Controls for accessories or equipment which require a two-handed operation.

(ix) Emergency external cargo release controls which cannot be activated without releasing the primary flight controls.

(x) Essential controls which cannot be actuated during emergency conditions with the shoulder harness locked.

(xi) Throttle controls which can be inadvertently moved through idle to the cutoff position.

(xii) Switches, buttons, or other controls which can be inadvertently activated during routine cockpit activity including cockpit entry.

(xiii) Failure to account for operation with the pilot wearing bulky winter clothing.

(xiv) Aft cyclic movement limited by the pilot's body with a fore and aft adjustable seat in the full forward position. *

331. § 29.779 (through Amendment 29-24) MOTION AND EFFECT OF COCKPIT CONTROLS.

a. Explanation. The section standardizes motion and effect of cockpit controls. While this paragraph specifically addresses primary flight controls, engine power controls, and landing gear controls, it applies to all cockpit controls not addressed in other paragraphs.

b. Procedures.

(1) The cyclic should be mechanized such that movement of the control results in a corresponding sense of aircraft motion in the same axis. While a certain amount of coupling may be present following a pure control input in a given axis, that coupling should not be objectionable to the pilot. Collective pitch control should be mechanized such that an upward movement of the collective results in a corresponding relative motion of the aircraft in the vertical plane. Again, coupling should not be objectionable. Care should be taken to ensure that the primary pilot perception of collective motion is in the vertical plane. The objective is to clearly differentiate collective motion from that associated with an airplane throttle. The rule is self-explanatory on the subject of engine power controls. A distinction is made between normal landing gear controls and emergency controls. Emergency controls may operate in a sense which might not correspond to the direction of resultant gear motion.

(2) The recommended operating convention and "switchology" for miscellaneous controls is:

(i) Up/forward - on/increase.

(ii) Down/aft - off/decrease.

(iii) Variable rotary controls should move clockwise from the OFF position, through an increasing range, to the full ON position. For some variable intensity controls such as instrument lighting, the desired minimum setting may not be completely off. Pushbuttons not giving an obvious indication of mechanical position should be configured such that the flightcrew has a clear indication of switch actuation under both day and night (if applicable) conditions. Failure of the indication should be shown to be free of hazards.

(3) Slew or "beep" switches associated with flight control system applications warrant special attention. The recommended conventions for control-mounted single, or multifunction, two or four-way "beep" switches are:

(operating instructions will, of course, still be required). If one "EXIT" sign is used to both locate and mark the exit, it should be attached to the fuselage above the exit and not to the exit itself. If it is attached to the exit itself and the exit is discarded from the cabin after opening, the locating function of the exit sign is lost when the exit is removed. That is, there is no sign to locate the exit for passengers other than for the one who discarded the exit. The exit locating sign is a necessity to direct all occupants.

(2) Operating instructions should be provided as specified. They should be kept short but clear; e.g., "rotate handle," "push," "pull," etc.

(3) Lighting should be provided as specified to illuminate the cabin for egress paths and to supplement lighting of the exit operating instructions signs.

(4) The reflectance of external exit markings can be checked by appropriate electro-optical instrumentation or by use of photometer card sets. AC 20-47, Exterior Colored Band Around Exits on Transport Airplanes, provides information for complying with identical standards contained in § 25.811. These are also acceptable for § 29.811. The Munsell Color Company, 2441 North Calvert Street, Baltimore, Maryland 21218, provides a set of cards which includes shades of most commonly used colors.

343. <u>§ 29.812 (Amendment 29-24) EMERGENCY LIGHTING</u>.

a. <u>Explanation</u>. Section 29.812 was added by Amendment 24. This change unified the requirements for an emergency lighting system into a single paragraph and required these systems only for Category A helicopters. The purpose of this change was to afford passengers flying in Transport Category A rotorcraft the same level of safety in an emergency evacuation at night as passengers flying in transport category airplanes.

b. <u>Procedures</u>. This paragraph is quite similar to the emergency lighting system required for Part 25 airplanes. The exception is there are no requirements in this paragraph for floor proximity emergency escape path markings. The following items should be considered in the design of emergency lighting systems:

(1) There is a requirement for two controls of the system. One of these controls is located in the cabin, where it can be operated by a flight crew member or a passenger. The other control is located in the cockpit. These switches <u>must</u> have an "ON," "OFF," and "ARMED" position. These switches should operate independently of each other, and any other systems in the helicopter. The emergency lights must become lighted or remain lighted if the switch is either turned on, or the switch is in the armed position and there is an interruption of the helicopter electrical power supply. Inertia switches should not be used to satisfy this requirement.

(2) Sharing of light bulbs with the normal cabin lighting is acceptable provided there is sufficient isolation of the emergency lighting system from the normal cabin lighting circuits. No single failure of the shared portion should render the emergency lighting system inoperative.

(3) The luminosity tests of the emergency lighting system should be accomplished with the emergency exits open. *

344. § 29.813 (through Amendment 29-19) EMERGENCY EXIT ACCESS.

a. Explanation. Paragraph (a) of § 29.813 prescribes design details for passageways, both between passenger compartments and for access to Type I and II emergency exits, should they be provided. Such passageways are not made mandatory by § 29.813 although most larger rotorcraft have used them. Some utility or "wide-body" rotorcraft may have open areas between the crew area (pilots) and passenger area (cabin). These configurations may have lateral seating arrangements providing access to emergency exits of Type I or II size, even though they may not be required by § 29.807(b). These designs may not have a main aisle.

(1) Paragraph (c) of this standard concerns access to Type III and Type IV exits. Although "passageways" with explicit requirements are not required for Type III and Type IV exits, "access from each aisle to each Type III and Type IV exit" is required.

(2) For exits whose thresholds are more than 6 feet above the ground, additional space adjacent to the exit is required to allow room for a crewmember to assist passengers with the descent device such as an escape slide or rope noted in § 29.809(f).

(3) In addition to requiring passageways and crewmember space adjacent to exits over 6 feet above the ground, this standard does not allow obstructions in the projected opening of Type III or Type IV emergency exits for one seat width from the exit, except as noted. For passenger seating configurations of 19 or less, minor obstructions into the projection of the exit are allowed only if "compensating factors to maintain the effectiveness of the exit" are provided.

b. Procedures.

(1) The provision for unobstructed passageways, at least 20 inches wide as specified, is straightforward for medium or large cabins with a main aisle and a typical rectangular floor plan. Care should be taken to assure that seats (with lateral or fore-and-aft movement) or galleys (with doors or drawers) are not installed so that they can encroach upon the required passageway. Design features such as stops in seat tracks, seat back mechanisms, stops in galley door (or drawer) mechanisms may be required to assure that unobstructed passageways are provided.

(2) The requirement (added by Amendment 29-12) that "access from each aisle to each Type III and Type IV exit" be provided may add design features to the interior of many typical compact interiors of medium-size rotorcraft. Rotorcraft with emergency exits located in either hinged or sliding doors and having passenger area encroachment or protusions by compartments for fuel cells, gear boxes, etc., may require special design features to assure that passengers seated to one side or one area of the cabin have "access" to all Type III or Type IV exits on the same or other side of the rotorcraft. The cabin must not be separated into compartments or partitioned. For example, fold down seat back mechanisms may be required for compact cabin configurations having only lateral aisles rather than longitudinal aisles and having Type III or Type IV exits located on each side of the cabin at the end of the lateral seat row or rows.

discomfort or fatigue. In addition, the ventilation devices provided should not excessively increase the noise level in the cockpit. Compliance with the first requirement of § 29.831(a) can therefore be shown by an analysis showing the existence of at least 10 ft^3/m per crewmember, and a report from the flight test crew indicating that the amount actually provided is satisfactory.

(2) The passenger and crew compartment should be monitored under normal operating conditions for the presence of carbon monoxide. A carbon monoxide test kit is normally used for this evaluation. Air is monitored around outlets and different combinations of windows closed/open, heat off/on, air-conditioner off/on, etc., are checked to ensure all conditions are evaluated.

(3) When engine bleed air is used to condition the passenger and crew compartments' air, it should be initially substantiated that under normal operation, the amount of air being extracted does not exceed the limit established by the engine manufacturer. To accomplish this, determine the flight condition that will give the maximum bleed air flow through the flow limiter (venturi). The flow calculations should use this maximum flow condition and should also be made using the maximum tolerance diameter of the venturi throat.

(4) The engine bleed air should also be evaluated under malfunction conditions to determine a worst-case air contamination condition. (A typical worst-case malfunction is for an oil seal to fail in the engine that allows the engine oil supply to be introduced into the airflow.) With information regarding the contaminant, flow rate calculations can be made to predict the contamination levels that will be reached in the passenger and crew compartments and also the associated time duration of passenger and crew exposure. The severity of the exposure to the contaminated air is related to the temperature of the oil when it is introduced into the airflow. For example, synthetic base oils manufactured to MIL-L-7808 or MIL-L-23699 begin to break down into toxic components when the temperature exceeds 300 °C (572 °F). The oil manufacturers have evaluated this problem and should be in a position to provide data regarding the amount and type of toxic components to be expected, and the effect of introducing those components into the passenger and crew compartments. Therefore, from information supplied by the engine manufacturer, the worst-case air contamination condition can be calculated, and this can be compared with results of the oil manufacturers' tests to determine if the concentrations are harmful or hazardous.

347. § 29.833 HEATERS.

a. Explanation. This standard provides that each combustion heater must be approved. The standard contains no provisions regarding functioning of the system, environmental considerations, or malfunctions, therefore, the provisions of §§ 29.1301 and 29.1309 should be used to evaluate those aspects of an installation. The provisions of § 29.831, ventilation, should also be considered, as well as § 29.859, concerning combustion heater fire protection.

b. Procedures.

(1) Technical Standard Order TSO-C20 was issued June 15, 1949, and amended on April 16, 1951, and concerns Combustion Heaters. If a heater chosen for installation has been qualified to the provisions of TSO-C20, it is considered to be approved. If a unit is not qualified to TSO-C20, a qualification program for the heater itself should be established with FAA certification engineers participating in the program as early as possible. The program should be based on the provisions of the TSO.

(2) The TSO refers to the SAE Aeronautical Standard, AS 143B, which specifies certain additional devices, design features, air supply considerations, performance tests, safety controls, environmental considerations, and so forth. Consideration of all of the provisions of the aeronautical standard should result in an approved unit; however, it will not necessarily result in a satisfactory installation. For environmental considerations, it should be possible to specify an environmental spectrum more suitable to helicopters by referencing the latest version of Document No. RTCA/DO-160, Environmental Conditions and Test Procedures for Airborne Equipment, rather than AS 143B. Other specifications may also be satisfactory.

(3) The installation evaluation should consider functioning of the system based on the provisions of § 29.1301. Section 29.1309(a) is the regulatory basis for consideration of environmental conditions, and the expected environmental conditions resulting from the installation should be compared to those specified in the TSO. If the two are not compatible, additional environmental considerations are appropriate. The provisions of § 29.1309(b) should be used to evaluate the possible malfunctions of the installed system, and this evaluation should be documented in a fault analysis. The provisions of § 29.831 should be considered since certain standards of ventilation air quality under normal and malfunction conditions are specified. Additionally the provisions of § 29.859 should also be considered. *

348.-356. RESERVED.

SECTION 21. FIRE PROTECTION

357. § 29.851 (through Amendment 29-24) FIRE EXTINGUISHERS.

a. Explanation.

(1) The standard concerns objective performance criteria for both handheld fire extinguishers in the crew and passenger compartments and built-in fire extinguisher systems if the system is required.

(2) Section 29.853(e) and (f) dictate the quantity and general location of the handheld fire extinguishers.

(3) Section 29.855(d) contains standards for cargo/baggage compartments.

(4) Sections 29.1541 and 29.1561 concern durable and conspicuous markings and placards for location and operation or use of the equipment.

(5) The rotorcraft flight manual should contain appropriate information as well.

(6) Advisory Circular 20-42C, Handheld Fire Extinguishers for use in Aircraft, provides an acceptable means of compliance with the standard.

b. Procedures.

(1) Advisory Circular 20-42C provides valuable information to select the type and size of the handheld extinguishers.

(2) The type design data shall contain appropriate information. One location should be used (recommended) for the crew compartment. Several locations may be selected to allow for evaluation and approval of several extinguishers and their locations in the passenger compartment.

(3) During a compliance inspection of a complete interior, the installation of required and optional extinguishers shall be checked for compliance.

(4) Whenever an extinguisher is installed, even though not required by § 29.853(f), it shall also comply with the standards.

359. § 29.855 (through Amendment 29-19) CARGO AND BAGGAGE COMPARTMENTS.

a. Explanation. This section contains standards for accessible and inaccessible compartments. The rotorcraft should be able to contain a fire until it is detected and extinguished or until a safe landing and evacuation are accomplished. The cabin may be used as a cargo compartment for rotorcraft used for carriage of cargo only. Protective breathing equipment is required (§ 29.1439) for an appropriate crewmember or crewmembers when a compartment is accessible in flight. The rule does not provide for classification of cargo compartments. Reference is made to § 29.853 for flammability standards of certain materials.

(1) The compartment must be constructed of, or lined with, materials that are at least fire resistant. Accessible and inaccessible compartments must comply.

(2) Inaccessible compartments must be sealed and designed to completely contain a compartment fire or to allow detection as stated in § 29.855(c) and (d).

(3) Inaccessible compartments must have a detector unless the compartment can contain a fire as stated. Accessible compartments must have a detector or be designed to ensure detection by a crewmember while at his station as stated in paragraph (d). Flight evaluations assure that an inaccessible compartment is sealed and will contain smoke, gases, etc., as stated.

(4) The cabin area may be used for carriage of cargo only as stated in paragraph (e). Crew emergency exit must be accessible; sources of heat protected, and air flow must be stopped.

(5) Section 29.853 of Amendment 29-17 provides flammability standards for cargo compartment liners, covers, cargo, baggage tiedown equipment, etc., as stated in that section. This section pertains to compartments used by passengers or crew. Section 29.855(a) requires a fire resistant liner and is the overriding requirement.

b. Procedures. It is intended to provide for adequate protection of the crew and passengers in the event of an in-flight fire. For Category B rotorcraft, one objective as stated in § 29.861 is that the rotorcraft should be protected for at least 5 minutes (after recognition) in the event of a fire. The correct time interval to consider for Category A or B helicopters may be derived from the policy stated in paragraph 361, § 29.861.

(1) An aluminum inner skin, fire resistant liner, or closure of the compartment, whether the compartment is accessible or inaccessible is required by the rule. In the event of a compartment fire, the inner skin or liner will protect the load-carrying structure from direct flame impingement until the fire is detected and appropriate action is taken. Flight Standards Service Release No. 453 provides the standards for fire resistant materials.

(2) Inaccessible compartments, in addition to having the inner skin or liner, must be sealed to prevent entry of air and thereby contain a fire in the compartment. Flight tests are generally necessary to assure the compartment, primarily doors, do not leak in flight. Sensitive pressure measuring equipment (range of 10 inches of H_2O) may be used to prove the compartment is sealed by finding no appreciable change in compartment pressure during ground and flight conditions. The appropriate tests should also be conducted to determine that no accumulation of harmful quantities of smoke, flame, extinguishing agents, or other noxious gases occur in any crew or passenger compartment. For compartments having a volume not in excess of 500 cubic feet, an airflow of not more than 1,500 cubic feet per hour is considered acceptable. For larger compartments lesser airflow may be applicable to assure fires are contained.

(3) Inaccessible compartments may have a detector as prescribed. A smoke detector is preferable in place of a fire detector. The instrument panel will have an illuminated red indicator, such as baggage/cargo, as a warning signal for the flightcrew. Although no specific standards for the detectors are contained in FAR Part 29, the following standards are recommended. The detection system should be designed to provide a visual indication to the flightcrew within one minute after start of a fire or within 5 minutes after smoke initiation appropriate to the detector used (30 seconds is allowed under TSO C 1b, for smoke detector actuation). There should be a means to allow the crew to check in flight the functioning of each fire or smoke detector circuit. For large compartments, the effectiveness of the detection system should be proven and the detection system should be capable of detecting a fire at a temperature significantly below the temperature at which the structural integrity of the rotorcraft would be substantially decreased.

(4) Accessible compartments must have a detector or detectors unless a crewmember can detect a fire while at his station. Flight evaluations are necessary to assure accessible compartments may be isolated from crew and passenger compartments as stated. The rule envisaged separate compartments for passengers or crew and cargo/baggage.

(5) Insulation blankets, cargo covers, cargo and baggage tie-down equipment, including containers, bins and pallets used in accessible and inaccessible compartments should meet the flammability standards specified in § 29.853 for the same counterparts noted therein.

360. § 29.859 (through Amendment 29-2) COMBUSTION HEATER FIRE PROTECTION.

a. Explanation. This regulation ensures that onboard combustion heating systems (of all type designs) are safe during normal and survivable emergency operations. Thus as a minimum, each combustion heater design must meet the requirements of § 29.859.

b. Definitions.

(1) Backfire. An improperly timed detonation (or explosion) of a fuel mixture which results in higher than normal temperatures and pressures.

(2) Reverse flame propagation. An event that occurs when the flame from a controlled combustion process (such as a heater) goes in an abnormal path (i.e., either a reverse or different path than the intended path) as a result of a change in internal pressure or internal pressure gradient (e.g., a backfire) from a detonation or a similar event.

(3) Safe distance. A maximum flow length dimension determined from the thermodynamics of a worst-case flow reversal (backfire) and the local heater system geometry.

(4) Heater zone (or region). A geometric zone defined by the heater type, heater size, the location of heater system components, and the maximum safe distance determined under (3) above. The heater system components may affect the heater zone's size if they are closely located to the heat source. For example a heater fuel tank would not be part of the heater zone if it were located far away from the zone boundary; however, if it were adjacent or close to the boundary, it would be included in the heater zone.

(5) Fireproof. Fireproof is defined in § 1.1 "General Definitions."

(6) Severe Fire. The following thermodynamic definitions are based on AC 20-135, "Powerplant Installation and Propulsion System Component Fire Protection Test Methods, Standards and Criteria" and on the definitions in § 1.1 for fire resistant and fireproof materials. These definitions are provided for analytical purposes. A severe fire, when used with respect to fireproof materials, is one which reaches a steady state temperature of 2,000 ± 150 degrees Fahrenheit for at least 15 minutes. A severe fire, when used with respect to fire resistant materials, is one which reaches a steady state temperature of 2,000 ± 150 degrees Fahrenheit for at least 5 minutes.

(7) Hazardous accumulation of water or ice. An accumulation of water or ice that causes a device to not perform its intended function in either normal operation or a survivable emergency situation. *

c. Procedures. When suitable data is available, the heating system design should be thoroughly reviewed to determine which system components and arrangements must comply with each subsection of § 29.859. The method-of-compliance relative to each subsection of § 29.859 should then be determined. Acceptable, but not the only, methods of compliance are discussed on a section-by-section basis as follows.

(1) For compliance with § 29.859(a), combustion heater designs, their installations and their heater zones must be identified and thoroughly evaluated. The most direct method of compliance for the heater, itself, is to procure units that already have internal design features that meet the relevant requirements of this section; otherwise, design features should be provided and evaluated during certification that meet these same requirements. Several combustion heaters are FAA approved under TSO-C20. TSO-C20 provides the procurement sources and the detailed approval standards for these combustion heaters. Each heater, its installation and its heater zone should be reviewed against the criteria of §§ 29.1181 through 29.1191 and §§ 29.1195 through 29.1203 (reference paragraphs 584 through 589 and 592 through 596 of this AC) to ensure compliance. Next, the fire detector installation drawings and specifications should be reviewed for each heater region. The review should consider all reasonable hazards and failure modes of the heater and the detection system, itself. If not previously TSO approved, the detectors themselves should be evaluated and approved during the overall system certification effort. Then, the drainage and venting system for each heater installation should be reviewed to ensure that areas of fuel or fuel vapor collection are properly drained or vented. The capacity of each drain or vent should be determined and, unless impracticable, the flow capacity should be a minimum of 3-to-1 over the worst-case leakage anticipated (including the adverse effects of surface tension). Phased inspections to eliminate clogging should be considered. Finally, the drainage and ventilation systems should be reviewed to ensure that discharges do not create external hazards by entering or contacting external ignition sources such as engine inlets and hot exhausts. If an accurate determination cannot be made by a design review, ground and/or flight test work with dyed, inert fluids or vapors should be conducted to accurately display discharge patterns.

(2) For compliance with § 29.859(f), the ventilating air duct design should be reviewed to determine what ducts are routed through heater zones. Once this has been determined, each duct section running through the heater zone should be made fireproof by either using a fireproof shroud around the existing duct or by using fireproof material for the duct wall. A primary purpose of these certification measures is to eliminate any system leakage that would allow carbon monoxide (a poisonous gas) to enter occupied areas, incapacitate the crew or passengers, and cause a crash. Regardless of the method-of-compliance chosen, periodic checks should be performed during certification using carbon monoxide detection equipment to certify the leak-free integrity of the system. Several such checks should be done during flight test, especially after rigorous maneuvers, to ensure no leakage. It is also recommended that periodic checks using a carbon monoxide detector be conducted in conjunction with phased visual inspections (typically at a less frequent interval than each visual inspection) to ensure continued airworthiness. Carbon monoxide tests are reliable and quickly accomplished without any system disassembly. Continued airworthiness considerations are very important since carbon monoxide is a colorless, odorless, tasteless, poisonous gas that incapacitates an occupant without warning. Carbon monoxide's ability to incapacitate increases with altitude, and has long been suspected as a probable cause for many aircraft accidents. It is the subject of General Aviation Airworthiness Alert No. 137, dated December 1983.

(3) For compliance with § 29.859(c), any design using combustion air ducts should be reviewed to ensure that the ducts are either made from fireproof material or shrouded with a fireproof shroud over a safe distance (see definition). The safe distance should be determined analytically, by test, or a combination, if the analytical results are not conclusive. The design should be reviewed to ensure that combustion air ducts are not connected to the ventilating airstream, except when an informal equivalent safety finding can be made that shows backfires or reverse burning cannot induce flames or fumes into the ventilating airstream under any failure condition or malfunction of the heater or its associated components. Such a finding should require analysis, testing, or a combination for a proper determination. A hazard FMEA should be conducted to ensure that no flames or fumes can be induced under any failure mode.

(4) For compliance with § 29.859(d), the design and installation of all standard heater control components, control tubing and safety controls should be reviewed to determine the probable points of water or ice accumulation (e.g., sumps, rough surfaces, joints, etc.). If a design review cannot accurately determine these accumulation points, then bench tests and flight tests should be conducted for proper determination. Once these points are identified, the ability of the effected part (or parts) to perform its intended function when water or ice has fully accumulated must be determined for both normal and survivable emergency operations. If the part (or parts) either has not lost its ability to function; has lost part of its ability to function; or has lost all of its ability to function; and the entire system's function is not impaired, then nothing further should be required. However, if the overall system's function is hazardously impaired or lost, as a result of water or ice accumulation on a part (or parts), then rectifying design improvements should be made prior to final approval. These improvements should either alter the part's environment (e.g., relocation, enclosure, insulation, etc.) or eliminate the hazardous accumulation of water or ice (e.g., provide drainage, better sealing, better location, different surface finish, etc.).

(5) For compliance with § 29.859(e), combustion heaters, if used, must have separate, independent safety controls from their standard controls (e.g., air temperature, air flow, fuel flow, etc.) which are remotely located in case of a heater fire, are operable by the crew and automatically shut off the ignition and fuel supply when a hazardous condition exists, (as defined by § 29.859(g)). These separate safety controls must comply with § 29.859(g)(1), must keep the heater off until restarted by the crew or ground maintenance, and must warn the crew when an essential heater is automatically shut down. The safety control system design should be thoroughly reviewed and tested to ensure that it complies and that no hazardous failure modes exist. An FMEA should be conducted to ensure proper compliance.

(6) For compliance with § 29.859(f), each combustion and ventilating air intake's location should be identified, reviewed, and tested to ensure that no flammable fluids or vapors can enter the heater system, ignite and create a fire. If a combustion or ventilating air intake's location is critical or questionable, it should be relocated, shielded, drained, or other equivalent means provided to eliminate the potential fire hazard. If engineering analysis and evaluation are not adequate to make an acceptable safety finding, testing using dyed, inert, leaked fluids or vapors should be conducted. *

(7) For compliance with § 29.859(g), each heater exhaust system design should be reviewed, tested, or a combination to ensure proper compliance with § 29.1121 and § 29.1123 (reference AC paragraphs 548 and 549, respectively). Each exhaust shroud should be sealed to ensure that leaked flammable fluids or vapors do not contact the hot exhaust and cause a fire. The seal design should be reviewed to ensure that the sealing material is fireproof, is chemically compatible with the relevant fuels and vapors, is durable and is functionally adequate. If the design review is not conclusive for compliance purposes, then the seal system should be bench tested under pressure while undergoing critical service loads and motions to ensure no leakage occurs. Phased seal inspections should be considered to ensure continued airworthiness. An analysis should be conducted to determine the structural effects on the exhaust system of the worse case restricted backfire (typically a shock wave analysis can be used to determine the peak internal pressure and the resultant load on the exhaust system.) If structural failure would occur, based on the analysis, either the backfire restriction should be reduced or the exhaust design should be structurally improved to eliminate the failure.

(8) For compliance with § 29.859(h), each heater's fuel system design must be reviewed to ensure compliance with the powerplant fuel system requirements of Part 29 that are necessary for safe operation to be achieved. An equivalent safety finding should be made if an application is received that requests partial compliance or non-compliance with the powerplant fuel system requirements of Part 29. The finding should ensure that the safety intent of § 29.859(j) is achieved. Analysis, engineering evaluation, testing, or a combination should be used to substantiate the heater fuel system design. Heater fuel system components that, by leakage or other failures, can induce flammable fluids or vapors into the ventilating air stream should be shrouded by drainable, fireproof shrouds.

(9) For compliance with § 29.859(i), the drain system design should be reviewed to identify parts that may be subjected to high temperature and parts that may be subjected to hazardous ice accumulation in service. The high temperature parts should be evaluated using the methods of compliance for heater exhausts (reference paragraph c(7), above and paragraph 549 of this AC). Drains that would be stopped up from ice accumulation should be protected by relocation, size, shields, heating, or a combination to ensure hazardous fluids and vapors are properly drained away. *

(v) Assure fluid lines are not located so as to be subject to abrasion during normal operations. Cargo compartments should be evaluated for potential line damage due to cargo movement.

(2) To minimize the hazards if ignition occurs:

(i) Provide fireproof designs, fire wall isolation, or equivalent means for critical structure, equipment and personnel areas.

(ii) Consider fire detection, extinguishment, shutoff valves, fire suppression systems, etc.

(3) In considering compliance the actual protective measures may be related to the situation, considering the quantity and flammability characteristics of the fluid, the fire damage tolerance of the area, and the means available to the crew to minimize hazards from the fire. If action by the crew is necessary, quick-acting means (not necessarily fire detectors) must be provided to alert the crew in the event of a fire. Details of any action required by the crew should be included in the Rotorcraft Flight Manual.

(4) Compliance with § 29.863(d) requires as a minimum, type design data defining each area where flammable fluids or vapors might escape.

363.-372. <u>RESERVED</u>.

SECTION 22. EXTERNAL LOAD ATTACHING MEANS.

373. § 29.865 (through Amendment 29-12) EXTERNAL LOAD ATTACHING MEANS.

a. Background. The external load attaching means standards for transport and normal category rotorcraft were originally contained in Subpart D, "Airworthiness Requirements of FAR Part 133, Rotorcraft External-Load Operations." Amendment 29-12, in 1977, added a new § 29.865, which moved these standards from Part 133 to Part 29. An identical transfer occurred in 1977 for Part 27. Transport Category A and B rotorcraft were initially used under Part 133 operations and, after Amendment 133-6, restricted category rotorcraft were also included under Part 133 operations. The use of restricted category first came about when an operator, exempt from Part 133, transferred harbor pilots to and from ships by a hoist and sling. The exemption was granted to study the feasibility of passenger transfer outside of the cabin. Subsequently, Amendment 133-9, adopted in January 1987, established a new Class D rotorcraft load combination for transporting passengers external to the rotorcraft. Amendment 133-9 also provided for the limitations and conditions for external passenger transportation and the necessary, associated safety requirements. Part 29 rules have not yet been changed to reflect the Class D requirements.

b. Explanation. While the regulation only addresses external load attaching means, this advisory material also includes guidance for certification of external load carrying devices for helicopters to be used in conjunction with Part 133, "Rotorcraft External Load Operations." Subpart D of Part 133 contains supplemental airworthiness requirements. Part 1 defines four classes of rotorcraft load combinations which are operationally approvable under the Part 133 operating rules and, thus, are eligible for certification under § 29.865. Parts 1 and 133 (through Amendment 133-9) contain a new rotorcraft load combination, Class D, that addresses personnel carried externally. The four classes of rotorcraft load combinations are summarized in Table 373-1 and are discussed in detail in paragraph c. For further information, AC 133-1A, "Rotorcraft External-Load Operations in Accordance with FAR Part 133," October 16, 1979, may be reviewed. Also, paragraph 43 of this AC (ref. § 29.25) concerns, in part, jettisonable external cargo.

c. Procedures.

(1) The applicant should clearly identify the Parts 1 and 133 rotorcraft load combination classes (A, B, C, or D) that are being applied for. The loads and operating envelopes for each class should be determined and used to formulate the flight manual supplement and basic loads report. The applicant should show by analysis, test, or both, that the rotorcraft structure, the external load attachment means, and (for Class D operations) the personnel carrying device meets the requirements of §§ 29.865(a), 133.41, 133.43, and 133.45(e)(3) for the proposed operating envelope. *

(2) For rotorcraft load combination classes A, B, and C, § 29.865 requires use of 2.5 g vertical limit load factor (N_{ZW}) at the maximum substantiable cargo load (which is typical for cargo hauling configurations). This 2.5 g limit load factor is based on an engineering evaluation and a rationalization of § 29.337 for high gross weight applications. However, for lower gross weight configurations (which are more typical of a Class D application; i.e., personnel transport or evacuation), a higher limit load factor is recommended to ensure that limit load is never exceeded in service. For example, a Class D external load carrying device which is certified to a limit vertical load factor of 2.5 g and is installed in a minimum gross weight configuration rotorcraft capable of generating a vertical limit load factor of 3.2 g's could experience ((3.2/2.5 x 1.5) x 100) = 85 percent of ultimate load under emergency conditions with new external hardware. However, if factors such as wear and corrosion have effected the structural integrity of the external hardware ultimate load could be exceeded in emergency service. In any case, FAA policy is to not exceed limit load in service. The higher load factor for Class D cases should be the analytically derived maximum vertical limit load factor for the restricted operating envelope being applied for; or, as a conservative option, a vertical limit load factor of 3.5 g's (reference § 29.337). Unless a more rational proposal is received, for Class D cases where maximum operating gross weight for external load is between design maximum weight and design minimum weight, linear interpolation can be used between $N_{ZW\ MIN}$ and $N_{ZW\ MAX}$ versus gross weight for design limit load factor determination.

(3) For applications that employ winches (or hoists) to raise or lower an external load from a hover (or another phase of flight), limit load must be properly determined based on the characteristics of the winch system and its installation such as mechanical advantage, static strength of the winch, static strength of its installation and the payload for any operating scenario being applied for. One acceptable method of determining limit load is by the following procedure:

(i) Determine the basic loads that fail and unspool the winch or its installation, respectively (Note: This determination should be based primarily on static strength; however, any dynamic load magnification factors that are significant should be accounted for).

(ii) Select the lower of the two values from (i) as the ultimate load of the winch system installation.

(iii) Divide the selected ultimate load by 1.5 to determine the limit load of the system.

(iv) Compare the system's derived limit load to the applied for one "g" payload multiplied by the maximum downward vertical load factor (N_{ZWMAX}) from paragraph (2) to determine the critical payload's limit value.

(v) If the critical limit payload is equal to or less than the system's derived limit load the installation is structurally approvable as presented.

(vi) If the critical limit payload exceeds the system's derived limit load then one of the following options should be considered:

(A) Disapproval.

(B) Application for exemption.

(C) Reduction of the applied for critical limit payload to less than or equal to the system's derived limit load.

(D) Redesign of the winch system (and installation) to increase its derived limit load to equal to or greater than the critical payload.

(E) A combination of options (C) and (D).

(F) Approvable operating restrictions to reduce N_{ZWMAX} and, the corresponding critical limit payload to less than or equal to the system's derived limit load.

(4) In all approved cases, appropriate winch system placards and flight manual restrictions should be provided. Also, for class D load combinations, the winch or hoist should have a demonstrated, acceptable level of reliability (for the phases of flight in which it is operable and in which the Class D load is carried externally). The winch should be disabled (or utilize an overriding mechanical safety device such as a flagged removable shear pin) to prevent inadvertent load unspooling or release during the phases of flight that the load is carried externally and operation is not intended. The maximum allowable winch cable angle should be determined and approved. This is primarily a structural requirement but should also be reviewed from an interference and flight handling criteria standpoint. *

(5) It is recommended that winch or hoist systems be demonstrated as follows:

(i) At least 1/3 of the demonstration cycles should include the maximum aft angular displacement of the load from the drum applied for under § 29.865(a).

(ii) The load versus speed combinations of the winch should be demonstrated by showing repeatability of the no load-speed combination, the 50 percent load-speed combination, the 75 percent load-speed combination and the system limit load-speed combination.

(iii) A minimum of six consecutive, complete operation cycles should be conducted at the system's critical limit load speed combination.

(iv) In addition, the demonstration should cover all normal and emergency modes of intended operation and should include operation of all control devices, limit switches braking devices, and overload sensors in the system.

(v) Quick disconnect devices, and cable cutters should be demonstrated at 25 percent, 50 percent, 75 percent, and 100 percent of system limit load. Any electrical load release devices for Class D loads should be treated as a novel design feature and should be coordinated with the Rotorcraft Directorate.

(vi) Any devices or methods used to increase the mechanical advantage of the winch should also be demonstrated.

(vii) During each demonstration cycle, the winch should be operated from each station from which it can be controlled.

(viii) Operating manuals, flight manuals, and associated placards should be used and proofed during the demonstration. *

TABLE 373-1: SUMMARY OF PART 133 ROTORCRAFT LOAD COMBINATIONS CERTIFIABLE UNDER § 29.865

Basic Definition and Intended Use	Typical Load Limits	Quick Release Requirements	Certification Requirements -- Considerations
Class A			
<u>Fixed External Cargo Container</u> - Is defined by § 1.1 as a load combination in which the external load cannot move freely, cannot be jettisoned, and does not extend below the landing gear. This category usually features multiple attachments (loadpaths) to the airframe. Typical example is a hard mounted cargo basket attached to the rotorcraft crosstubes which is used to carry cargo from point A to point B.	Certification limit is N_{ZW} X Maximum Substantiable External Load. N_{ZW} is 2.5 per § 29.865 (See Procedure, paragraph (2)).	None. Cargo and its container are not jettisonable.	o For cargo only. o Flight Manual Restrictions - § 133.47 requires a rotorcraft load combination flight manual supplement. Any flight envelope restrictions of § 29.865 should be a part of this supplement. o Load limit placards are required by § 29.865(c). o Flight envelope restriction placards may also be required for gross weight limitations, e.g., limitations, elimination of dangerous maneuvers, etc. o Cargo tiedowns to prevent load shifting relative to airframe may be required. o Effect of external cargo carrier and its maximum cargo weight on load paths, loads and fatigue of existing structure should be determined. o TIA testing may be necessary to determine whether or not the system performs as intended and if placards and flight manual supplements are adequate. o The applicant may elect to test the aerodynamic effect of several representative load shapes and include applicable information in the flight manual supplement. If such information is not in the RFM, then the operator may be required to obtain an operations approval under Part 133.

Basic Definition and Intended Use	Typical Load Limits	Quick Release Requirements	Certification Requirements -- Considerations
Class B			
<u>Single Point Suspension External Load Airborne</u> Is defined by § 1.1 as a load combination in which the external load is jettisonable and is lifted free of land or water during the rotorcraft operation. The payload is typically suspended from a hook or a similar device. The hook may be attached to the rotorcraft structure or it may be attached to a movable hoist cable and the hoist itself attached to the rotorcraft. Typical use is to lift a cargo load until it is completely airborne and fly it from point A to point B. The load on the hoist may be stowed in the fuselage (in some cases) while being transported.	Certification limit load is N_{ZW} X Maximum Substantiable External load. N_{ZW} is 2.5 per § 29.865 (See Procedure, paragraph (2)). Load may be limited by hoist allowables (reference paragraph (3)).	Yes - § 29.865(b)(1) requires that a primary quick release system control device be installed on a primary control. Also, a manual quick release system backup actuation device must be available and readily accessible.	o For cargo only. o Flight Manual Restrictions - § 133.47 requires a rotorcraft load combination flight manual supplement. Any flight envelope restrictions from § 29.865 should be a part of this supplement. o Load limit placards are required by § 29.865(c). o Flight envelope restriction placards may also be required. o Certifiable external cargo load capacity may be further limited by §§ 133.41 and 133.43. o Quick release devices must be approved and be operable on a nonhazard basis by the pilot per § 29.865(b). o Manual backup must be reliable but need not be overly sophisticated (cable cutters, axes, etc., used by crew members). o Effect of maximum suspended load and its attachment to rotorcraft structure on load paths, loads and fatigue of existing structure should be determined. o TIA testing may be necessary to determine whether or not the system performs as intended and if placards and flight manual supplements are adequate.

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Basic Definition and Intended Use	Typical Load Limits	Quick Release Requirements	Certification Requirements -- Considerations
Class C			
<u>Single Point Suspension External Load Partially Airborne</u> - Is defined by § 1.1 as a load combination in which the external load is jettisonable and remains in contact with land or water during the rotorcraft operation. The payload is typically partially suspended by a net or cables from a cargo hook or a similar device. The cargo hook may be attached to the rotorcraft structure or may be attached to a movable hoist cable and the hoist itself attached to the rotorcraft. Typically used for stringing wire or laying cable where the payload is only partially suspended from the ground. (Note: Many applications combine both Category B and C operations because of the obvious utility involved.)	Certification limit load is N_{ZW} X Maximum Substantiable External load. N_{ZW} is 2.5 per § 29.865 (See Procedure, paragraph (2)). Load may be limited by hoist allowables (reference paragraph (3)).	Yes - § 29.865(b)(1) requires that a primary quick release system control device be installed on a primary control. Also, a manual quick release system backup actuation device must be available and readily accessible.	o For cargo only. o Flight Manual Restrictions - § 133.47 requires a rotorcraft load combination flight manual supplement. Any flight envelope restrictions from § 29.865 should be a part of this supplement. o Load limit placards are required by § 29.865(c). o Flight envelope restriction placards may also be required. o Certifiable external cargo load capacity may be further limited by §§ 133.41 and 133.43. o Quick release devices must be approved and be operable on a nonhazard basis by the pilot per § 29.865(b). o Manual backup must be reliable but need not be overly sophisticated (cable cutters, axes, etc., used by a crew member). o Effect of maximum suspended load and its attachment to rotorcraft structure on load paths, loads and fatigue of existing structure should be determined. o TIA testing may be necessary to determine whether or not the system performs as intended and if placards and flight manual supplements are adequate.

Basic Definition and Intended Use	Typical Load Limits	Quick Release Requirements	Certification Requirements -- Considerations
Class D			
<u>Single Point Suspension External Airborne Personnel Load</u>. Is defined by § 1.1, as a load combination in which the external load is other than Class A, B, or C and has been specifically approved by the Administrator for that operation. This load combination includes human cargo. For human cargo operations, the payload which typically consists of personnel and their containment device is suspended from a hook or a similar device during all or part of a flight. The hook may be rigidly attached to the rotorcraft or may be attached to a movable hoist cable and the hoist itself rigidly attached to the rotorcraft. Typical use is for transfer of personnel to a ship.	Certification limit load is N_{ZW} X Maximum Substantiable External load. N_{ZW} varies from 2.5 at max gross weight to 3.5 at minimum gross weight. (See Procedures (2)). Load is usually limited by hoist allowable or by personnel carrying device allowable (See Procedure (2), (3), and (10)).	Section 29.865(b) does not currently contain quick release requirements for Class D rotorcraft - load combinations, but § 133.45(e)(4) requires that a primary emergency release system control device (requiring two distinct actions) be installed on a primary control or be installed near a designated crew member's station. Also, a manual quick-release system backup actuation device must be available and readily accessible.	o For loads other than class A, B, or C loads. Is used for external personnel loads. o § 29.865 has not been revised to reflect this category's requirements (it is currently covered by § 133.45(e)(4) only. o Unless a public-use rotorcraft is being certified, only transport Category A rotorcraft are eligible to use this load category. o Transport Category A rotorcraft must be certified for an OEI weight and altitude envelope which becomes the maximum envelope that can be used for Class D operations. This is currently required for a Class D rating by § 133.45(e)(1). o Personnel lifting devices must be approved separately or as part of the certification project. o Devices must carry personnel internally or secure them safely in a harness or equivalent device. o Flight Manual Restrictions - § 133.47 requires a rotorcraft load combination flight manual supplement. Any flight envelope restrictions from § 29.865 should be a part of this supplement. o Load limit placards are required by §29.865(c). o Flight envelope restriction placards may also be required. o Certifiable external load capacity is further limited by §§ 133.41, 133.43 and 133.45(e)(3), the load limit of the personnel carrying device.

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Basic Definition and Intended Use	Typical Load Limits	Quick Release Requirements	Certification Requirements – Considerations
Class D (continued)			
Carrying devices may transport one or more persons. Typical carrying devices are vest and straps, baskets, life preservers with straps and attachment devices, cages, or a suspended container.			o Quick release devices must be approved and be operable on a nonhazard basis by the pilot or a designated crew member per §§ 133.44(c)(6) and 29.865(b). o The lifting device must have an emergency release requiring two distinct actions § 133.45(e)(4). o Manual backup must be accessible and reliable. o Rotorcraft must be equipped to allow direct intercom among all crew members per § 133.45(e)(2). This may affect § 29.865 indirectly if human error or placarding could cause inadvertent load release or retention. o Effect of maximum external load and its attachment to rotorcraft structure on load paths, loads and fatigue of existing structure should be determined. o TIA testing may be necessary to determine whether or not the system performs as intended and if placards and flight manual supplements are adequate.

(6) For all applications, it is good practice to obtain the gross weight range limits, the corresponding limit load factors (N_{ZW}), and substantiate the system, accordingly, for the critical loads. This procedure determines the critical basic loads and associated operating envelope for the rotorcraft load combination categories requested.

(7) For a request involving more than one class of rotorcraft load combinations, structural substantiation is required only for the critical case if accurately determinable from analysis.

(8) Appropriate placards, markings, and flight manual restrictions should be provided as determined by load capacities and operational restrictions. Each placard, marking, and flight manual supplement should be checked during TIA flight testing.

(9) For load Classes A, B, C, and D, the basic vertical limit load factor (N_{ZW}) from (c)(2) is converted to ultimate by multiplying the maximum applied load (i.e., the sum of the carrying device load and cargo or personnel loads) by 1.5. (For restricted category approvals, see guidance in Paragraph 785.) This load is used to substantiate all existing structure affected and all added structure associated with the external load carrying device and its attachments. Casting and/or fitting factors are to be applied where appropriate. For load Class D, the weight of each occupant carried externally should be assumed, for analysis purposes, to be that of the 95 percentile (202 pound) man (reference MIL-STD-1472, "Human Engineering Design Criteria for Military Systems, Equipment and Facilities).

(10) For load Classes B, C, and D, the maximum limit external load for which certification is requested, even though it may otherwise be much less than the maximum system capacity; e.g., cargo hook capacity, etc., should not exceed the rated capacity of the quick release device used in the applicant's proposed design or, for Class D only, the rated capacity of the personnel carrying device. The quick release and personnel carrying devices should be strength tested (with FAA witness) or otherwise structurally substantiated to determine their allowable limit load capacity, if it has not been previously approved or was not produced to a recognized and approvable industry or military standard. *

* (11) For load Classes B, C, and D, in substantiating analyses and tests, the maximum ultimate external load is specified to be applied at the sling-load-line to rotorcraft vertical axis (Z axis) angles up to 30 degrees, except for the forward direction. The 30 degree angle may be reduced, if impossible to obtain due to physical constraints, or operating limitations. If the angle is reduced, appropriate placards and flight manual changes are required.

(12) For load Classes B and C, an external releasing system is mandated which requires an approved primary quick release device to be installed on one of the pilot's primary controls. The quick release device (typically installed on the cyclic stick) is designed and located to allow the pilot to accomplish load release without hazardously limiting his ability to control the rotorcraft during emergency situations. A manual (backup) mechanical quick release device is also required. This control must also be readily accessible to the pilot or another designated crew member, such as a hoist operator. For Class B and C cargo applications, a sufficient amount of slack should be provided in the control cable to permit cargo hook movement without tripping the hook release.

(13) For Load Class D, an emergency release system is specified by § 133.45(e)(4) which requires two distinct actions for load release. This is intended for the phases of flight that the load is carried (and/or retrieved) externally. This release can be operated by the pilot from a primary control or, after a command is given by the pilot, by a dedicated crew member from a remote location. Two distinct actions are required for the primary release to provide a higher level of safety for Class D human external loads. If the manual backup device is a cable cutter, it should be properly secured but readily accessible to the dedicated crew member intended to use them.

(14) For Class D (human) load applications, to ensure personnel safety, the emergency release system design and associated placarding should be given special consideration. As stated previously, electrical release designs should be reviewed by the Rotorcraft Directorate prior to approval.

(15) For the majority of Class D applications, an approved single or multiple personnel carrier or container is required. The carrier or container may be previously approved or may be approved as part of the certification process. In any case, the single or multiple personnel carrier or container should be substantiated for the allowable ultimate load as determined under paragraphs c(2), (3), (4), (5), (6), (7), (8), and (9) above. The personnel carrier or container should be placarded for this capacity and show the proper internal arrangement and/or location of the intended occupants. Some exceptions may exist that are certifiable under class D that involve the technique of "Rappelling" from a rotorcraft. Rotorcraft load-combination D allows for such applications by definition (reference § 1.1). Other types of human cargo devices can be applied for under the Class D external load combination definition. An example is external carriage of personnel in a conveyance rigidly attached to the rotorcraft (e.g., cage, pod, secured litter or strap harness/seat arrangement).

(16) The personnel carrier or container should be easily and readily ingressed or egressed. Appropriate placards are required to provide ingress and egress instructions. For door latch fail-safety, more than two fastener or closure devices are recommended. Direct visual inspectability of the latch device by both crew and passengers is recommended to ensure it is fastened and secured. Any fabric, if used, should be durable and should meet the flammability standards of safety belts as stated in TSO C-22. Sharp corners and edges should be avoided, and padding should be used when necessary to protect the carrier and container occupants.

(17) The U.S. Coast Guard has three containers or devices that are used with helicopters for emergency rescue work. These devices and their National Stock Numbers are listed below. These devices have not been FAA approved; however, applications which involve them may be submitted for approval.

National Stock No.	Title
6530-00-042-6131	Stokes litter (one person)
1670-00-HRO-7970	Rescue basket
1680-090-511-2712	Rescue sling (one person)

NOTE: The rescue sling is a "collar" device that requires a person to exert some effort to remain in the collar. This sling should only be used in conjunction with properly written instructions and with personnel trained in the proper use of the sling. *

(18) Flight test verification work that thoroughly checks out the operational envelope should be accomplished with every device approved for external cargo carriage (especially rotorcraft load combination D which includes external human cargo). The flight test program should show that all aspects of the applied for operations are safe, uncomplicated and can be conducted by an average flight crew under the most critical service environment and, in the case of human external cargo, under the pressures of an emergency scenario.

374.-383. RESERVED.

448. § 29.953 (through Amendment 29-24) FUEL SYSTEM INDEPENDENCE.

a. Explanation.

(1) Section 29.953(a)(1) stipulates that fuel systems for Category A rotorcraft must meet the requirements of § 29.903(b) engine isolation.

(2) Section 29.953(a)(2) specifies independent fuel feed systems for each engine for Category A rotorcraft unless other provisions are made to meet the § 29.903(b) engine isolation requirement.

(3) Section 29.953(b) specifies independent fuel feed systems for each engine for Category B rotorcraft, except that separate fuel tanks are not required.

b. Procedure.

(1) The purpose of § 29.953 is to ensure an independent fuel supply system for each engine. Multiengine Category B rotorcraft do not require separate fuel tanks, as are intended for Category A.

(2) The assessment to ensure compliance with § 29.903(b), engine isolation, should include consideration of component failure, malfunction, and damage. For multiengine Category B helicopters, leakage of the fuel cell could be excluded from consideration since § 29.953(b) explicitly states that separate fuel tanks are not required for this category rotorcraft.

NOTE: Of interest is that § 23.903(c), engine isolation for normal category airplanes, also excludes the fuel tank from consideration if only one tank is used.

(3) Consideration of fuel tank leakage under § 29.903(b) has dictated separate fuel tanks for Category A rotorcraft, but the regulation leaves the door open for unique designs by the expression, "Unless other provisions are made...," in § 29.953(a)(2). Separate tanks are intended for Category A as evidenced by the identical fuel system independence requirements for multiengine Category B helicopters, except that separate tanks are specifically not required.

(4) A common supply tank, with individual "collector" tanks for each engine for Category A rotorcraft, has been allowed under § 29.953 provided that the capacity of the collector tanks will allow 20 minutes of maximum allowable en route OEI power.

(5) The fuel system independence regulations are not intended to preclude single-point fueling designs.

(i) For multiengine Category B rotorcraft, the assessment of an independent fuel supply system for each engine would begin at the fuel supply pickup point within the tank and continue to the engine fuel inlet at the engine.

(ii) For Category A rotorcraft, the assessment would begin with the tanks and continue to the engine fuel inlet.

(6) If supply line crossfeed capability is included as a feature, care must be exercised to ensure that the opening of the crossfeed does not jeopardize the continued safe operation of more than one engine. For example, if the crossfeed valve is automatically operated by a low pressure signal in the supply line for one engine, the possibility that fuel line leakage could cause opening of the crossfeed and jeopardize the continued safe operation of both engines should be considered. Similarly, opening the crossfeed valve with a suction lift system following engine or system malfunction should not allow air into the fuel supply line of the remaining engine.

(7) The independent fuel supply system requirement for each engine is for normal fuel system operations. Care should be exercised to ensure that flight manual procedures do not authorize normal usage of fuel system configurations which may violate the engine isolation principle. For example, routine fuel balance procedures should not allow usage of a common supply line if a failure can jeopardize the continued safe operation of more than one engine.

(8) Fuel system designs which allow the continued safe operation of all engines under expected fuel system component failure conditions (for example, a failed boost pump) by using common fuel flow paths under failure conditions are not prohibited.

(9) For APU's which perform a required in-flight function, a separate, independent fuel system complying with the corresponding engine fuel system rules should be provided. Other APU's (which do not perform a required in-flight function) may be supplied with fuel from a tee connection to a main engine fuel supply. The fuel shutoff valve for the APU should be located as close as possible to the APU system's connection to the main engine fuel system and a checkvalve should be included in the APU fuel system to prevent reverse-flow if negative pressure exists momentarily in the main engine fuel system. Maximum fuel demand of the APU will not jeopardize compliance with § 29.955.

449. § 29.954 (through Amendment 29-26) FUEL SYSTEM LIGHTNING PROTECTION.

a. Background. During the initial development and promulgation of the standards concerning the airworthiness of rotorcraft, it was not deemed necessary to specify design features that would protect the rotorcraft from the meteorological phenomenon of lightning. This was due, in part, to the fact that rotorcraft were primarily operated in a VFR and non-icing environment. Also, a prudent pilot avoided thunderstorms where the possibility of encountering severe weather and a lightning strike was much greater. The construction, design, and operating environment of civil rotorcraft have changed markedly within the past two decades. Many rotorcraft are now authorized to fly IFR in all types of weather environment. One transport design has been approved for flight into known icing conditions. Additionally, many rotorcraft now use the same advanced technologies in structures and systems as do airplanes. Because of these facts the possibility of a lightning strike encounter to the rotorcraft has been greatly increased. If the fuel system of the rotorcraft has not been properly designed and constructed, a fuel vapor ignition may occur. This occurrence generally results in a catastrophe to the rotorcraft. To prevent such a catastrophe and provide a level of safety equivalent to transport category airplanes, a specific rule for the lightning protection of transport category rotorcraft fuel systems was adopted in Amendment 29-26.

b. Explanation.

(1) This regulation requires that the rotorcraft's fuel system be designed and constructed so that an ignition of fuel vapor will not occur when the rotorcraft is involved in a lightning strike. For the purposes of this regulation the fuel system is comprised of the fuel tank with all its associated plumbing and any other areas of the rotorcraft likely to have fuel vapor present (such as sumps and drains for the tank itself). Externally mounted fuel tanks are also considered to be part of the "fuel system."

(2) Other associated installations such as electrical wiring in the fuel tanks which could provide a source of ignition due to an indirect or induced effect should also be considered.

c. Procedure.

(1) The current revision of Advisory Circular 20-53 provides guidance on an acceptable method and procedure to be utilized to demonstrate that the design and construction of the fuel system is compliant with § 29.954.

(2) FAA Report No. DOT/FAA/CT-89/22 contains additional information regarding the lightning environment. Also contained in this report are design and test techniques which provide for a design that will be adequately protected from fuel vapor ignition when the rotorcraft encounters the lightning environment. This report is available to the public by order from the National Technical Information Service, Springfield, VA 22161. *

460. § 29.975 (through Amendment 29-24) FUEL TANK VENTS AND CARBURETOR VAPOR VENTS.

a. Explanation. This section sets forth design requirements that address siphoning of fuel, pressure differentials, moisture accumulation, fumes in personnel compartments, and carburetor vapor vents.

b. Procedures. The design of the vent for the fuel system should be adequate to preclude problems associated with this section. Analysis and/or flight testing may be required to demonstrate this adequacy depending upon the fuel system design. If flight testing is required, the following flight test procedure is one method of verifying proper vent system operation.

(1) Using a rotorcraft with a fuel tank and vent system which conforms to production design specifications, install differential pressure instrumentation to measure the difference between the gas pressure inside each fuel tank expansion space and the air pressure in the cavity or area surrounding the outside of the fuel tank.

(2) Conduct ground and flight tests, recording the differential pressures between the inside and the outside of the fuel tanks. The following conditions should be evaluated:

(i) Refueling and defueling (if applicable).

(ii) Level flight to V_{NE}.

(iii) Maximum rate of ascent and descent.

(3) Compare the measured differential pressure values with the maximum allowable for the fuel tank design being evaluated. For flexible, bladder-type fuel cells, the pressure inside the tank should not be significantly less than the surrounding pressure to avoid the possibility of collapsing the bladder.

461. § 29.977 (through Amendment 29-24) FUEL TANK OUTLET.

a. Explanation.

(1) This section prescribes a fuel strainer for the fuel tank outlet (suction lift system) or for the booster pump (boosted systems) for both reciprocating and turbine engine installations.

(2) This requirement ensures that relatively large, loose objects which may be present in the fuel tank do not interfere with fuel system operation. The provisions of § 29.997 should ensure protection from smaller contaminants which may occur in service.

b. Procedures.

(1) Section 29.977(a) specifies an 8- to 16-mesh-per-inch strainer for reciprocating engine installations and a strainer which will prevent passage of any object which could restrict fuel flow or damage any fuel system component for turbine installations.

(2) In addition to the requirement of § 29.977(a), the flow area of the strainer should be at least five times the area of the outlet line. Furthermore, the diameter of the strainer must be at least that of the fuel tank outlet line.

(3) Each finger strainer should be accessible for inspection and cleaning.

(4) Compliance with § 29.977 is usually verified by inspection, and testing is not required. The ice protection provisions of § 29.951(c) are applicable to the strainer at the fuel outlet, and testing to show compliance with that provision may be required.

462. § 29.979 (through Amendment 29-12) PRESSURE REFUELING AND FUELING PROVISIONS BELOW FUEL LEVEL.

a. Explanation.

(1) Each fueling system that has the fueling connection below the fuel level in the tanks must prevent the loss of fuel if the fuel entry valve malfunctions.

(2) For pressure refueling systems, a back-up limiting device must be provided in addition to the primary means for limiting the amount of fuel in the tank.

(3) Components of the pressure fueling and defueling systems must be able to withstand an ultimate load that is 2.0 times the maximum pressure (positive or negative) most likely to occur during fueling or defueling. This requirement provides a level of structural integrity for the pressure fueling and defueling system components in the event a system malfunction occurs, which would result in an overpressurization of the fuel system. The fuel tanks and vents are not included in this requirement.

b. Procedure.

(1) Designs which have the pressure refueling and fueling provisions below the fuel level in each tank must demonstrate that when there is a malfunction of the fuel entry valve, no hazardous quantity of fuel will be lost. Generally, any amount of fuel loss in excess of 8 ounces is considered to be hazardous. Any amount of fuel that can come in contact with an ignition source is hazardous and unacceptable. Compliance should be demonstrated by test and supported by a failure mode and effects analysis.

(2) For pressure refueling systems, one of the most hazardous failure modes is an undetected overpressurization of the fuel tank which could lead to a number of potential fuel system failures. The pressure refueling system must contain a device which insures that fuel tank capacity cannot be exceeded. This device can operate on a differential pressure principle or can sense fluid level. A back-up limiting device is required in case of failure of the primary limiting device. Compliance must be demonstrated by test. A failure mode and effects analysis should be performed which verifies that the failure of either the primary or back-up limiting device will not result in the failure of the other limiting device. *

* (3) The rotorcraft pressure fueling and defueling systems must be designed to withstand an ultimate internal pressure load that is twice the maximum pressure that is likely to occur during fueling or defueling. The maximum pressure will include surges that could occur from the fueling source and/or from any single tank valve or combination of valves being either intentionally or inadvertently closed. System substantiation may be demonstrated by analysis or test. The substantiation should include all components of the pressure fueling and defueling system except the fuel tank and the fuel tank vents. The rotorcraft defueling system must also be substantiated for a negative pressure application. If tests are conducted, the pressure measurements for both tests (positive and negative) will be made at the fueling connection and the test set-up should conform to the installed system. *

463.-482. <u>RESERVED</u>.

SECTION 28. OIL SYSTEM

498. § 29.1011 (through Amendment 29-26) ENGINES: GENERAL.

a. Explanation.

(1) The oil system provided for each installed engine should provide all of the lubrication required by the engine and supply it at a temperature which is within the operating temperature limits established for that engine when it was certified.

(2) The usable oil capacity of each oil system should be sufficient to provide oil to the engine at the maximum oil consumption limit of the engine under critical operating conditions. All circulating requirements and operating temperature limits for the oil should be met.

b. Procedure.

(1) There are three basic engine oil supply and cooling system concepts that are used. There are self-contained systems (a complete system certified with the engine), systems that have both engine and airframe components, and systems that are totally supported by airframe components. Any one of these three concepts can be used to meet the requirement of having an independent oil system for each engine.

(2) Oil tank capacity is primarily determined by the engine's oil consumption rate. Other factors which should be considered when sizing the oil supply system are the endurance of the rotorcraft under critical operating conditions, and the amount of oil circulating in the system to maintain proper cooling. Adequacy of the engine oil supply system can be shown by analysis supported by engine oil consumption and cooling system data. For reciprocating engines, the ratio of one gallon of oil for each 40 gallons of fuel can be used; however, an oil-fuel ratio lower than 1:40 can be used if properly substantiated by oil consumption data on the engine.

(3) The engine oil cooling requirements are defined in §§ 29.1041 through 29.1049. The design of the engine oil cooling system will be influenced by hot day conditions, by the engine heat rejection rate, and other oil system operating data provided by the engine manufacturer. Sizing of the oil cooler will depend upon the engine data and whether the oil cooler will also be used for main transmission oil cooling. Oil cooler size should be kept as small as possible due to its effect on rotorcraft structure, but in all cases, adequate cooling should be demonstrated throughout the operating envelope of the rotorcraft. *

499. <u>§ 29.1013 (through Amendment 29-19) OIL TANKS</u>.

a. <u>Explanation</u> This regulation identifies the requirements that each oil tank must meet. It also specifies that the oil tank installation must meet the installation requirements of § 29.967.

b. <u>Procedures</u>.

(1) The oil tanks usually are constructed of aluminum, aluminum alloy, or stainless steel and are of such a design to permit installation in the aircraft as close to the engine as the design allows. The choice of materials will generally be determined by the selected location of the tank. The tank envelope or outline will generally be determined by the location within the structure of the helicopter.

(2) The design of the tank is required to meet the expansion space requirements as specified in the regulation for the particular installation. This is generally accomplished by locating the filler cap in such a manner that the expansion space cannot be inadvertently filled with the rotorcraft in normal ground attitude.

(3) The tank is required to be properly vented and the vent requirements are identified in the regulation.

(4) Unless alternate means are provided, it is good design practice to locate the oil tank with respect to the engine so that when the helicopter is in its normal ground attitude, a positive head to the oil pump inlet is provided.

(5) Sections of the regulation address specific requirements when Category A certification is requested.

(6) The designer should be aware of the requirements associated with the location of the oil tank outlet and the marking requirements specified in § 29.1557(c)(2).

(7) Flexible oil tank liners may be used; however, they must be approved or shown to be suitable for the particular installation.

(8) An "external oil system" which is defined as being those components, lines, etc., of an oil system which are outside the engine and not supplied as part of a certificated engine. The components of such a system which are within the fire zone and required to be fire resistant. Those outside the fire zone need not be fire resistant.

SECTION 31. EXHAUST SYSTEM

548. § 29.1121 (through Amendment 29-24) GENERAL.

a. Explanation.

(1) This section addresses the arrangement of exhaust components and the protection against hazardous conditions which exist with hot exhaust gases for powerplant and auxiliary power unit installations.

(2) The objective is to ensure safe disposal of exhaust gases without fire hazard or physical impairment to any occupant.

b. Procedures.

(1) During the certification process, carbon monoxide levels should be monitored in the personnel compartments to verify that the gas levels are well within the acceptable range. The conditions under which the measurements are taken should be representative of the normal operating limitations of the rotorcraft. This paragraph is not applicable to gas turbine-engine-powered rotorcraft.

(2) Exhaust system surfaces hot enough to ignite flammable fluids or vapors must meet the isolation or shielding requirements of this section in addition to the requirements of §§ 29.1183 and 29.1185. Good design practice suggests that the isolation and shielding features incorporated would continue to be effective under the emergency landing conditions specified in § 29.561.

(3) Compliance with the § 29.1121(c) fireproof requirements can be accomplished by demonstrating that the material or component will withstand a 2000 °F ± 50 °F flame for 15 minutes while still fulfilling its design purpose. This testing should accurately simulate, as near as practicable, the operating environment of the material or component in service. In addition to the fireproof requirements, the requirements of § 29.1191 must be met.

(4) Compliance with § 29.1121(d) can be accomplished by locating the vents and drains where fumes and fluids cannot interact with the hot exhaust gases. Drains should discharge positively and be a minimum of 0.25 inches in diameter. No drain may discharge where it will cause a fire hazard. This can be demonstrated by discharging a colored liquid through the drain system in flight and on the ground. The dye should not impinge on any ignition source.

(5) It should be demonstrated that exhaust gases are discharged in such a manner that they do not cause distortion or glare seriously affecting the pilot's visibility at night. One method of compliance would be a night flight evaluation at critical azimuth and variable wind conditions to verify that no degradation exists.

(6) Hot spots that can occur on exhaust system components should be eliminated by providing deflectors and/or adequate ventilation. Exhaust shrouds can either be ventilated or insulated to keep the temperatures low enough so that ignition of flammable vapors or fluids cannot occur under normal operation or under the emergency landing conditions specified in § 29.561.

(7) Compliance with § 29.1121(h) can be accomplished by ensuring that the drain will not discharge where it might cause a fire hazard. This can be demonstrated by discharging a colored liquid through the drain system in flight and on the ground. The dye should not impinge on any ignition source.

549. § 29.1123 EXHAUST PIPING.

a. Explanation. This section contains the following requirements that must be met for proper certification of exhaust piping on engines, auxiliary propulsion units (APU's) and other similar devices.

(1) § 29.1123(a) requires that the piping be heat and corrosion resistant so that it performs its intended function during its operational life (either the life of the rotorcraft or a specified limited life) without significant metal corrosion, metal erosion, or creation of hazardous hot spots. The piping system should be designed, have an installation design, or a combination that allows performance of its function without thermal expansion (thermal strain) induced structural failures, such as ruptures caused by operating temperature excursions and by overpressurization during its operational life.

(2) § 29.1123(b) requires that the piping must be supported to withstand the vibration and loading environment (including inertia loads) to which it will be subjected in service.

(3) § 29.1123(c) requires that piping that connects to components between which relative motion exists in service must have the necessary flexibility and structural integrity to withstand the relative motion without exceeding limit load (at the maximum operating temperature) of the piping, or creating unintended loads (or load paths) on the components to which the piping connects.

b. Procedures. Exhaust piping is typically certified by analysis and installation tests conducted during the basic certification process, including flight tests, as follows:

(1) For compliance with § 29.1123(a), because of its durability in the hot exhaust environment, exhaust piping is typically made from stainless steel or alloy steel of the appropriate structurally and thermally derived wall thickness. Hot aircraft exhaust gases are very corrosive; thus, proper material selection and corrosion protective design should be performed and validated during certification. Advisory Circular (AC) 43-4, "Corrosion Control For Aircraft" contains a detailed discussion of exhaust gas corrosion problems. Analysis and/or verification tests of the exhaust system should be conducted. This work is necessary to ensure thermal and structural integrity; to ensure that thermal expansion does not cause a structural overload or failure; and, to ensure that exhaust piping does not contact (or come close to) ambient temperature materials (such as structure or system components). Hot exhaust piping in contact with (or close to) ambient temperature materials can either create a fire hazard or cause an unintended strength reduction. To ensure that thermal expansion analyses and tests are properly conducted, the maximum in-service temperature excursion should be properly defined. *

The maximum temperature excursion should be based on the maximum temperature of the piping and exhaust gases, as affected by the insulatory characteristics of the piping's enclosure, and as affected by a worst-case hot day. The worst-case temperature environment used for analysis can be verified by a temperature survey. If run on cooler days, the survey can be adjusted for the worst-case hot day environment using methods identical to those used for engine cooling tests (reference paragraph 517, Cooling Tests). The piping should be designed to expand freely so that thermal expansion (thermal strain) induced loads on the piping and its restraint system are minimized. If thermal expansion induced loads (in conjunction with deflection induced loads and exhaust flow loads, discussed in b(4)) are significant relative to limit load of any item in the load path, then a fatigue check on the critical design point(s) should be performed. The fatigue check should establish a safe life or an approved limited life for the critical component(s) in the system. An accurate analytical fatigue check on exhaust piping may be difficult to perform because of erosion, corrosion, etc., in service; therefore, phased inspections should be considered to ensure the exhaust piping's continued airworthiness.

(2) For compliance with § 29.1123(b), exhaust piping should be properly supported so that the maximum loads anticipated in service are properly distributed and reacted, and, as previously discussed, so that thermal expansion induced loading is minimized. Typically the worst-case static design load conditions are either the inertia loads from an emergency impact (reference § 29.561) or the combined loading from thermal expansion, in-flight deflections and internal exhaust gas flow (see paragraph b(4)). It should be noted that several combinations of these loads should be examined to determine the critical combination. The piping should be supported and restrained such that critical frequencies are avoided and the induced vibration environment's effect is minimized. Flight test vibration surveys may be necessary, in some cases, to properly define or validate the critical modes and environment and their effect on the exhaust piping design. Operating modes such as ground idle, flight idle, 40 percent and 80 percent of maximum continuous power, maximum continuous power, OEI power settings and other power settings should be investigated to determine their vibratory effect on the exhaust gas piping system. The strength reduction of the piping materials at operating temperature (and at worst-case temperature) should be properly considered in the design and structural substantiation. MIL-HDBK-5D contains material allowables versus temperature data for a wide variety of metallic engineering materials.

(3) For compliance with § 29.1123(c), the piping and its restraint system should be designed to minimize loading induced on the piping by the relative motion (in-service deflections) of the components to which the system attaches. Isolation of significant deflection induced loading (if required based on analysis and strain surveys) by use of flexible joints or other equivalent devices or designs should be considered. Any such in-line device used to reduce deflection loading should be fireproof and leak free when performing its intended function.

(4) For critical load case determination, the expansion-induced thermal loading should be added in with mechanical relative-motion-induced loads and internal exhaust gas flow loads to provide total critical loads for both a proper static and a proper fatigue structural substantiation. The critical combined static load should be compared with the emergency impact loads of § 29.561 (paragraph b(2)) to determine the critical design load case for static strength substantiation.

(5) It should be noted that the majority of the exhaust piping verification testing required for certification can be accomplished during the rotor drive system tie down testing of § 29.923. *

* 550. § 29.1125 (through Amendment 29-12) EXHAUST HEAT EXCHANGERS

a. Explanation. This section applies only to rotorcraft powered by reciprocating engine(s) or equipped with reciprocating auxiliary propulsion units (APU). This regulation states the certification requirements for exhaust heat exchangers (EHE's) which are summarized as follows:

(1) § 29.1125(a) requires that each EHE be constructed and installed to withstand vibration, inertia and other operational loads.

(2) § 29.1125(a)(1) requires that each EHE be able to operate continuously at the highest anticipation service temperature.

(3) § 29.1125(a)(1) requires that each EHE be corrosion resistant to exhaust gases and other corrosion sources.

(4) § 29.1125(a)(2) requires that each EHE have provisions for inspecting its critical parts and areas.

(5) § 29.1125(a)(3) requires that each EHE have cooling provisions where it is subjected to hot exhaust gases.

(6) § 29.1125(a)(4) requires that each EHE muff design eliminate stagnation areas or liquid traps that would contribute to ignition of leaked flammable fluids.

(7) § 29.1125(b) requires that each EHE used to heat ventilating air for occupants--

(i) Either have a secondary heat exchanger between the primary EHE and the ventilating air system; or

(ii) Have other equivalent means to prevent harmful contamination of ventilating air.

b. Procedures. EHE's and their installations are typically certified by analysis and installation tests conducted during the basic certification process, including flight tests or simulated flight tests, as follows:

(1) Because of their durability in the hot exhaust environment, EHE's are usually constructed from stainless steel or alloy steel of the appropriate structurally and thermally derived wall thickness. The EHE and its system should be designed to expand freely to minimize thermal expansion (thermal strain) induced loads on the EHE and its restraint system. If thermal expansion induced loads (in conjunction with deflection induced loads and exhaust flow loads) are significant relative to the limit load of the EHE or its attachments, a fatigue check on critical design point(s) should be performed. The fatigue check should establish a safe life or an approved limited life for the critical component(s) in the EHE system. *

(2) EHE's should be properly supported so that the maximum loads anticipated in service are properly distributed and reacted and so that thermal-expansion-induced loading is minimized. Typically, the worst-case static design load conditions are either the emergency impact loads acting alone (reference § 29.561), or the critical combination of loads from thermal expansion, in-flight deflections and internal exhaust gas flow. Several combinations of these loads should be examined to determine the critical combination. The EHE should be supported and restrained so that critical frequencies are avoided and the induced vibration environment is minimized. Flight tests or bench tests, such as vibration surveys conducted during rotor system endurance testing, may be necessary in some cases, to properly define or validate the vibration environment and EHE's critical modes and their effect on EHE design. Operating modes such as ground idle, flight idle, 40 percent and 80 percent of maximum continuous power, maximum continuous power, OEI power settings, and other critical power settings should be investigated to determine their vibratory effect on the EHE system. The strength reduction of EHE materials at operating temperature and at critical temperatures should be properly considered in EHE design and structural substantiation (MIL-HDBK-5D contains material allowables versus temperature data for a wide variety of metallic engineering materials). The EHE and its restraint system should be designed to minimize loads induced by the relative motion (in-service deflections) of the components to which the EHE attaches. Isolation of significant-deflection-induced loading (as required, based on analysis and strain surveys) by use of flexible joints, other equivalent flexible devices, or designs should be considered. Any such in-line device used to reduce deflection loading should meet applicable certification requirements and be leak-free.

(3) Expansion analysis and verification tests of the EHE should be conducted to ensure its thermal (and structural) integrity and to ensure that thermal expansion does not cause the EHE to contact (or come close to) ambient temperature aircraft materials, structure or system components and either create a fire hazard or an unintended reduction in strength. To ensure that expansion analyses and tests are properly conducted, the maximum in-service temperature excursion should be properly defined. The maximum temperature excursion should be based on the maximum temperatures of the EHE and exhaust gases, as affected by the insulatory characteristics of the EHE's enclosure, and as affected by a worst-case hot day. The worst-case temperature environment used for analysis can be verified by a temperature survey which, when run on cooler days, can be adjusted to the worst-case hot day environment using methods identical to those used for engine cooling tests (reference paragraph 517, Cooling Tests).

(4) Hot aircraft exhaust gases are very corrosive; thus, proper material selection and corrosion protection design should be performed and validated during certification. Advisory Circular (AC) 43-4, "Corrosion Control For Aircraft" contains a detailed discussion of exhaust gas corrosion problems. The in-service corrosive environment should be identified and characterized as thoroughly as possible by chemical analysis, tests and service experience. Once defined, appropriate design techniques and materials should be selected. Certification tests may be required to ensure proper substantiation. Phased inspections and inspectability should be considered (reference (4)).

(5) The EHE's design should be reviewed for inspectability to ensure that structural and thermal integrity is maintained over the intended life of the EHE. Also, if the design review is not conclusive relative to inspectability, a tear down inspection should be conducted.

* (6) Each EHE design should be reviewed, analyzed, and tested to ensure that cooling provisions are adequate where EHE surfaces are subjected to hot exhaust gases. This is necessary to prevent hazardous hot spots or a burn through which may cause a fire and contaminate the occupied environment.

(7) Each EHE design should be reviewed, analyzed, and tested to ensure that stagnation areas and liquid traps do not exist. This can be done using bench flow tests. These stagnant areas and traps could become ignition sources if wetted with a leaking flammable fluid. A review of potential leaking flammable fluid hazards should be conducted and appropriate preventative measures such as drains and drip fences installed to ensure they are routed away from EHE's.

(8) Each EHE design which will be used to heat ventilating air for occupants should be reviewed to ensure that the EHE is a double walled system, (i.e., it would require failure of two EHE surfaces to allow toxic exhaust gases to intermix with cabin ventilating air). Each EHE wall should be designed with equal thermal and structural resistance since a single undetected inner wall failure would subject the outer wall to the primary heat load. Also, inspectability provisions should be provided or means identified to ensure that inner wall failures can be detected in service. Any equivalent means which is applied for must clearly provide an equivalent level of safety to a double walled EHE.

551.-560. RESERVED.

SECTION 32. POWERPLANT CONTROLS AND ACCESSORIES

561. § 29.1141 (through Amendment 29-17) POWERPLANT CONTROLS: GENERAL.

a. Explanation.

(1) Section 29.1141(a) references §§ 29.777 and 29.1555. The detailed compliance procedures for powerplant control arrangement and markings are found in these sections.

(2) Section 29.1141(b) requires that controls be located and/or shielded such that normal movement of cockpit personnel will not cause inadvertent control movements.

(3) Section 29.1141(c) requires that each flexible control (push-pull cables) be properly approved.

(4) Section 29.1141(d) requires that each control maintain its set position without movement from an inadvertent source such as vibration or control system loads. This is required so that constant flightcrew attention is not necessary.

(5) Section 29.1141(e) requires that each control be able to withstand operating loads without excessive deflection. Excessive deflection is interpreted to be that deflection that would cause erratic movement, lack of crispness, or premature failure.

(6) Section 29.1141(f) specifies acceptable open/close positions for manual valves to prevent power failure due to improper control valve positioning. Power-assisted valves should have means to indicate to the flightcrew that the valve is either in the fully open or fully closed position or that the valve is moving between these two positions.

(7) The control system is subject to evaluation under § 29.901(c); i.e., for turbine installations, no single failure or malfunction, or probable combination thereof, of any powerplant control system should cause the failure of any powerplant function necessary for safety. One acceptable way to determine this is by use of a failure modes and effects analysis (FMEA).

b. Procedure.

(1) For compliance with § 29.1141(a), review the procedures for paragraph 747 of this AC. Evaluation by the flight test pilot during the official flight test program is appropriate.

(2) Compliance with § 29.1141(b) is normally evaluated during the flight test program and documented in the flight test report.

(3) Compliance with § 29.1141(c) may be accomplished by qualifying the control to MIL-C-7958, "Controls, Push-Pull, Flexible, and Rigid," or other approved standards or by previous approval in a similar function, installation, or arrangement.

(4) Compliance with § 29.1141(d) may be shown during the flight test program by monitoring the means to prevent control creep. This device or arrangement should be effective without crew attention and should not impose undue control displacement loads or interfere with accurate settings.

(5) Compliance with § 29.1141(e) may be shown by an appropriate structural analysis and/or a witnessed static load test using the factors specified under § 29.397 unless a lower value can be shown to be applicable. Operation tests and design details described in §§ 29.683 and 29.685 should also be considered.

(6) Compliance with § 29.1141(f)(1) may be accomplished by installing manual valves which have positive stops in the fully open and closed positions. The fuel valves, however, may have an arrangement to facilitate the capability of switching to different fuel tanks if suitable indexing is provided. Compliance with § 29.1141(f)(2) may be accomplished by installing a device which displays to the flightcrew one indication with valve fully open and another with the valve fully closed. Alternatively, an indication could be given when the valve is moving from fully open to fully closed with the indication ceasing when the valve position corresponds to the selected switch position (open or closed). An example would be a light that is "off" when the valve is fully open or fully closed and illuminates while the valve is transitioning.

562. <u>§ 29.1142 (through Amendment 29-17) AUXILIARY POWER UNIT CONTROLS</u>.

a. <u>Explanation</u>.

(1) This section addresses control requirements for any APU installed in a helicopter.

(2) The requirement for starting, stopping, and emergency shutdown of the APU from the flight deck is primarily to control APU operation in the event of improper operation or malfunction which could affect the safety of the aircraft.

b. <u>Procedure</u>.

(1) The requirements of this section apply to all APU installations in helicopters without regard to whether or not the APU is to be operated on the ground only, or operated in flight and on the ground.

(2) The APU installation must provide sufficient controls to the flight crew to enable them to control the operation of the APU under normal and emergency conditions.

(3) Compliance can be shown by both demonstration and a failure analysis. *

b. Procedures.

(1) Certification data submitted by the applicant should be reviewed to ensure that the design features stated in § 29.1147 exist.

(2) Proper mixture control functioning (to verify the design features of § 29.1147) should be verified as part of the TIA for the certification project.

(3) Compliance is typically shown by use of a side-by-side arrangement of the controls, provided that the arrangement is compatible with other controls and considering that crew attention to the primary flight controls may be a full-time, "hands-on" operation.

566. § 29.1151 (through Amendment 29-19) ROTOR BRAKE CONTROLS.

a. Explanation.

(1) Paragraph (a) of § 29.1151 is intended to require design features which, for all practicable purposes, prevent brake application in flight even under conditions of reasonably expected crew error or confusion.

(2) Paragraph (b) of § 29.1151 would require warning devices to alert the crew if the brake has not been completely released.

b. Background. Inadvertent or undetected application of the rotor brake is expected to result in excessive heat and fire in the rotor brake area. Rotor brake components are usually located integral with, or in close proximity to, rotor drive system components and, in many cases, close to critical hydraulic main rotor control system components. Fires in these areas would be extremely hazardous.

c. Methods of Compliance.

(1) For paragraph (a), literal compliance can be achieved by lock-out devices sensitive to the higher r.p.m. ranges of the main rotor or other flight parameters, hydraulic bypass or lockout devices controlled by flyweight governor systems, etc. The guard required by § 29.921 does not, in itself, provide compliance with this requirement. For some designs, if careful evaluation of the overall control, including location, guard mechanism, control manipulation requirements, accessibility, etc., provides an extremely high degree of assurance that inadvertent application will not occur, compliance may be assumed. Also, if brake application does occur, annunciation appears, and no immediate hazard to flight operation exists, compliance may be assumed.

(2) Warning devices supplied to comply with this rule should provide a signal at any time the rotor brake is engaged, including partial engagement. Typically, micro-switches installed to close a circuit to a cockpit warning (red) light when the brake puck moves out of the retract position will provide compliance, provided the designer gives full consideration to the vibration, temperature, moisture, and other environmental considerations appropriate to configuration. Other methods such as system pressure switches, brake handle position indicators, etc., may not provide the warning required by this rule.

* 567. § 29.1157 CARBURETOR AIR TEMPERATURE CONTROLS.

a. Explanation.

(1) This section addresses the air temperature control for carburetor equipped reciprocating engines.

(2) For rotorcraft which have more than one such engine installed, a separate carburetor air temperature control must be provided for each engine.

b. Procedure.

(1) The engine air induction system should incorporate a means for the prevention and elimination of ice accumulations by preheating the air prior to its entry into the carburetor.

(2) Manually operated push/pull systems have been used which operate a flapper valve inside the air induction system. One such system for each engine is one method of compliance.

568. § 29.1159 SUPERCHARGER CONTROLS.

a. Explanation.

(1) This section addresses the accessibility to supercharger controls in the cockpit, if installed.

(2) These controls must be located so they are easily reached by the pilots or, if the rotorcraft is so configured, by a flight engineer.

b. Procedure.

(1) The location and shape of the controls should be conveniently accessible and sufficiently unique to preclude inadvertent actuation of the wrong control.

(2) Compliance is typically shown by a cockpit evaluation. *

569. § 29.1163 (through Amendment 29-26) POWERPLANT ACCESSORIES.

a. Explanation.

(1) This section addresses the interface requirements for powerplant accessories which are mounted on the engine or rotor drive system components.

(2) Areas which should be addressed include structural loads imposed upon the engine case and isolation between the accessory and engine oil systems. Electrical equipment isolation from flammable fluids or vapors should be addressed as well as the effect of an accessory failure on the continued operation of the engine and drive system components.

b. Procedure.

(1) Accessories installed and certified by the engine manufacturer can be mounted on the engine without additional justification.

(2) Any accessory to be mounted on the engine, which was not certificated with the engine and does not meet the engine installation design manual requirements, should have a structural analysis showing the mounting of that accessory on the engine will not induce loads into the engine case which are higher than the original design loads.

(3) When the accessory is mounted and operating on the engine, it should not be possible to contaminate either the engine or accessory oil systems. This contamination can take the form of debris following a failure, airborne dirt or water, or any other substance that would impair proper operation of the engine or accessory. Compliance with these requirements can be accomplished by a combination of test and analysis. The design interface should be such that when the equipment is operating, there are no high/low pressure differentials between the components which would induce fluid transfer between components resulting in a low fluid level in one component and an overfill condition in the other component. Where this potential exists, an analysis and/or test should be used to demonstrate compliance.

(4) Engine mounted accessories which are subject to arcing and sparking, must be isolated from all flammable fluids or vapors to minimize the probability of fire. This can be accomplished by isolating the electrical equipment from the flammable fumes or vapors or by isolating the flammable fumes or vapors from the potential ignition source. Compliance can be shown by analysis.

(5) A failure mode and effect analysis should be submitted which shows that a failure of any engine mounted and driven accessory will not interfere with the continued operation of the engine. If a hazard is created by the continued rotation of an engine driven accessory after a failure or malfunction, provisions to stop its rotation or eliminate the hazard must be provided. The effectiveness of this device should be demonstrated by test.

(6) The main transmission and rotor drive system should be protected from excessive torque loads and damage imposed upon them by accessory drives. One method which has been used is a torque limiting device; i.e., shear section of main rotor drive shaft. The effectiveness of any protection device should be demonstrated by test. *

* 570. § 29.1165 (through Amendment 29-12) ENGINE IGNITION SYSTEMS.

a. Explanation.

(1) This section defines the design requirements for battery, generator, and magneto ignition systems installed in either reciprocating or turbine engine powered rotorcraft.

(2) The requirements specify common failure modes of batteries, generators, and installed wiring which must be considered in the design process and provides for crew warning of malfunctions.

b. Procedure.

(1) In a battery ignition system, a generator should be available to supply current to the engine ignition system if the battery fails. The generator power should be switched over automatically with an appropriate warning to the crew. The automatic switchover can be accomplished by a low voltage sensor which activates a relay that simultaneously activates a caution light in the cockpit.

(2) An electrical load analysis should be conducted to insure that the capacity of the batteries and generator is large enough to meet the worst-case demands in the system. If there are other electrical system components installed which draw from the same source, the analysis should show that there is sufficient electrical power available from either the battery or the generator to operate all components simultaneously.

(3) The requirements of § 29.1165(c)(1) through (3), should be demonstrated by test. A proposed test plan should be coordinated with the FAA prior to conducting the testing.

(4) Compliance with the requirements of § 29.1165(d) can be shown by a failure mode and effect analysis.

(5) The requirements of § 29.1165(e) and (f) are self-explanatory.

571.-583. RESERVED. *

SECTION 33. POWERPLANT FIRE PROTECTION

584. § 29.1181 (through Amendment 29-26) DESIGNATED FIRE ZONES: REGIONS INCLUDED.

a. Explanation. A designated fire zone is a zone on a rotorcraft within which it is assumed (based on past operational experience) that a severe fire (see definitions) will occur sometime in the service life of each rotorcraft; therefore, proper protection must be provided for each new or modified unit by meeting the requirements of §§ 29.1183 through 29.1203. Some common examples of designated fire zones are:

(1) For reciprocating engines:

(i) The power section.

(ii) The accessory section.

(iii) The complete powerplant compartment,if there is no isolation between the power and accessory sections.

(2) Any auxiliary power unit (APU) compartment.

(3) Any fuel burning heater or other combustion equipment installation described under § 29.859.

(4) For Turbine Engines:

(i) The compressor section.

(ii) The accessory section.

(iii) The combustor turbine and tailpipe section unless they--

(A) Do not contain lines and components carrying flammable fluids or gases; and

(B) Are isolated from the designated fire zone prescribed in § 29.1181(a)(6) by a firewall that meets § 29.1191.

(5) Any other essential or non-essential device or system (such as spray rigs using flammable fluids) capable of leaking flammable fluid or gas and creating a severe fire.

b. Definition.

Severe fire. See definition in paragraph 360.

c. Procedures. A FAA/applicant design review should be conducted early during certification to identify all designated fire zones and to define the detailed method-of-compliance to be used to meet the requirements of §§ 29.1183 through 29.1203. If significant design changes are made the design change and the method-of-compliance should be re-reviewed to insure they properly support the certification requirements. *

* 585. § 29.1183 (through Amendment 29-22) LINES, FITTINGS, AND COMPONENTS.

a. Explanation. This section requires that any line, fitting or other component of a flammable fluid, fuel or flammable gas system which carries, conveys or contains the fluid or gas in any area subject to engine fire conditions (i.e., a severe fire) must be at least fire resistant (reference § 1.1 for definition of fire resistant and see paragraph 584 which defines a severe fire). An exception is for flammable fluid tanks and supports which are part of and attached to the engine or are in a designated fire zone. These items are required to either be fireproof (see § 1.1 for definition of fireproof and see paragraph 584 which defines a severe fire) or to be enclosed by a fireproof shield, unless fire damage to any non-fireproof part (e.g., secondary line or valve support) will not cause leakage of a flammable gas, flammable fluid or otherwise prevent continued safe flight and landing of the rotorcraft. All such components must be shielded, located, otherwise protected, or a combination to safeguard against the ignition of leaking flammable fluids or gases. Integral oil sumps of less than 25 quarts capacity on a reciprocating engine need not be fireproof or enclosed by a fireproof shield; however, they should be fire resistant. Most integral sumps in this category are, by natural design and material selection, fire resistant. Exemptions to the preceding requirements are as follows:

(1) Lines, fittings and components already approved under Part 33 as part of the engine itself;

(2) Vent and drain lines (and their fittings) whose failure will not result in or add to an operational fire hazard. In addition, all flammable fluid drains and vents must discharge clear of the induction system air inlet and other obvious ignition hazards.

b. Procedures. A detailed review of the design should be conducted to identify and quantify all lines, fittings, and other components which carry flammable fluids and/or gases and are in areas subject to engine fire conditions such as engine compartments and other fire zones. Once these items are identified the design means of fire protection should be selected and validated, as necessary, during certification. For materials and devices that cannot be qualified as fireproof or fire resistant by similarity or by known material standards, testing to severe fire conditions (see definition, AC 20-135 and AC 23-2 for detailed requirements) should be conducted on full-scale specimens or representative samples to establish their fireproof or fire resistance capabilities. Exceptions to these standards (as provided in the regulatory section) should be reviewed and approved/disapproved on a case-by-case basis during certification. Also, operational fire hazards from drains, vents, and other similar sources should be identified and eliminated during certification.

586. § 29.1185 FLAMMABLE FLUIDS.

a. Explanation. This section requires that fuel, flammable fluid or vapor tanks, reservoirs or collectors be sufficiently isolated from engines, engine compartments, and other designated fire zones so that hazardous heat transfer from these areas to fuel, flammable fluid, and vapor tanks, reservoirs or collectors is prevented in either normal or emergency service.

b. Definitions.

(1) Fuel or Flammable Fluid Collector. Any device such as a large valve, accumulator, or pump that contains a significant amount of flammable fluid, fuel, or vapor (e.g., the volume equal to 10 ounces or more of fluid).

(2) Flammable Fluid or Vapor Tank. Any fuel, flammable fluid or vapor tank, reservoir or collector.

(3) Sufficiently Isolated. Fuel, flammable fluids, or vapors in a tank, reservoir, or collector are insulated, removed, otherwise protected or a combination such that their worst-case temperatures (the worst-case measured or calculated surface temperature of their containers) in either normal or emergency service is always 50 degrees Fahrenheit or more away from the autoignition temperature of the fuel, flammable fluid, or vapor in question.

(4) Minimum Autoignition Temperature. The temperature at a given vapor pressure at or above which liquid fuel or fuel vapor will self combust. When determining the minimum design value of autoignition temperature which will occur in either normal or emergency operations, the critical, in-service combination of vapor pressure and fuel temperature should first be determined.

(5) Hazardous Heat Transfer. A total incident heat flux (a combination of conduction, convection and radiation, as applicable) from or in an engine compartment or any other designated fire zone which would raise the temperature level of a flammable fluid or fuel, their vapors, or the surface temperature of their containers to within 50 degrees Fahrenheit or less of the minimum in-service autoignition temperature. Typically, the most critical heat transfer case to be considered is emergency service where a severe fire (see definition) is assumed to occur in each engine compartment and each designated fire zone on a case-by-case basis.

(6) Severe Fire. See definition in paragraph 360.

c. Procedures.

(1) The fuel, flammable fluid and vapor system designs should be reviewed early in the certification process to insure that all fuel or flammable fluid or vapor tanks, are properly identified and isolated from engines, engine compartments and other designated fire zones during both normal and emergency operations such as in-flight engine compartment or other fire zone fires. In some cases fuel or flammable fluid components must be located in an engine compartment or other designated fire zone. In these cases, an equivalent safety finding (which considers the design, construction, materials, fuel lines, fittings, and controls used in the

* system, or system segment, contained in the engine compartment or other designated fire zone) should be undertaken as a part of the normal certification process. If the level of safety provided is equivalent to that provided by removing the system or system segment from the engine compartment or designated fire zone,then the design should be accepted. For fuel tanks only, isolation is required by regulation to be achieved by use of either a firewall (reference Paragraph 589 for Firewall Requirements) or by use of a shroud. A shroud if used should be fireproof (see § 1.1 for definition, and the definition of a Severe Fire for further details) and should be drainable (or otherwise inspectable) to insure the fuel tank is not leaking in service. For other flammable fluid or vapor tanks, the regulations allow either the identical treatment previously described for fuel tanks (i.e., firewalls or shrouds) or, alternatively, use of an equivalent safety finding. Regulations require that the equivalent safety finding be based on system design, tank materials, tank supports, and flammable fluid system connectors, lines, and controls. In all cases the flammable fluids, fuels, and vapors should be sufficiently isolated from hazardous heat fluxes during both normal and emergency operations to prevent autoignition.

(2) In addition, the regulations require at least one-half inch of clear airspace between each flammable fluid or vapor tank, and each firewall or shroud that isolates the system, unless equivalent means (such as fireproof insulation) are used to prevent hazardous heat transfer from each engine compartment or other fire zone to the flammable fluid or vapor mass (or its container surface) at the fluid or vapor's minimum autoignition temperature. If in-service structural deflections are significant, they must be taken into account when certifying the one-half inch minimum clear airspace requirement. For example, if a one-half inch clearance exists on the ground but in some normal and emergency flight conditions (e.g., autorotation) the one-half inch is reduced to one-fourth inch at a critical time (in-flight engine fire), then the design (static) configuration should have at least a one-half plus one-fourth equals three-fourths inch static clear airspace to insure the regulation's intent is met. Alternatively, fireproof insulation or additional stiffeners could be used to insure the regulation's intent is met (i.e., the thermal equivalent of one-half clearance is maintained at all times). Any material used as insulation on or used adjacent to flammable fluid or vapor tank, should be certified as chemically compatible with the flammable fluid or vapor and to be non-absorbent in case of fuel or vapor leaks. Otherwise, the material should either be treated for compatibility and non-absorbency or not accepted.

587. § 29.1187 DRAINAGE & VENTILATION OF FIRE ZONES

a. Explanation. To insure that any component malfunction which results in fuel, flammable fluid or vapor leaks is safely drained or vented overboard and to insure that a fire hazard is not created during either normal or emergency service, there should be complete, rapid drainage and ventilation capability present for each part of the rotorcraft powerplant installation and any other designated fire zone which utilizes flammable fluid or vapor carrying components. As a minimum, the routing, drainage and ventilation system should accomplish the following:

(1) It should be effective under normal and emergency operating conditions.

(2) It should be designed and arranged so that no discharged fluid or vapor will create a fire hazard under normal and emergency operating conditions.

(3) It should prevent accumulation of hazardous fluids and vapors in any engine compartments and other designated fire zones.

b. Definitions.

(1) Drip Fence. A physical barrier that interrupts the flow of a liquid on the underside of a surface, such as a fuel tank, and allows any leaked liquid to drip from the surface away from a hazardous locations to a safe external drain.

c. Procedures. The design of flammable fluid and gas systems running through engine compartments and other designated fire zones should have a thorough hazard analysis performed early during certification. The analysis should be updated periodically as design changes dictate. The hazard analysis should identify and quantify all normal and emergency service failures that could result in leakage of fuel, flammable fluids and vapors. Once these potential hazards are identified and quantified, appropriate design features, such as drains, drip fences and vents, that minimize or eliminate the hazard should be provided. These means should be analyzed and/or tested, as necessary, to insure that their size, flow capacity, and other design parameters are adequate to rapidly remove hazardous fluids and vapors safely away from the rotorcraft under normal and emergency flight conditions. Typically a venting or draining system should be designed to a 3-to-1 flow capacity margin over the probable worst-case leak to which it could be subjected. Adverse effects such as clogging and surface tension flow reduction should be accounted for in design. Testing, including flight testing, using inert fluids or vapors may be necessary for proper design certification. In some instances it may be appropriate to include ventilation and drainage tests when the aircraft is parked. *

* 588. § 29.1189 (through Amendment 29-26) SHUTOFF MEANS.

a. Explanation.

(1) This section establishes the requirements for controlling hazardous quantities of flammable fluids which flow into, within, or through designated fire zones.

(2) When any shutoff valve is operated, any equipment, including a remaining engine, which is essential for continued flight, cannot be affected.

b. Procedure.

(1) Combustible fluid supply lines which pass into, within, or through a firewall into the fire zone must incorporate shutoff valves. This requirement does not apply to lines, fittings, and components which were certified with and are part of the engine. These requirements do not apply to oil systems for Category B rotorcraft with reciprocating engines with less than 500 cubic inches displacement or to any other installation where all components, including the oil tanks, are fireproof or are located in an area that will not be affected by an engine fire.

(2) Eight fluid ounces or less of a combustible fluid is not considered hazardous and no more than this amount should be present after activating the shutoff valve.

(3) Engine isolation is to be maintained when incorporating shutoff valves into engine fuel and lubrication lines. The design must insure that when one engine is shut down or fails and the fuel and lubrication fluid shutoff valves are activated, the remaining good engine is not affected in any way, and the rotorcraft can continue safe flight to a landing. This should be demonstrated by test.

(4) Each shutoff valve located in a fire zone should be fireproof. If the shutoff valve is located outside of the fire zone, then it should be at least fire resistant or protected so that it will function under a worst-case fire condition within a fire zone. This should be demonstrated by test.

(5) Except for ground-use-only auxiliary power unit installations, the flammable fluid shutoff to all engine installations must be protected from inadvertent operation. Where electrical shutoffs are used, the switches must be guarded or require double actions. If the shutoffs are mechanically activated, the design of the knob and the location of the lever must be such that inadvertent actuation cannot occur. It must be possible to reopen the shutoff valve in flight after it has been closed and this should be demonstrated by test.

589. § 29.1191 (through Amendment 29-3) Firewalls.

a. Explanation. This section states the certification requirements for the proper certification of fireproof protective devices such as firewalls, shrouds, or equivalent. These devices are necessary to isolate each engine (including combustor, turbine, and tailpipe sections of turbine engines and auxiliary propulsion units (APU); each APU; each combustion heater; each unit of combustion equipment; or each high temperature device (or source) from personnel compartments and critical components (not already protected under § 29.1191). The isolation of these fire zones is necessary to prevent the spread of fire, prevent or minimize thermal injuries and fatalities, and prevent damage to critical components that are essential to a controlled landing. Even though § 29.1191(b) implicitly excludes APU's, combustion heaters, and other combustion equipment that are not used in flight; they should be protected by fireproof enclosures, because of the requirements of § 29.901(d) and relevant parts of §§ 29.1183 through 29.1203. This is because, even if the device is rendered inoperative in flight, it typically contains residual heat, fuel, fumes and potential ignition sources (i.e., "potential hazards"). Each fireproof protective device must, by regulation, meet the following criteria:

(1) Its design and location must take into account the probable fire path from each fire zone or source considering factors such as internal airflow, external air flow and gravity.

(2) It must be constructed so that no hazardous quantity of air, fumes, fluids or flame can propagate through it to unprotected parts of the rotorcraft.

(3) Its openings (e.g., shaftholes, lineholes, etc.) must be sealed with close fitting fireproof grommets, bushings, bearings, firewalls fittings, or equivalent that prevent burn through and leakage of hazardous fumes or fluids from the fire zone.

(4) It must be fireproof (see definition).

(5) It must be either corrosion resistant or otherwise safely protected from corrosion.

b. Definitions.

(1) Fireproof Protective Device. A fireproof protective device is a device such as a firewall, shroud, enclosure, or equivalent used to isolate a heat or potential fire source (severe fire) from personnel compartments and from critical aircraft components which are essential for a controlled landing.

(2) Fireproof. Fireproof is defined in § 1.1 "General Definitions."

(3) Controlled Landing. A landing which is survivable (i.e., does not fatally injure all occupants) but may produce an unairworthy, partially salvageable, or unsalvageable rotorcraft.

(4) Severe Fire. See definition in paragraph 360. *

* c. Procedures. Fireproof protective devices are typically certified by analysis, tests, or a combination conducted during the certification process, including flight tests or simulated flight tests, as follows:

(1) Fireproof protective devices should be provided wherever a hazard exists which requires isolation from a severe fire (see definition) to avoid fires in personnel compartments and to avoid thermal damage to critical components (such as structural elements, controls, rotor mechanisms, and system components) that are necessary for a controlled landing. A thorough hazard analysis should be conducted during certification to identify, define and quantify in order of severity (i.e., maximum temperature, hot exposed area, etc.) all thermal hazards or zones that require fireproof protection in a given design. Engines (including the combustor, turbine, and tailpipe sections of turbine engines), APU's, combustion heaters, and combustion devices are required by regulation to be isolated. Other high temperature devices may also require isolation because of local hot spots (which occur during normal operations or from failure modes) that can thermally injure occupants or cause spontaneous combustion of surroundings. A hazard analysis shoul identify these potential problems and provide proper certification solutions.

(2) Fireproof protective devices should be able to withstand at least 2000 ± 150 degrees Fahrenheit for at least 15 minutes (reference AC 20-135). The fireproof protective device should allow the protected parts, subsystems or systems to perform their intended function for the duration of a severe fire (see definition). For firewalls, examples of flat, geometry materials undergoing unifor heat fluxes with material gauges that automatically meet the certification requirements are given in Table 589-1. If firewalls are utilized that involve othe materials, significant geometric changes, or significantly non-uniform heat fluxes, then automatic compliance may not be assured. In such cases the fireproof protective devices should be analyzed and, in some cases, tested in accordance with AC 23-2 to insure proper certification. For example, a curved protective surface may absorb a uniform incident heat flux unevenly and create a local hot spot that exceeds 2050 degrees Fahrenheit that burns through in less than 15 minutes; whereas a flat surface of equal thickness would not exceed 2050 degrees Fahrenheit and woul not burn through in less than 15 minutes. It should be noted that composite materials are not generally used for protective devices because of their inability to withstand high temperatures (i.e., exceedance of the glass transition temperature); however, some specially formulated composites have been previously certified as engine cowlings. Titanium is an acceptable material for fireproof protective devices such as firewalls. However, use of titanium should always be carefully considered and reviewed, because it can lose all structural ability and burn severely (self combust) above 1,050 degrees Fahrenheit, under certain thermodynamic environments, and contribute to the fire instead of providing the intended fire protection. AC 33-4, "Design Considerations Concerning the Use of Titanium in Aircraft Turbine Engines" and MIL-HDBK-5D contain more detailed information on the unique thermal properties of titanium.

TABLE 589-1
TABLE OF MATERIALS AND GAGES ACCEPTABLE
FOR FIREPROOF PROTECTIVE DEVICES WITH FLAT
SURFACE GEOMETRIES(1)

MATERIAL(2)	MINIMUM THICKNESS(3)
Titanium Sheet	.016 in
Stainless Steel	.015 in
Mild Carbon Steel	.018 in
Terne Plate	.018 in
Monel Metal	.018 in
Firewall Fittings (Steel or Copper Base)	.018 in(4)

NOTES:

(1) Assumes essentially flat vertical or horizontal surfaces undergoing a uniform heat flux. Any significant variation in either geometry or heat flux distribution should be examined in detail for adequate gauge thicknesses on a case-by-case basis.

(2) Must have corrosion protection if not inherent in the material itself.

(3) The minimum thickness is for thermal containment only. Structural integrity considerations may require thickness increases. MIL-HDBK-5D contains material allowable versus temperature data for common metallic materials.

(4) This is the minimum wall thickness measured at the smallest dimension (e.g., thread root or other location) of the part.

(5) Distortion of thin sheet materials and the subsequent gapping at lap joints or between rivets is difficult to predict; therefore, testing of the simulated installation is necessary to prove the integrity of the design. However, rivet pitches of 2 inches or less on non load-carrying titanium firewalls of .020 inch or steel firewalls of .018 inch are acceptable without further testing.

(3) The probable path of a fire (as affected by internal and external air flow during normal flight and autorotation, gravity, flame propagation paths, or other considerations) should be taken into account when performing the hazard analysis of item (1). Such a review will insure that fireproof protective devices are placed in the proper location for intercepting, blocking or containing a severe fire before occupants are injured and a controlled landing is prevented. If the probable path cannot be readily determined by inspection or analysis, testing using simulated airflows, rotorcraft attitudes and dyed inert fluids or vapors can be used to aid in this determination. *

* (4) Each opening in a protective device should be sealed with close fitting sealing devices such as fireproof grommets, bushings, firewall fittings, rotating seals or equivalent that are at least as effective as the fireproof protective device itself. This is necessary to insure that no local breakdowns in protection occur. For materials not listed as acceptable in item (1), FAA standards and analysis and testing should be required in accordance with the definition of a severe fire for proper substantiation.

(5) Each protective device should be fireproof in order to withstand a severe fire (see definition). Unless designs and materials have been previously FAA approved (e.g., see Item 1), the protective device's design and material selection should be tested to insure its fireproof thermal and structural integrity. A full-scale test of a structurally loaded article or a representative sample should be conducted to insure proper compliance is achieved. Also, the continued sealing ability of the protective device in its deformed state due to a hard controlled landing should be considered during certification (e.g., use of ductile materials). The corrosion environment should be defined and appropriate protection provided. Phased inspections should be specified, if necessary, to insure continued corrosion integrity. Certification tests for adequacy of corrosion protection should be conducted, using sample plates or by other equivalent means, as required. *

590. § 29.1193 (through Amendment 29-24) COWLING AND ENGINE COMPARTMENT COVERING.

a. Explanation.

(1) Section 29.1193(a) requires the cowling and engine compartment coverings to withstand structurally loads experienced in flight.

(2) In order to prevent pooling of flammable fluids, § 29.1193(b) requires ventilation and complete drainage from the cowling and engine compartment as specified in § 29.1187.

(3) In § 29.1193(c), (d) and (e), clarification of fireproof requirements is provided along with interaction between the requirements of § 29.1191 for firewalls.

b. Procedures.

(1) Compliance with § 29.1193(a) can be shown by analyzing the cowling and engine compartment covering and determining that no structural degradation will occur under the highest loads experienced on the ground or in flight.

(2) Compliance with § 29.1193(b) can be accomplished by ensuring that the drain will discharge positively with no traps and is a minimum of 0.25 inches in diameter. No drain may discharge where it might cause a fire hazard. This can be demonstrated by colored liquid flowing through the drain system while in flight. The dye should not impinge on any ignition source during any approved flight regime.

(3) Compliance with the fireproof requirements of § 29.1193(c), (d), and (e) can be accomplished by demonstrating that the material will withstand a 2,000 °F ± 50 °F flame for 15 minutes while still fulfilling its design purpose. This testing should accurately simulate, as near as practicable, the likely fire environment to prove the materials and components will provide the necessary fire containment when exposed to a fire situation in service. In addition to the fireproof requirements, the requirements of § 29.1191 must also be met. The primary objectives are:

(i) To contain and isolate a fire and prevent other sources of fuel and/or oxygen from feeding the existing fire; and

(ii) To ensure that components of the engine control system will function effectively to permit a safe landing and/or shutdown of the engine.

* 591. § 29.1194 (through Amendment 29-3) OTHER SURFACES.

a. Explanation. This section states the fire resistance requirements for material surfaces near engine compartments and designated fire zones (other than tail surfaces not subject to heat, flames or sparks emanating from a designated fire zone or engine compartment).

b. Definition.

(1) Other Surface. Any airframe, system or powerplant component aft of and near an engine compartment, a designated fire zone, or another heat source which would receive a heat flux as a result of a fire in the engine compartment or fire zone that would require the component to be fire resistant.

(2) Fire Resistant. In accordance with § 1.1, is defined as follows:

(i) Sheet metal or structural members with the capacity to withstand the heat associated with the fire at least as well as aluminum alloy in dimensions appropriate for the purpose for which they are used.

(ii) Fluid carrying lines, fluid system parts, wiring, air ducts, fittings and powerplant controls with the capacity to perform their intended functions under the heat and other conditions resulting from a fire.

(3) Fire. A fire in either an engine compartment or a designated fire zone is assumed to occur that produces a heat flux on a system, airframe or powerplant component aft of or near the fire. The effect of each such fire on other surfaces must be considered on a case-by-case basis to determine the critical case. Unless a more rationale definition is furnished and approved during certification, the fire in any engine compartment or designated fire zone should be assumed, for purposes of analysis, to be a severe fire (see definition in paragraph 360).

c. Procedures.

(1) Other surfaces should be identified during certification by a design review and by a conservative, thorough hazard analysis based on an analytical estimate of the total heat flux (i.e., conduction, convection and radiation in combination, as applicable) using the definition of a severe fire and of the resultant "other surface" temperature based on a single fire occurring in each engine compartment and designated fire zone, on a case-by-case basis. Once the other surfaces are identified and their severe fire-induced maximum temperatures determined, their configuration and material selection should be reviewed on a case-by-case basis to determine either that they are fire resistant, that they can be made fire resistant (within the limits of practicability) or that it is impracticable to make them fire resistant. If the non-fire resistant other surfaces can be readily made fire resistant they should be. If it is impracticable to make them fire resistant, then they should be relocated, insulated or a combination in order to reduce the total incident heat flux (and, thus, lower their surface temperature) so that they are no longer need to be fire resistant. If insulation is used to shield a surface that is subjected to a significant temperature, it must be fire resistant.

(2) A partial validation of analytical heat flux models using the definition of a severe fire can sometimes be achieved during certification tests by using thermocouples or heat-sensitive stickers to measure in-flight temperature ranges and distributions on other surfaces from known thermal environments in engine compartments or other designated fire zones.

592. 29.1195 (through Amendment 29-17) FIRE EXTINGUISHING SYSTEMS.

a. Explanation. This section specifies the types of rotorcraft which must have fire extinguishing systems and the number of discharges. The types of tests and airflow conditions are also specified for demonstration of compliance.

b. Procedures.

(1) The requirements are applicable to each turbine engine powered rotorcraft, Category A reciprocating engine powered rotorcraft, and each Category B reciprocating engine powered rotorcraft with an engine of more than 1,500 cubic inches. There must be a fire extinguishing system for the designated fire zones defined in § 29.1181.

(2) A fire extinguishing system should dilute all of the atmosphere within and entering a compartment with sufficient inert agent that it will not support combustion, and continue the process long enough to extinguish the existing flame and either dissipate the vapors or eliminate the ignition sources. Conventional systems utilize perforated tubing or discharge nozzles to distribute a specific quantity of agent in approximately 2 seconds. HRD (high rate of discharge) systems utilize open end tubes to deliver a given quantity of agent within 1.35 seconds for CO_2 and 1 second for all other agents. The HRD systems are recommended for use in compartments having high airflow where the required discharge rates can be more effectively provided by a HRD rather than a perforated tubing system. Tests indicate that unrestricted release through such an open end tube distribution system can be relied on for adequate distribution, provided the outlets are located properly. Although the discharge times given above are considered satisfactory, any reduction in discharge time below that specified would improve system effectiveness. However, consideration should be given to the time requirements for draining accumulated combustibles, dissipating combustible vapors and cooling or eliminating ignition sources to assure that the minimum agent concentration is maintained for a duration sufficient to prevent reignition of the combustibles.

(3) The possible variety of tankage and plumbing configurations to accomplish the result should be examined for each specific aircraft in order to achieve the optimum. Systems can vary from tankage in a central location, which is directed through complex distribution systems to various hazards, to agent which is tanked adjacent to each hazard. Terminology generally accepted to define various arrangements is as follows:

(i) Central System: A single supply of agent, centrally located, with valves to direct the agent to any protected zone or zones.

(ii) Individual System: A separate supply of agent for each protected zone or zones. *

* (4) The selection of the distribution system should be made with full cognizance of the hazards to be covered. The distributor system (i.e., discharge nozzles fed individually by lines from a central manifold) is the most efficient. The complexity of such a system, however, may show it in a less favorable light than the loop or ring system (i.e., orifices drilled in a distribution line, the loop being fed from one end, and the ring being fed from a point on a continuous circle) as far as weight, complexity of manufacture and types of hazard to be covered are concerned. For HRD systems, open feed lines are recommended. In high air flow zones, outlets should be located as far upstream as possible with the discharge directed across the air stream and slightly downstream such that a helical spray pattern is produced. In zones of low or negligible air flow, the outlet location is not critical but a location at the top-center of the zone with the agent directed downward is suggested.

(5) In a conventional CO_2 system, all lines upstream of direction valves should be 4,000 psi (27,600 kPa) burst and lines which are open should be 2,000 psi (13,800 kPa) burst. Care should be taken to insure that all valving and/or equipment in the distribution line has an appropriate flow rate. Expansion of fittings, tee, etc., should be checked to insure that not over 150% of the inflow area exists downstream. 130% is accepted as the safest target value. If overexpansion occurs, snow will form and plug the lines. Because of high storage pressure, the orifice areas of a conventional CO_2 system seem to act as the flow control with system flow losses as a minor effect. Because of this, distribution systems of 50 ft (15.24m) or less can be satisfactorily computed by the following factor:

(i) Line Area = .10 sq in./lb CO_2/sec(142.2 $mm^2/kgCO_2/sec$).

(ii) Orifice Area = .072 sq in./lb CO_2/sec(102.4 $mm^2/kgCO_2/sec$)(72% of equivalent line area).

(iii) Min. Orifice Size = 1/16 in. (1.6 mm) diameter.

(6) In low pressure systems such as "CB" and CH_3Br, line and fitting losses become a greater effect in the discharge rates and distribution than was true with CO_2. Consideration should be given to the small I.D. of an AN line fitting with respect to the I.D. of the mating tube sizes. This may be done by extra pressure drop allowances, by enlarging these fittings, or by making special fittings. Within reasonable line lengths, however, area factors can be used with fair accuracy. (It is generally conceded that a system designed to these factors, especially a complex layout, should be carefully tested or analyzed for time of discharge and distribution.) These areas are as follows:

(i) Line Area = .07 sq in./lb agent/sec (99.6 mm^2/kg agent/sec)

(ii) Orifice Area = .05 sq in./lb agent/sec(71.1mm^2/kg agent/sec) (72% of equivalent line area)

(iii) Min. Orifice Area = 1/32 in. (.8mm) diameter *

(7) For HRD systems of all types, feed line cross-sectional area is dependent upon the rate desired and upon system volume considerations. The minimum diameter of the feed line is established by the required rate; the maximum diameter of the feed line, and by the need for keeping the system volume at a minimum. Specifically, with the propelling gas in a system pressurized to 400 psi (2760 kPa), the "volumetric efficiency" should be at least 0.50; that is, the original volume of the propelling gas in the system should be at least one-half the volume of the entire system, including that of the agent container. It is recommended that for HRD systems the feed lines be open. No nozzles or series of perforations are required. It is believed that the unrestricted release of the more volatile liquid agents, as well as carbon dioxide, can be relied upon for adequate distribution, provided the outlets are properly located. It is important that any such system be carefully tested for time of discharge, distribution, and minimum concentrations.

(8) From the basic definition, the system should be effective if the distribution of the agent floods the various portions of a compartment simultaneously and dilutes the incoming air. It is noted that the typical high air flow compartment requires a greater proportion of its total agent discharged at the air inlet than does the conventional low air flow zone. All parts of the fire extinguisher system directed to any one powerplant installation should be discharged simultaneously. The theory behind the HRD type system is that with rapid discharge of the agent, the concentration necessary for extinguishment is reached more rapidly with correspondingly less time for dissipation or dilution of the agent by incoming air. The duration of this critical concentration necessary for extinguishment is believed to remain the same as for conventional systems.

(9) Detailed system configuration recommendations are not available for conventional systems; however, the recommendations on the configuration of HRD systems would probably apply equally well to all types. For HRD systems, it is recommended that feed lines be as short as possible, requiring that agent containers be as close as practical to the zones to be protected. Feed lines should be direct; the fewer fittings and turns, the better. Expansions and restrictions have adverse effects on rate; and it is probable that in a feed line with long rises or many changes or direction, quantities of propelling gas can get past a liquid agent, thus reducing the discharge rate and making the discharge sporadic and ineffective. Where such fittings, changes of direction, and long vertical rises are unavoidable, compensation in the form of additional agent may be necessary.

(10) A fixed "one shot" fire extinguisher system should be provided for the heater extinguisher system in order to extinguish the fire in the combustion chamber. The regions surrounding the heater and combustion chamber must also be protected if these regions contain components with potential combustible leakage. No fire extinguishment is needed in cabin air passages. *

* 593. 29.1197 (through Amendment 29-13) FIRE EXTINGUISHING AGENTS.

a. Explanation.

(1) Fire extinguishing agents used in rotorcraft fire extinguishing systems must be capable of extinguishing any fire in the area where the system is installed.

(2) The extinguishing agent must maintain its effectiveness after prolonged storage under the environmental conditions of the compartment in which it is stored.

(3) If a toxic extinguishing agent is used, the harmful concentration level of the fluid vapors must be determined and it must be shown that it is not possible for this concentration level to enter into any personnel compartment.

b. Procedure.

(1) The fire extinguishing system should dilute all of the atmosphere within and entering a compartment with sufficient inert agent so that combustion cannot be supported. The extinguishing process should continue for a duration sufficient in length to extinguish any existing flame. When a compartment is to be flooded with agent and there is a source of fresh air entering the compartment, the incoming air should be either shut off prior to the release of the agent or rendered inert by directing extinguishing agent into the air blast (preferably the former) or the quantity of agent should be increased to offset the incoming airflow.

(2) There are a number of extinguishing agents which have been used on rotorcraft in the past. The following list identifies the agent and some advantages and disadvantages of each.

Agents	Advantages	Disadvantages
Carbon Dioxide CO_2	Safest agent to use from the standpoint of toxicity and corrosion hazards.	Mental confusion and suffocation hazard to occupants if sufficient gas is discharged into personnel compartments. CO_2 has an extremely large variation in vapor pressure with temperature which makes it necessary to use stronger (heavier) containers than are required for methyl bromide.
Methyl Bromide CH_3Br	More effective for equal mass than CO_2. Approx. 80% of this agent by weight as compared to CO_2 is required.	Much more toxic than CO_2. Due to its toxic effects on humans, CH_3Br should not be used as a fire extinguisher agent in areas where harmful time concentrations can enter personnel compartment. *

(continued)

Agents	Advantages	Disadvantages
	Less variation in vapor pressure than CO_2. Much lower container pressure required resulting in lighter containers. Treated magnesium alloys are satisfactory for use in CH_3Br systems outside of the potential fire zones.	A warning agent such as colored smoke or a harmless irritant should be mixed with the agent. Aluminum alloy material should not be used in methyl bromide systems due to serious corrosion and possible spontaneous ignition. Rapidly corrodes aluminum, magnesium, and zinc. Tubing systems should be vented at all times and steps should be taken to free the tubing of residual methyl bromide after each discharge. Containers must be recharged at the extinguisher manufacturer's plant or at a depot by specially trained personnel.
Bromo-chloro-methane ("CB") CH_2BrCl	Low vapor pressure compound - 3 psia (20.7 KPa) at 70°F (21.1°C). One of the more effective agents.	Toxic when burned.
Dibro-modi-fluoro-methane CF_2Br_2	Low vapor pressure compound - 14 psia (96.5 KPa) at 70°F (21.1°C). One of the more effective agents. Non-corrosive to aluminum, steel and brass.	Very toxic when burned.
Bromotri fluoro-methane CF_3Br	One of the more effective agents. Low toxicity in natural condition and when burned. Non-corrosive to aluminum, steel and brass.	High vapor pressure compound - 220 psia (1517 KPa) at 70°F (21.1°C). Least toxic of agents in burned condition except for CO_2.

*

(continued)

Agents	Advantages	Disadvantages
Nitrogen N_2	If a fuel tank inerting system using N_2 is provided, use as extinguishing agent may be considered. N_2 offers cooling not available with CF_3Br.	3-4 times quantity and rate of conventional agents required.

Note: The relative effectiveness of the various agents listed above is considerably influenced by the type of system employed, high rate discharge or conventional; by the method of distribution, open end outlet, nozzle, or spray ring; and by the air flow conditions.

(3) The extinguishing agent must not be affected by the temperature extremes experienced in the compartments in which they are stored. The agent containers should be either "winterized" for extreme temperature operation or so located in the rotorcraft that they will not be subjected to extreme temperatures. Safe limits for unwinterized carbon dioxide cylinders are approximately 0°F (-18°C) to 140°F (60°C). Safe limits for "CB" and CH_3Br spheres are approximately -65°F (-54°C) to 200°F (93°C). The cartridge detonators have a variable age-with-temperature limit. Contact should be made with the manufacturer for the latest information available for both installation and storage temperatures.

(4) It must be shown by test that the harmful level of toxic fluid or vapors cannot enter into any personnel compartment due to leakage or activation of the system during normal operation of the rotorcraft in flight or on the ground. The entire fire extinguishing system should be mocked-up or installed in the aircraft down to and including distribution tubing and outlets. The tests should be conducted under actual or simulated cruise conditions. The system should be discharged, and compliance verified by use of an appropriate method for measuring agent concentration. *

594. <u>§ 29.1199 (through Amendment 29-13) EXTINGUISHING AGENT CONTAINERS</u>.

a. <u>Explanation</u>.

(1) This section presents the requirements for fire extinguisher containers. The containers are subjected to high internal pressures for the propulsion of the agent as well as a wide range of external environmental temperatures.

(2) The containers must be adequately protected to preclude any adverse effect on the operation of the system from these external influences.

b. <u>Procedure</u>.

(1) Each extinguishing agent container must have a pressure relief valve which will open at a pressure that is below the burst pressure of the agent container. The pressure relief valve lines must be located and protected so that they cannot be clogged by dirt, ice, or other contaminates. Both the agent container burst pressure and the relief valve opening pressure limits should be verified by test. Agent containers which meet military specification, MIL-C-22284, requirements are acceptable.

(2) The containers should be located so that an indicator is readily visible to determine if the container has discharged or the charging pressure is below operating minimums. The number and size of agent containers should be adequate to obtain the established agent concentration and duration for the intended compartment. It is preferred that the agent supply containers and the flow control valves are not located in a fire zone.

(3) The brackets for mounting the containers and securing the discharge lines should be designed to withstand all loads to which they may be subjected due to recoil during discharge or any other applied load factor.

(4) The agent containers should be protected from extreme temperature excursions which could have an adverse effect upon the operation of the extinguishing system. Safe temperature limits for "unwinterized" carbon dioxide cylinders are approximately 0°F (-18°C) to 140°F (60°C). Safe limits for "CB" and CH_3Br spheres are approximately -65°F (-54°C) to 200°F (93°C). The cartridge detonators have a variable age-with-temperature limit and the manufacturer should be contacted for the latest information on installation and storage temperatures. Location of the container in the aircraft should take these temperature limits into consideration. *

* 595. <u>§ 29.1201 FIRE EXTINGUISHING SYSTEM MATERIALS</u>.

a. <u>Explanation</u>.

(1) Many different fire extinguishing agents are available for use in fire extinguishing systems. The choice of extinguishing agent should take into account the chemical reaction (if any) between the extinguishing agent and the materials utilized in the extinguishing system. If there are any incompatibilities, they should not create a hazard by creating volatile or toxic vapors or fumes which could feed a fire or cause injury to passengers, crew or other personnel.

(2) The fire extinguishing components in an engine compartment must be fireproof to ensure operation in the event of a compartment fire.

b. <u>Procedure</u>.

(1) Compliance with the requirements of § 29.1201(a) can be demonstrated by analysis, test, or a combination of both.

(2) Certification data submitted by the applicant should contain a listing of the chemical ingredients of the extinguishing agent and the other materials in the extinguishing system. These data should also show that the chemical reaction (if any) of these materials, when combined, does not create a hazard.

(3) Where chemical compounds exist and the chemical reaction is not predictable when two different compounds are combined, actual tests may be necessary to determine the hazard potential.

(4) Analysis, test, or a combination of both may be used to demonstrate compliance with the fireproof requirement for all fire extinguishing components located within the engine compartment. *

596. <u>§ 29.1203 (through Amendment 29-3) FIRE DETECTOR SYSTEMS</u>.

a. <u>Explanation</u>.

(1) Fire detection systems are required in turbine engine powered rotorcraft, Category A reciprocating engine powered rotorcraft and each Category B reciprocating engine powered rotorcraft where the engine displacement is greater than 900 cubic inches.

(2) This section specifies material, installation, and some operational requirements for fire detectors to ensure prompt detection of fire in the fire zones and other designated areas.

b. <u>Procedure</u>.

(1) The detector system should be designed for highest reliability to detect a fire and not to give a false alarm. It is desirable that it only responds to a fire and misinterpretation with a lesser hazard should not be possible. Engine overtemperature, harmless exhaust leakage, and bleed air leakage should not be indicated by a fire detector system. A fire detection system should be reserved for a condition requiring immediate measures such as engine shutdown or fire extinguishing. There are three general types of detector-procedure systems that are commonly used:

(i) A manual system utilizes warning lights to alert the pilot who then follows prescribed cockpit procedure as a countermeasure. A manual system is adequate for hazards in which a few seconds are not important.

(ii) There is also a semi-automatic system. Occasionally a helicopter becomes so complex that the emergency procedure exceeds reasonable expectations of the pilot. In such cases, psychology should be weighted against complexity, and "panic switches," combining multiple procedure functions, should be provided to simplify the mental demands on the pilot. Speed is gained by such designs for hazards which may need it.

(iii) The detector of an automatic system automatically triggers the appropriate countermeasures and warns the pilot simultaneously. Such a system should be carefully evaluated to assure that the advantages outweigh the disadvantages and potential malfunctions.

(2) Fires, or dangerous fire conditions can be detected by means of various existing techniques. The following is a partial list of available detectors:

(i) Radiation-sensing detectors.

(ii) Rate-of-temperature-rise detectors.

(iii) Overheat detectors.

(iv) Smoke detectors.

(v) CO detectors. *

* (vi) Combustible mixture detectors.

(vii) Fibre-optic detectors.

(viii) Ultraviolet.

(ix) Observation of crew or passengers.

(3) In many rotorcraft it is desirable to have a detection system which incorporates several of these different types of detectors. Radiation-sensing detectors are most useful where the materials present will burn brightly soon after ignition, such as in the powerplant accessory section. Rate of rise detectors are well-suited to compartments of normally low ambient temperatures and low rates of temperature rise where a fire would produce a high temperature differential and rapid temperature rise. It should be noted that under certain circumstances, where a relatively slow temperature increase occurs over a considerable period of time, a fire can occur without detection by rate of rise detectors. Overheat detectors should be used wherever the hazard is evidenced by temperatures exceeding a predicted, set value. Smoke detectors may be suited to low air flow areas where materials may burn slowly, or smolder. Fibre-optic detectors can be used to visually observe the existence of flame or smoke. The three major detector types used for fast detection of fires are the radiation-sensing, rate-of-rise, and overheat detectors. Radiation-sensing detectors are basically "volume" type which senses flame within a visible space. Overheat-fire detectors can be obtained in either "continuous" or "unit" type.

(4) The detector system should:

(i) Indicate fire within 15 seconds after ignition, and show which engine compartment in which the fire is located.

(ii) Remain on for the duration of the fire.

(iii) Indicate when the fire is out.

(iv) Indicate re-ignition of the fire.

(v) Not by itself precipitate or add to the potential of any other hazards.

(vi) Not cause false warnings under any flight or ground operating condition.

(5) Additional features of the detection system are as follows:

(i) A means should be incorporated so that operation of the system can be tested from the cockpit.

(ii) Detector units should be of rugged construction, to resist maintenance handling, exposure to fuel, oil, dirt, water, cleaning agent, extreme temperatures, vibration, salt air, fungus, and altitude. Also, they should be light in weight, small, and compact, and readily adaptable to desired positions of mounting. *

(iii) The detector system should operate on the rotorcraft electric system without inverters. The circuit should require minimum current unless indicating a fire or unless a monitoring system is in use.

(iv) Fixed temperature fire detectors should preferably be set at 100°F (37.7°C) to 150°F (65.6°C) above maximum safe ambient temperature, or higher when in compartments where extremely high rate of rise is normally encountered.

(v) Detector system components located within fire zones should be fire resistant.

(vi) Each detector system should actuate a light which indicates the location of the fire. If fire warning lights are used, they must be in the pilot's normal field of view.

(vii) Two or more engines should not be dependent upon any one detector circuit. The installation of common zone detection equipment prevents the detection system from distinguishing between the engine installations, necessitating shutting down more than one engine.

(6) The sensing portion of the fire detection system should not extend outside of the coverage area into another fire zone. Detectors, with the exception of radiation-sensing detectors, should be located at points where the ventilation air leaves compartments. If a reverse-flow cooling system is used, detectors should be installed at locations which are outlets under both flight and ground operating conditions. Stagnant air spaces should be avoided and the number of ventilation air exits should be kept to a minimum. The ventilation requirements of § 29.1187(e) must also be taken into consideration. Compliance with these recommendations allows the effective placement of a minimum amount of detectors, and still ensures prompt detection of fire in those zones. Radiation-sensing detectors should be located such that any flame within the compartment is immediately sensed. This may or may not be where the ventilation air leaves the compartment.

(7) Fire detectors must be installed in designated fire zones, the combustor, turbine and tailpipe sections of turbine installations.

(i) Engine Power Section (Combustor, Turbine and Tailpipe): This zone is usually characterized by predictable hazard areas which facilitate proper detector location. It is recommended that coverage be provided for any ventilating air outlet as well as intermediate stations where leaking combustibles may be expected.

(ii) Compressor Compartment: This is usually a zone of relatively low air flow velocities, but wide geographical possibility for fires. When fire detectors other than radiation-sensing detectors are used, detection at air outlets provides the best protection, and intermediate detector locations are of value only when specific hazards are anticipated.

(iii) Accessory Bullet Nose: Where such a compartment is so equipped that it is a possible fire zone, its narrow confines permit sufficient coverage with one or more detectors at the outlets. *

* (iv) Heater Detector Location: An overheat detector should be placed in the hot air duct downstream of the heater. If the heater fuel system or exhaust system configuration is such that it is a fire hazard, the compartment surrounding the heater should also be examined as a possible fire zone.

(v) Auxiliary Power Unit Detector Location: The use of a combustion-driven auxiliary power unit creates another set of typical engine compartments defined and treated as above. Some units are so shrouded with fireproof material that these compartments exist only within the confines of the shroud. They are still, however, fire zones and must have a detection system. *

597.-616. RESERVED.

SECTION 34. EQUIPMENT - GENERAL

617. § 29.1301 (through Amendment 29-24) FUNCTION AND INSTALLATION.

a. Explanation. It should be emphasized that this rule applies to each item of installed equipment which includes optional equipment as well as required equipment.

b. Procedure.

(1) Information regarding installation limitations and proper functioning is normally available from the equipment manufacturers in their installation and operations manuals. In addition, some other paragraphs in this AC include criteria for evaluating proper functioning of particular systems. (An example is paragraph 776 for avionic equipment.)

(2) This general rule is quite specific in that it applies to each item of installed equipment. It should be emphasized, however, that even though a general rule is relevant, a rule that gives specific functional requirements for a particular system will prevail over a general rule. Therefore, if a rule exists that defines specific system functioning requirements, its provisions should be used to evaluate the acceptability of the installed system and not the provisions of this general rule. It should also be understood that an interpretation of a general rule should not be used to lessen or increase the requirements of a specific rule. Section 29.1309 is another example of a general rule, and this discussion is appropriate when applying its provisions.

Then at sea level:

$$T_{1\ mar} = -(-40 - (0 + (-25 - (-25)))) = 40°F$$

At 5,000 feet:

$$T_{1\ mar} = -(-40 - (-9 + (-43 - (-45)))) = 33°F$$

At 10,000 feet:

$$T_{1\ mar} = -(-40 - (-21 + (-61 - (-59)))) = 17°F$$

At 15,000 feet:

$$T_{1\ mar} = -(-40 - (-32 + (-69.7 - (-65)))) = 3.3°F$$

At 20,000 feet:

$$T_{1\ mar} = -(-40 - (-40 + (-69.7 - (-69)))) = -.7°F$$

It can be seen that there is an acceptable margin at all altitudes up to and including 15,000 feet. However, at 20,000 feet, the margin is negative and the system fails.

(ii) Vibration tests. Normally, installed vibration tests are not necessary for equipment qualified in accordance with RTCA document DO-160A. This paper categorizes vibration tests according to installed rotorcraft equipment location such as fuselage, engine compartment, instrument panel, equipment rack, etc. However, installed equipment vibration tests may be necessary when it appears the equipment location environment may exceed the laboratory tested equipment vibration limits.

(iii) Altitude tests. If the equipment has been laboratory tested to the maximum certified altitude of the helicopter, installed altitude tests are unnecessary. The installed equipment must be either laboratory tested or tested in the rotorcraft to the maximum certified altitude of the helicopter.

(3) Lightning Strike Protection of Full Authority Digital Engine Controls.

(i) Explanation.

(A) The following discussion is written specifically for full authority digital engine controls (FADEC) with an alternate technology backup fuel control installed on rotorcraft with Category A engine isolation. The requirement for increased consideration of lightning strike encounter effects on avionic equipment and systems has been brought about by the increased use of avionics to perform functions, the failure or malfunction of which could result in a hazard to the helicopter. The susceptibility of current high technology avionic systems is increased by the use of large scale integration (LSI), very large scale integration (VLSI), and complementary metallic oxide silicon (CMOS) technologies which exhibit a greatly reduced tolerance to large amplitude, low energy electrical transients as compared to convential bipolar technology, and the reduced physical protection and electromagnetic shielding afforded aircraft avionic systems by the advanced technology composite airframe materials. Additionally, processor-based systems have the failure phenomenon of digital upset. A digital upset occurs when a system, perturbed by an electrical transient, ceases proper operation in accordance with its embedded software while suffering no apparent component or device damage.

(B) Since elements of electrical/electronic engine subsystems are typically spread throughout much of the helicopter, transients caused by lightning are coupled into the subsystem interface cables and may damage the system or cause upset. Effective lightning protection must be designed and incorporated into these systems. Reliance upon redundancy as a means of protection against lightning effects is generally not adequate because lightning electromagnetic fields and structural IR voltages usually interact (to some extent) with all electrical wiring aboard a helicopter.

(C) The testing and analysis outlined in this discussion are methods by which the FAA may be assured that when the helicopter experiences "the foreseeable operating condition" of a worst-case lightning strike encounter that the electronically controlled engines will continue to "perform their intended function" and therefore be in compliance with § 29.1309(a) as installed.

(D) The definition of what constitutes a full authority engine control is not at this time clearly defined. However, it has been accepted in past certification that any control which relies upon the electronics for the function on which Civil Certification or Military Qualification is based (e.g. rotor speed governing) is a full authority control, regardless of the backup control mode provided. If engine certification or qualification can be achieved without the electronic control which is subsequently added to achieve improved operational efficiency in the aircraft, the control is "supervisory." However, if the controls are used in a multiengine helicopter, a common failure caused by a lightning strike could result in simultaneous failures which would cause a reduction in power greater than the loss of one engine. This would also be considered "full authority."

NOTE: If OEI ratings are approved, cumulative loss of power from all engines should be limited to allow flight manual performance based on OEI ratings.

(ii) Procedure. Although not a regulatory requirement, it is recommended that a formal written certification plan be used to assure regulatory compliance. The use of this plan is beneficial to both the applicant and the FAA because it identifies and defines an acceptable resolution to the critical issues early in the certification process. These are the usual steps to be followed when utilizing a certification plan:

(A) Prepare a certification plan which describes the analytical procedures and/or the qualification tests to be utilized to demonstrate protection effectiveness. Test plans should describe the helicopter and FADEC system to be utilized, test drawing(s) as required, the method of installation that simulates the production installation, the lightning zone(s) applicable, the lightning simulation method(s), test voltage or current waveforms to be used, diagnostic methods, and the appropriate schedules and location(s) of proposed test(s).

(B) Obtain FAA concurrence that the certification plan is adequate.

(C) Obtain FAA detail part conformity of the test articles and installation conformity of applicable portions of the test setup.

(D) Schedule FAA witnessing of the test.

(E) Submit a final test report describing all results and obtain FAA approval of the report.

(iii) Definition of Environment. The SAE AE4L Committee report dated June 20, 1978, is an acceptable criteria to define the worst-case lightning strike which may be encountered by the helicopter in service. An additional explanation of the lightning environment may be found in FAA Report DOT/FAA/CT-83/3, "User's Manual for AC-20-53A, Protection of Airplane Fuel Systems Against Fuel Vapor Ignition Due to Lightning," paragraph 5.2, "Establishment of the Lightning Environment." This definition of the environment is considered the minimum acceptable standard for a FADEC with an alternate technology backup. For a full fly-by-wire system, with no mechanical backup or other future technology designs, this criteria may not be sufficient to describe a worst-case lightning encounter. On 3/5/90, the FAA Advisory Circular AC 20-136, "Protection of Aircraft Electrical/Electronic Systems Against the Indirect Effects of Lightning" was issued. Appendix 3 of that document contains an updated quantification of the severe natural lightning environment. It is recommended that for new designs and applications after 3/5/90, this revised definition of the lightning be used. *

(iv) Certification Plan. The following subjects are not intended to provide a complete list of the items which should be included in the certification plan, but rather highlight some of the areas which should receive consideration. The certification plan should address the total protection which is required to allow the FADEC to continue to operate properly when the helicopter experiences a worst-case lightning strike encounter.

(A) Determination of Lightning Strike Attachments. Determine the locations on the helicopter where lightning strike attachment is likely to occur and the portions of the airframe through which currents may flow between attachments. The main and tail rotors are recognized as likely attachment points; however, consideration should be given to all possible attachment points. The swept stroke phenomenon may not exist for all lightning strike encounters due to the fact that the helicopter may be airborne with little or no airspeed.

(B) Establish the Lightning Environment. Establish the components of the total lightning event to be considered. These are the currents and voltages which are described in the definition of the environment.

(C) Full-Level, Complete Vehicle Testing. In accordance with traditional FAA Policy, the demonstration that the FADEC installed in a complete type design helicopter will continue to operate properly when exposed to a worst-case lightning strike is sufficient to demonstrate compliance with § 29.1309(a). Because of the difficulties involved in utilizing this type of an approach, it is generally not used.

(D) Analytical Processes. A description should be given in the certification plan of the analytical process and/or certification tests to be utilized to demonstrate protection effectiveness. Typically, the certification plan will include a combination of analysis and tests. (Analytical techniques are most often utilized to predict the levels of lightning-induced transients in interconnecting wiring.) In most cases, successful analyses are based upon well-defined geometrical or electrical parameters such as structural dimensions and materials resistivities. When electrical characteristics of structural materials are not well established, development tests are often utilized to obtain this data which is subsequently utilized in an analysis. In more complex structures and/or electrical/electronic system installations, it is sometimes difficult or impossible to define the problem in terms that can be analyzed. In these cases, development or verification testing is often relied upon. The purpose of the certification plan is to show how developmental tests, analyses, and verification tests are combined to demonstrate protection design adequacy. In certain cases, previously verified designs can be incorporated and their adequacy confirmed by reference to previous verifications. Such reference should also be incorporated in the certification plan.

(1) The verification testing should be conducted on a system which simulates as closely as possible the installed configuration. As few items as possible of actual hardware should be simulated.

(2) The use of various analytical processes usually requires that the system component tolerance is established. The SAE AE4L Committee Report No. AE4L-81-2 is the recommended reference to be used for the testing accomplished to determine these tolerances. The testing which is performed to determine the tolerance level of the control computer should include a consideration for the occurrence of a nonrecoverable digital upset. One method to provide this consideration is to have the unit powered and the processor operating normally under software control (usually this should be the exact software for which approval is sought) when the test is performed. If strike testing is used, then several shots should be made to develop enough data to provide a reasonable confidence level. It is an acceptable procedure for the engine manufacturer, while he is obtaining his type certificate, to accomplish this bench testing to determine the level of tolerance of the FADEC system components to lightning encounter indirect effects. This approach has the advantages that the bench tests are not necessarily required to be repeated when the engine is installed in a different airframe. This recommendation is not meant to add a requirement to the engine manufacturer but to propose a more efficient method of certification. If this tolerance was not determined by the engine manufacturer, the applicant installing the FADEC in a helicopter would be expected to furnish this data.

(3) For complete airframe verification testing, a minimum level of at least 4KA peak and a current rise time of 2KA/microsecond are recommended. It is often difficult to obtain valid results at lower levels due to poor signal-to-noise ratios. When complete vehicle testing is accomplished at some lower level, or through some alternate test technique such as low level swept CW testing, consideration should be given to nonlinear airframe response, diffusion effects, and alterations in current paths caused by arcing and flashover.

(4) As with any analytical method, it is prudent to include a margin of safety to account for the uncertainties involved in the analytical and testing processes. A level of 6 dB is recommended for those analyses which are confirmed by the use of reduced level, full-scale vehicle testing. This safety margin is the difference between the airframe installed system responses and

<u>NOTE</u>:

If a quantitative analysis is used to help show compliance with Federal Aviation Regulations for equipment which is installed and required only for a specific operating condition for which the rotorcraft is thereby approved, credit may not be taken for the fact that the operating condition does not always exist. Except for this limitation, appropriate statistical randomness of environmental or operational conditions may be considered in the analysis. (However, the particular condition and probability of that condition should be agreed to with the FAA.)

The three probability terms defined in paragraph d(2)(ii) above are intended to relate to an identified failure condition resulting from or contributed to by the improper operation or loss of a function or functions. These terms do not define the reliability of specific components or systems.

e. <u>Documentation</u>. All laboratory, ground and flight tests, and failure analyses, must be documented in sufficient detail to show compliance with § 29.1309 and included in the type design file. Section 21.31(a) provides the regulatory basis for requiring this documentation. If the applicant elects to use a numerical reliability/probability analysis it must also be documented in sufficient detail.

f. <u>Computer Software</u>.

(1) If implementation of the equipment, systems, or installations includes computer software, the RTCA Document DO-178A "Software Considerations in Airborne Systems and Equipment Certification," dated March 22, 1985, is the recommended standard to be used for the approval of system software. This document defines three levels of software; i.e., levels 1, 2, and 3. The level of the software is related to the consequence of a system malfunction caused by an error in the software. The criticality categories are:

(i) Critical - Functions for which the occurrence of any failure condition or design error would prevent the continued safe flight and landing of the aircraft.

(ii) Essential - Functions for which the occurrence of any failure condition or design error would reduce the capability of the aircraft or the ability of the crew to cope with adverse operating conditions.

(iii) Nonessential - Functions for which failures or design errors could not significantly degrade aircraft capability or crew ability cue.

(2) The different software levels are related to the criticality categories. Level 1 software, the most error-free software, is usually required for critical functions. However, level 1 software may sometimes be reduced by system architecture techniques such as the use of redundant (dissimilar) software performing the same function. Level 2 software is required for essential functions. It should be noted that those systems, equipment, and installations, with functioning required by 14 CFR subchapter C, are by this definition essential functions. The criticality of the function should be determined by the use of a fault/failure or hazard analysis. The Society of Automotive Engineers Aeronautical Recommended Practice Document Nos. 926A and 1834 are the recommended reference for performing these analyses.

CAVEAT: The user of DO-178A is cautioned by a caveat in Chapter 3 that for a certain class of systems, the techniques in DO-178A, level 1, software are not by themselves sufficient consideration for reliance on system software to preclude a catastrophic event. Additional considerations are required with this class of system for software verification and validation (V&V) in addition to those required for DO-178A, level 1. This class of systems is one which has been called "full flight regime critical." An example of such a system is a fly-by-wire flight control. This system must perform its intended function through the full flight regime to provide for the continued safe flight and landing of the rotorcraft. For this system, software and system level validation beyond the scope of DO-178A are required. Also, DO-178A cautions the user against the assignment of probabilities of residual software errors. The conclusion of Special Committee No. 152 (RTCA committee that wrote DO-178A) was that the present methods available for assigning "reliability" numbers to software do not yield credible results for certification purposes.

g. High Intensity Radiated Fields (HIRF).

(1) Explanation. A regulatory project is active to add requirements for the protection of aircraft electrical and electronic systems from the effects of the HIRF environment. This effort is the result of technological advances in airframe and electronic systems design and a concurrent increase in the levels of radiated power in the aircraft environment. These changes have raised vulnerability to the electromagnetic environment of the electrical and electronic systems which perform critical and essential functions. In current type certification programs involving advanced electrical and electronic systems the FAA has adopted special conditions to provide an adequate level of safety.

(i) The special conditions are directed toward the operation and operational capability of the installed electrical and electronic systems that perform critical functions. The applicant may demonstrate that these systems are not adversely affected when the aircraft is exposed to the HIRF environment, or as an alternative a laboratory test may be conducted, as discussed in the "Discussion" associated with each special condition. The laboratory tests would be conducted at a peak electromagnetic strength of 100 or 200 volts per meter, as appropriate, in a frequency range of 10 KHz to 18 GHz.

(ii) An acceptable definition of the HIRF environment is included in an FAA Aircraft Engineering Division Memorandum dated December 5, 1989, (Subject: High Energy Radiated Electromagnetic Fields (HERF) Interim Policy Guidelines on Certification Issues).

(iii) If the laboratory test alternative is selected the 100 volts/meter level is considered appropriate for a function that is critical during IFR operations and the 200 volts/meter level is considered appropriate for a function that is critical during VFR operations. This is because the minimum en route altitude for IFR flight is 1,000 feet or 500 feet (FAA or ICAO), and helicopters operating VFR can and do operate regularly at lower altitudes. The attitude system is an example of a system performing a critical function during IFR operation. A full authority digital engine control (FADEC) system is an example of a critical function during VFR and IFR operation. *

* (2) Procedure. It is recommended that the applicant present a plan to the cognizant FAA Aircraft Certification Office (ACO) for approval, outlining how the compliance with the HIRF requirements will be attained. This plan should also propose a pass/fail criteria for the operation of critical systems in the HIRF environment.

(i) A preliminary hazard analysis should be performed by the applicant for approval by the cognizant FAA ACO to identify electrical and/or electronic systems that perform critical functions. The term "critical functions" means those whose failure would prevent the continued safe flight and landing of the rotorcraft.

(ii) The systems performing critical functions that are identified by the preliminary hazard analysis are candidates for the application of HIRF requirements. A system may perform both critical and non-critical functions; however, the HIRF requirements only apply to critical functions. If redundant systems are used, all systems should be subjected to test/analysis for the HIRF requirements.

(iii) RTCA-DO-160C, Section 20 is an appropriate reference for laboratory test procedures. In addition a separate advisory circular and users guide on the subject of HIRF is being drafted for the FAA by the SAE AE4R Subcommittee.

622.-631. RESERVED.

655. <u>§ 29.1357 (through Amendment 29-19) CIRCUIT PROTECTIVE DEVICES</u>.

a. <u>Explanation</u>. Circuit protective devices are normally installed to limit the hazardous consequences of overloaded or faulted circuits. These devices are resettable (circuit breakers) or replaceable (fuses) to permit the crew to restore service when nuisance trips occur or when the abnormal circuit condition can be corrected in flight.

b. <u>Procedure</u>.

(1) Overvoltage protection is specifically required for Category A rotorcraft. For Category B rotorcraft, the possible types of operation should be considered in combination with the presence of an overvoltage condition in the generating system. The regulatory requirement to support this assumption is § 29.1309(b). If the presence of an overvoltage condition in the electrical system will not cause a hazard to the rotorcraft, the electrical system could be approved for Category B without overvoltage protection. If a hazardous condition will result from the overvoltage condition, then overvoltage protection must be provided.

(2) Automatic reset circuit breakers, which automatically reset themselves periodically, should not be applied as circuit protective devices. If an abnormal circuit condition cannot be corrected in flight, the decision to restore power to the circuit involves a careful analysis of the flight situation. The necessity of the circuit for continued safe flight should be weighed against the hazards of resetting on a possibly faulted circuit. Such an evaluation is properly an aircraft crew function which cannot be performed by automatic reset circuit breakers. To assure crew supervision over the reset operation, circuit protective devices should be of such design that a manual operation is required to restore service after tripping. Circuit breakers must be designed such that the tripping mechanism cannot be overridden by the operating control, and these circuit breakers are known as the "trip free" type.

(3) Automatic reset circuit breakers may be used as integral protectors for electrical equipment (e.g., thermal cutouts) provided that circuit protection is also installed to protect the cable to the equipment.

(4) If the installation of a system is required as a prerequisite to showing compliance with the regulations, it is generally considered to be essential to some phase of flight or it would not be required. It follows from this that the circuit protective device associated with those systems is generally considered to be essential for safety in flight and should therefore be accessible to the crew in the cockpit. This includes the basic electrical system, the distribution system, and utilization systems that are required. Some examples of required utilization systems are those specified by §§ 29.1303, 29.1305, 29.1307, 29.1381, 29.1383, 29.1385, 29.1401, and 29.1431. Where continued safe flight to the destination is considered to be sufficiently assured, certain required circuits have been excepted from being accessible to the crew in the cockpit. Voltmeter and ammeter circuit protective devices are examples of ones that have been excepted. Some utilization systems, although not specifically required by FAR 29, may be required because of the particular design presented for certification. Circuit protective devices for systems in this category are considered to be required and must be accessible.

(5) The following are considered to be acceptable compliance with the "readily reset" provision of § 29.1357(d).

(i) For a crew of two (2) pilots, it is satisfactory for one of the crewmembers to move his seat and loosen his shoulder harness in order to properly identify and reset or replace a circuit protective device. It is not satisfactory for one of the crewmembers to leave his crew station to reset the circuit protective device.

(ii) For a single pilot situation, with the seat belt and shoulder harness normally adjusted, the circuit protective device location should allow for identification of the opened circuit protector and reset capability while the pilot is flying the rotorcraft.

(6) If fuses are used, there should be spare fuses for use in flight equal to at least 50 percent of the number of fuses for each rating required for complete circuit protection. This only applies to fuses used to protect systems that are required to show compliance with the regulations. Spare provisions need not be made for nonrequired convenience type installations although it is encouraged. The spare fuses should be stored in a location where they are readily accessible to the crew. If not directly visible to the crew, information regarding location of the spare fuses should be provided. One acceptable location is on the fuse panel in a holder with no wire terminations and identified "spare" with the "size."

(7) Refer to paragraph 777 of this advisory circular for specific tests of circuit protection for the total electrical system.

656. § 29.1359 (through Amendment 29-19) ELECTRICAL SYSTEM FIRE AND SMOKE PROTECTION.

a. Explanation. This regulation requires that all electrical system components meet the applicable fire and smoke protection provisions of §§ 29.831 and 29.863, and further requires that certain items in designated fire zones must be at least fire resistant. This regulation becomes very significant when failure conditions are considered, and in accordance with the provisions of § 29.831 "reasonably probable failures" must be considered when assuring compliance.

b. Procedures.

(1) When selecting a type of wire, the burning characteristics of that wire are important. Both composition and quantity of resultant smoke and fumes should be considered. The impact of the smoke and fumes on the aircraft cabin occupants should be accounted for.

(2) Wire qualified to MIL-W-25038 is normally used in circuits that "must be at least fire resistant." Wire qualified to other specifications may also be satisfactory; however, the provisions of the other specifications should be compared to the provisions of MIL-W-25038 to assure the critical areas are not compromised.

(3) Electrical connectors that are located in a designated fire zone and are used in emergency procedures should be at least fire resistant and capable of maintaining the integrity of the circuit. When evaluating these connectors, careful attention should be directed to the entire connector - the contact, the insert, and the shell.

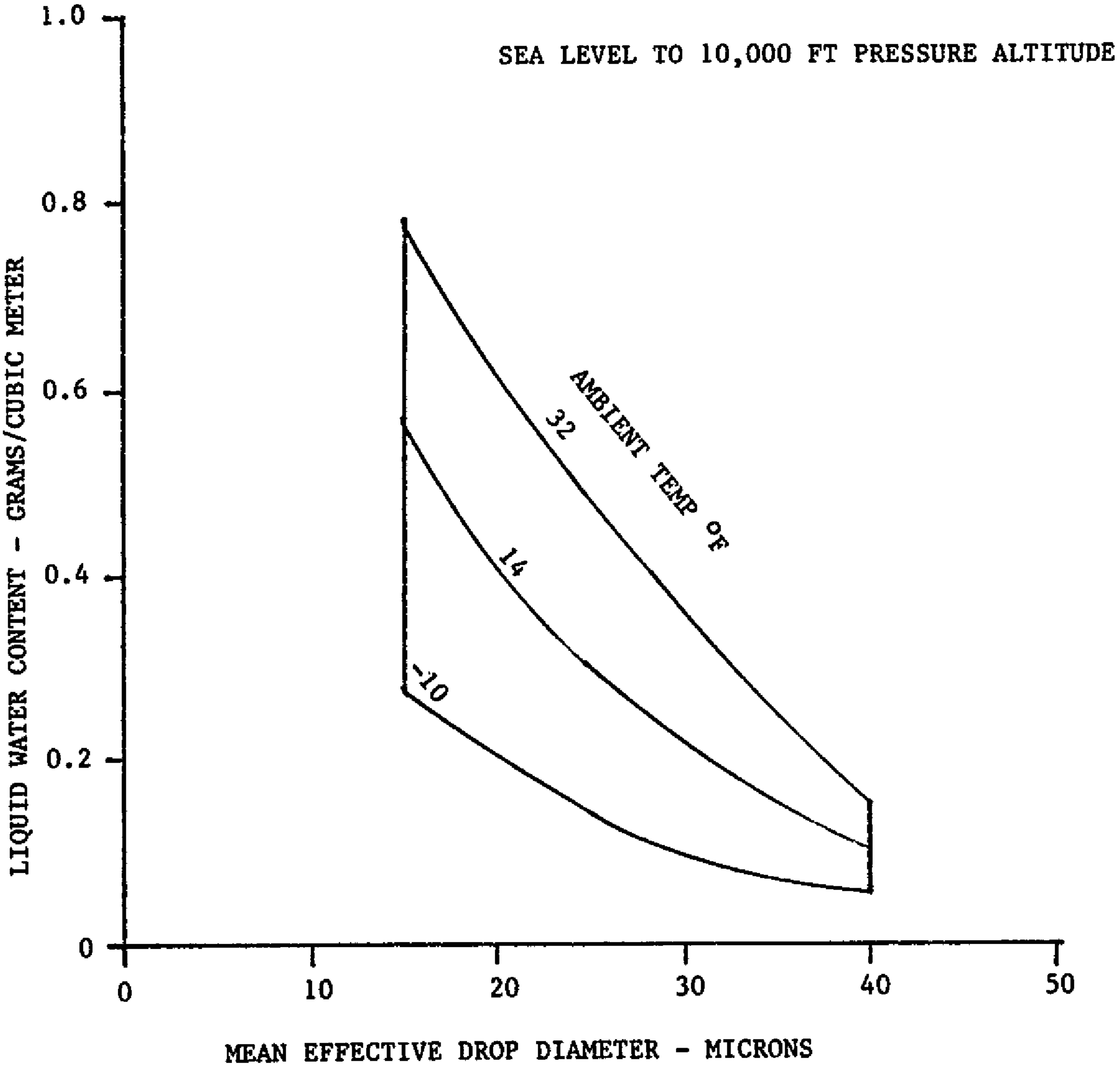

FIGURE 693-3. CONTINUOUS ICING - LIQUID WATER CONTENT VS. DROP DIAMETER *

Figures 693-1 through 4 represent one approach to a 10,000-foot altitude limit. See paragraph b(5)(iii) for a

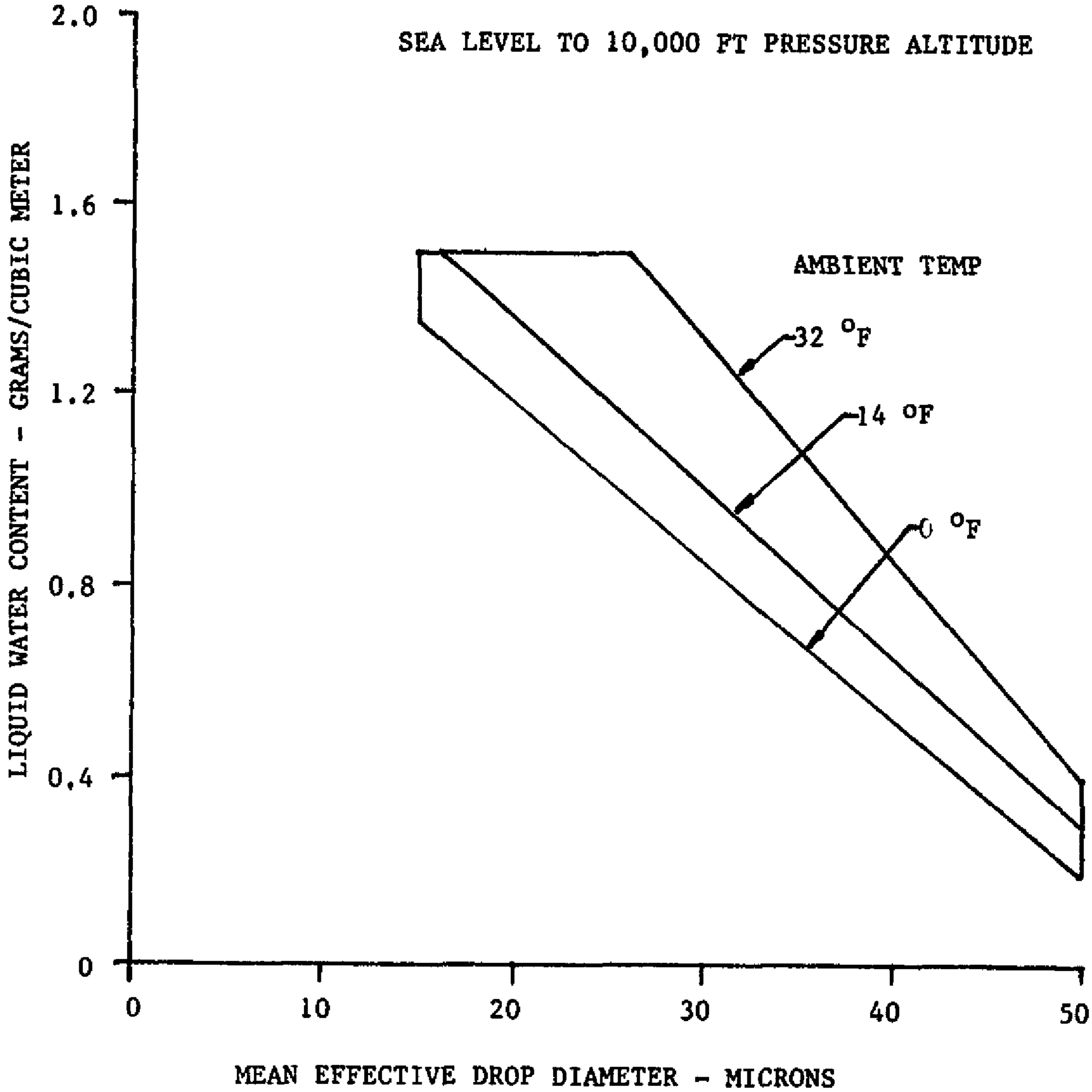

FIGURE 693-4. INTERMITTENT ICING-LIQUID WATER CONTENT VS DROP DIAMETER

Figures 693-1 through 4 represent one approach to a 10,000-foot altitude limit. See paragraph b(5)(iii) for a discussion of this approach.

694.-701. RESERVED.

(4) Wire insulated with KAPTON® polyimide film manufactured to MIL-W-81381A, has been used in aeronautical products with varying degrees of success. The U.S. Navy had such a bad service history with KAPTON® insulated interconnect wire in aircraft that in the mid-1980's the Navy no longer allowed the use of KAPTON® insulated wire. Although the FAA has taken no such action, the use of KAPTON® insulated wire requires very special handling. The following areas should be observed when utilizing KAPTON® insulated wire:

(i) The instructions in the KAPTON® wire "Handling Manual" should be strictly followed. This manual may be obtained from E. I. Du Pont de Nemours and Company, Polymer Products Department, Industrial Film Division, Wilmington, Delaware 19898.

(ii) Use in special wind and moisture problem (SWAMP) areas, such as wheel wells, usually requires additional protection for the cable bundles.

(iii) The wire should not be exposed to a combination of either high stress (U.V. or physical) in the presence of water, high humidity, or high PH factor liquids.

(iv) The stiffness and permanent set (memory) of KAPTON® may cause chafing in unrestrained bundles or where KAPTON® insulated wire is bundled with wires of other insulation types.

(v) Care should be exercised in the stripping, stamping, and terminating of KAPTON® insulated wires.

NOTE: KAPTON® is a registered trademark of E. I. Du Pont de Nemours and Company. *

657. <u>§ 29.1363 (through Amendment 29-24) ELECTRICAL SYSTEMS TESTS</u>.

a. <u>Explanation</u>. Most of this rule is self-explanatory. Since other regulatory paragraphs also contain requirements regarding functioning and malfunctioning of the electrical system, a recommended test procedure has been included in paragraph 777 of this AC instead of being made a part of this paragraph.

b. <u>Procedures</u>.

(1) Reference paragraph 777 of this AC for a recommended test procedure.

(2) When simulating the electrical characteristics of the distribution system wiring, emphasis should be placed on duplicating the type, gage, and length of the wiring being evaluated. As much as possible, cable bundling and grounding considerations should also be duplicated.

(3) Most laboratory test connected loads would normally be in the form of load banks rather than providing the actual aircraft system. If load banks are used during laboratory testing, additional consideration should be given to these loads when an actual aircraft installation is available.

(4) Limited aircraft testing should also be accomplished to verify that the response of the laboratory drives does adequately simulate the response of the rotorcraft engines under normal and malfunction conditions.

658.-667. <u>RESERVED</u>.

SECTION 37. LIGHTS

668. § 29.1381 (through Amendment 29-19) INSTRUMENT LIGHTS.

a. Explanation. This section provides minimum performance standards for the instrument lighting system. Section 29.1309(b) is used to evaluate the malfunction aspects of the system. If appropriate, § 29.1309(a) is used to evaluate the equipment under environmental considerations.

b. Procedures.

(1) The overall instrument lighting system should be designed and installed such that single failures that occur will not result in the loss of both primary and secondary (backup) lighting for any instrument or area of the cockpit. In some instances, the system is divided such that the controls for the pilot's panel are separate from the copilot's panel and both of these are separate from the center panel. The ideal is to divide the system such that the impact of single failures will be minimized.

(2) Secondary (backup) instrument lighting should be provided, and this is accomplished in some instances by eyebrow lights. A system that provides general cockpit lighting from a source in the aft area of the cockpit is normally not acceptable since normal positioning and movement of the crew will block this type of light.

(3) The standard does not specify any color requirements for instrument lighting. White is normally provided. The color provided should ensure that the color coding of the instruments is readily identifiable.

(4) The final installed system should be evaluated by a flight test pilot. An actual night flight should be conducted for initial certification of an aircraft. In some instances, the vibration characteristics and other flight induced factors have been demonstrated to seriously affect the pilot's ability to see in the cockpit environment at night. Evaluations following modifications may be conducted with a darkened cockpit on the ground. It should be verified that direct rays are shielded from the pilot's eyes, and that objectionable reflections do not exist. The pilot should also assume failures of various controls, electrical busses, etc., to account for all appropriate failures.

(5) In some instances manufacturers have provided high intensity instrument lighting systems as an option associated with IFR approvals. If provided, this capability should be included in the overall evaluation of the instrument lighting system.

669. § 29.1383 (through Amendment 29-19) LANDING LIGHTS.

a. Explanation. This section provides minimum performance standards for the installation and normal operation of the landing lights. Certification to this standard is all that is required for approval of the helicopter; however, the different operating rules should also be reviewed since they may contain additional requirements. The malfunction considerations are based on the provisions of § 29.1309(b).

SECTION 39. MISCELLANEOUS EQUIPMENT

702. § 29.1431 ELECTRONIC EQUIPMENT.

a. Background. This section contains some specific requirements for electronic equipment in the rotorcraft. The principal requirements of this section are that radio and navigation equipment must be free from hazards, both in themselves and in their effect on any other items installed in the rotorcraft, and that operation of the radio and navigation equipment does not interfere with operation of any other required avionics.

b. Procedures. In showing compliance with this section, tests and analysis should be performed as necessary to determine that:

(1) All radio and navigation equipment is installed and operated in such a manner that it does not result in hazards to the rotorcraft. It also should not have an effect on any other components of the rotorcraft such that a hazardous condition is created. Note that consideration should be given to the effects of critical environmental conditions.

(2) All radio and navigation systems and equipment should be installed and operated in a manner that will not have a detrimental effect on the proper functioning of any electronic equipment or system required by the FAR. It should be noted that §§ 29.1301 (ref. paragraph 617) and 29.1309(b) through (d) (ref. paragraph 621) apply to all installed equipment and systems and § 29.1309(a) applies to all systems and equipment required by Parts 21 through 49. As an example of showing compliance with this section, consider a high frequency radio (HF) system installation. The first thing to determine is that the installation and operation of the HF system cannot create a hazard. Consideration may be necessary in hazardous situations such as precipitation on the antenna. Next, it should be determined that the operation of the HF does not cause interference to a system whose functioning is required by the FAR. An example of unacceptable interference would be if operating the HF transmitter caused one of the navigation radios to malfunction.

(3) Finally, it should be determined that other systems do not interfere with the HF system. Additional guidance on the testing of avionics equipment and installation is contained in paragraph 776.

* 703. § 29.1433 VACUUM SYSTEMS.

a. Explanation. Vacuum systems have been utilized on some helicopters to provide an energy source for the flight instruments. This specific rule addresses the potential hazards which are peculiar to vacuum system installations. The possible fire hazards presented by these systems are of particular concern.

b. Procedure. The following items should be specifically addressed when evaluating a vacuum system installation:

(1) Pressure and Temperature Protection. The high pressure outlet of the vacuum pump should have a means to automatically relieve the pressure if it becomes excessively high or the air temperature becomes excessively hot.

(2) Fire Hazard Protection. The components of the vacuum system which are mounted in a designated fire zone should be fire resistant. This includes engine or transmission driven pumps if they are in a fire zone. The discharge side of the pump may emit flammable fluids. This discharge side of the pump, along with its associated lines and fittings, should meet the criteria in paragraph 585 of this AC.

704. § 29.1435 (through Amendment 29-19) HYDRAULIC SYSTEMS.

a. Reference Regulations. The following sections of Part 29 are either incorporated in the provisions of § 29.1435 or are otherwise applicable to hydraulic system design:

(1) Section 29.695. Paragraph 287 of this advisory circular covers power boost and power operated control systems.

(2) Section 29.861. Paragraph 361 of this advisory circular covers fire protection of structure, controls, and other parts.

establish that limit loads are not exceeded where the aircraft loads are in the linear range of loading (i.e., aerodynamic coefficients for the flight condition are adequately established and no significant nonlinear air loadings exist). If significant nonlinear effects could exist, flight load survey measurements may be necessary to substantiate that the limit loads are not exceeded. The power for climb should be the most critical of: (1) that used in the performance climb demonstrations; (2) that used in the longitudinal stability tests; or (3) that actually used for operational climb speeds. The altitude loss should be measured.

(3) Maneuvering flight. Malfunctions should also be induced into the automatic pilot system similar to paragraph c(1). When corrective action is taken, the resultant loads and speeds should not exceed the values contained in paragraph c(2). Maneuvering flight tests should include turns with the malfunction induced when maximum bank angles for normal operation of the system have been established and in the critical aircraft configuration and/or stages of flight likely to be encountered when using the automatic pilot. The altitude loss should be measured.

(4) Oscillatory tests.

(i) An investigation should be made to determine the effects of an oscillatory signal of sufficient amplitude to saturate the servo amplifier of each device that can move a control. The investigation should cover the range of frequencies which can be induced by a malfunction of the automatic pilot system and systems functionally connected to it, including an open circuit in a feedback loop.

(ii) The results of this investigation should show that the peak loads imposed on the parts of the aircraft by the application of the oscillatory signal are within the limit loads for these parts.

(iii) The investigation may be accomplished largely through analysis with sufficient flight data to verify the analytical studies or largely through flight tests with analytical studies extending the flight data to the conditions which impose the highest percentage of limit load to the parts.

(iv) When flight tests are conducted in which the signal frequency is continuously swept through a range, the rate of frequency change should be slow enough to permit determining the amplitude of response of any part under steady frequency oscillation at any critical frequency within the test range.

(5) Recovery of flight control. Recovery of the rotorcraft should be accomplished by the pilot by first overpowering the malfunctioning autopilot and then disconnecting it. The control to disconnect the autopilot should be easily available to the pilot who is now resisting the malfunctioning force of the autopilot. It is recommended that the disconnect button be placed on the cyclic control. It should be red and conspicuously marked "Autopilot Disconnect." The pilot should be able to return the rotorcraft to its normal flight attitude under

(v) Most of the autopilots which have been approved utilize series actuators or servos such as those required for a SAS. However, this does not preclude the approval of an autopilot which uses outer loop parallel actuation. This type of autopilot may be particularly helpful in a VFR aircraft.

(2) Cockpit controls. Evaluation of the cockpit controls should include the following items:

(i) Location of the automatic pilot system controls are such that their operation is properly labeled and is readily accessible to the pilot(s).

(ii) Annunciator colors conform to the colors specified in § 29.1322 (ref. paragraph 633 of this AC).

(iii) A determination is made that the controls, control labels, and placards are readable and discernible under all expected cockpit lighting conditions.

(iv) Motion and effect of the autopilot cockpit controls should conform with the requirements of § 29.779.

(v) Any disconnect of the autopilot should be annunciated.

c. Malfunction evaluations. To preclude hazardous conditions which may result from any failure or malfunctioning of the autopilot the following failures should be evaluated. This evaluation should also account for any hazards which also might be caused by inadvertent pilot action. The guidance in paragraph 775 of this AC should be used to determine the appropriate reaction times of the human pilot to an autopilot malfunction. *

(1) Climb, cruise, and descent flight regimes. The more critical of the following should be induced into the automatic pilot system.

(i) A signal about any axis equivalent to the cumulative effect of any single failure, including autotrim (if installed).

(ii) The combined signals about all affected axes, if multiple axes failures can result from the malfunction of any single component.

(2) Limit Loads. The simulated failure and the subsequent corrective * action should not create loads in excess of structural limits or result in dangerous dynamic conditions or deviations from the flight path. Additional guidance regarding the method of determining pilot recognition times and reasonable flight path deviation due to these simulated failures is contained in paragraph 775b(6) of this AC. Resultant flight loads outside the envelope of zero to 2g will be acceptable provided adequate analysis and flight test measurements are conducted to establish that no resultant aircraft load is beyond limit loads for the structure, including a critical assessment and consideration of the effects of structural loading parameter variations (i.e., center of gravity, load distribution, control system variations, maneuvering gradients, etc.). Analysis alone may be used to

(3) Section 29.863. Paragraph 362 of this advisory circular flammable fluid fire protection.

(4) Section 29.1183.

(5) Section 29.1185.

(6) Section 29.1189.

(7) Section 29.1309. Paragraph 621 of this advisory circular requirements for functioning and reliability, and prevention of hazards malfunctions or failures occur.

(8) Section 29.1322. Paragraph 633 of this advisory circular warning, caution, and advisory lights.

b. System Design. It is assumed that the hydraulic system is to to operate the primary control system of the rotorcraft and the rotorcr safely operated without the hydraulic system.

(1) Section 29.1309, paragraphs (a) and (b), provides for fun reliably under any foreseeable operating condition and prevention of ha any malfunction or failure.

(2) The substantiating data should include a failure analysis considers every possible system component failure, such as (but not lim ruptured lines, pump failure, regulator failure, ruptured seals, clogge broken pilot valve connections, etc.

(3) The requirements of § 29.1309(a) and (b) are met by dual hydraulic systems from the reservoir, hydraulic pump, regulator, connec and hoses through the actuators. There must be no commonality in the fluid-containing components. A break in one system should not result i in the remaining system.

(4) The pumps should be separated as far as practicable; i.e. sides of the rotor drive transmission, on separate engines, or one pump and the other on the rotor drive transmission. The tubing and hoses sh routed with as much physical separation as practicable. The purpose of separation is to prevent total loss of the hydraulic systems in the eve malfunction such as fire, or rotor burst wherein one projectile could d systems.

(5) Dual actuators must be designed to assure that any single such as a cracked housing, broken interconnecting input, or output link result in loss of total hydraulic system function.

(6) If the assumption under (b) above does not apply and the control the rotorcraft without undue fatigue after loss of the hydrauli then a single hydraulic system is acceptable.

ating system required by § 29.1435,
d with a dial, vertical scale, or digital
enable the crew to detect pressure trends.
circular concerns § 29.1322 regarding proper colors
used to supplement the indicating system.

or a combination of analysis and tests must be included in
file to show compliance with paragraphs (a)(1), (a)(2), and

caution should be exercised to assure that control input forces
connection to the actuator pilot valves do not exceed their
Consideration should be given to the most adverse tolerance buildup
ation and control system rigging.

The substantiating data should show that the hydraulic components will
intended function reliably under the most adverse continuous and
ironmental conditions to which they are exposed. These variables
e not limited to temperature, humidity, vibration, altitude, and
aph 621, b(2)(i), of this advisory circular is a method of temperature
cover the entire operating temperature envelope being certified.

The system component strength must be sufficient for its material
o exceed the number of cycles imposed by pump ripple pressure.

llation Precautions and Fire Protection.

All components and tubing routed through fire zones may be designed to
e fire protection requirements of §§ 29.1183, 29.1185, and 29.1189.
ive, a fireproof shield may be used around the component to be
e component should be sufficiently protected to assure fluid leakage
and fuel the fire.

All hydraulic lines should be sufficiently isolated from the engine,
s, environmental control unit, oil cooler, or other heat source to
d line life.

If flammable hydraulic fluid is used, the hydraulic components should
om ignition sources to assure that failure of any of the hydraulic
l not result in a fire or explosion. In the case of electrical
es in the proximity of hydraulic components, the electrical equipment
etically sealed or otherwise substantiated as not being an ignition
rence paragraph 621b(1)(i) of this advisory circular.)

The installation detail should be thoroughly reviewed for adequacy of
and clearance from sharp edges. As much physical separation as
d be provided between hydraulic lines and electrical cables.

While the control system is being moved from stop to stop, observation
to determine that hose flexing and tube bending is minimized.

d. Testing.

(1) Individual components should be substantiated by either vendor's or primary manufacturer's laboratory test reports. These tests should establish performance ratings such as pressures, flow rates, environmental capability, etc., to be approved.

(2) After the total system is installed, ground tests should be conducted to assure the system performs as intended and that each component is functioning within its design rating.

(3) If the total system design permits each combined independent power source and actuator to be disabled by shutoff valves, engine shutdown, etc., each combination should be disabled and the remaining combination verified to perform the necessary control functions. The test should be accomplished again with the functioning combination disabled and the disabled combination functioning. These tests should be accomplished first by ground tests, then repeated in flight.

(4) Temperature and pressure instrumentation should be provided at the critical points in the system to meet the provisions of d(2) above. Temperature results should be corrected for hot day conditions. (Paragraph 621b(2)(i) of this advisory circular gives a recommended procedure.)

(5) All controls should be cycled throughout their complete range of travel while accomplishing d(2) above.

(6) Satisfactory hydraulic system performance should be verified while the pump drive sources (rotor, engine, etc.) are individually varied throughout their approved operating range.

(7) Flight tests should be conducted throughout all altitudes, maneuvers, and control ranges while the system is instrumented as in d(2) and (4) above to determine that component ratings are not exceeded.

* 705. <u>§ 29.1439 PROTECTIVE BREATHING EQUIPMENT</u>.

a. <u>Explanation</u>. This paragraph prescribes minimum requirements for eye and respiratory protection from toxic atmospheres during in-flight emergencies if one or more cargo or baggage compartments are to be accessible in flight. The equipment provided shall assure the crew protection against an oxygen deficient, toxic or highly irritating environment such as smoke.

b. <u>Procedures</u>.

(1) The equipment should provide a good fit for the range of intended users.

(2) A donning procedure should be provided by the manufacturer, evaluated, and the final procedure included in the Rotorcraft Flight Manual.

(3) The equipment should accommodate crew members who wear corrective glasses. Nominal position of eyeglasses should not be compromised. The equipment should not cause distortion nor undue discomfort.

(4) The equipment donned under the stress of emergency should orient to the face and head, and interface to mating equipment, if required, in an obvious and uncomplicated manner. Respiratory and eye protection should be provided in a manner which does not compromise the crew's ability to perform required tasks.

(5) Any system which interfaces with existing components, should demonstrate satisfactory performance when operated with these components.

(6) For systems which require positive pressure to furnish satisfactory protection, a positive pressure vs. gas consumption curve should be supplied with the system along with instructions on the proper matching of the system or components to assure the minimum duration requirements of the standard are met.

(7) TSO-C99 and C116 are for Protective Breathing Equipment. If equipment is considered that is not qualified to one of these TSO's, it is recommended that their provisions be reviewed and used as a basis for a qualification program for the equipment being considered. TSO-C99 provides minimum performance requirements for emergency equipment which provides flight deck and cabin crew members with eye and respiratory protection from toxic atmospheres during in-flight emergencies. TSO-C116 results in protective breathing equipment that provides any crew member with the ability to locate and combat a fire within the aircraft cabin or any other accessible compartment.

(8) Additional information regarding oxygen supply systems can be found in Paragraph 786 of this document. *

706. <u>§ 29.1457 (through Amendment 29-25) COCKPIT VOICE RECORDER</u>.

a. <u>Explanation</u>. The function of the cockpit voice recorder (CVR) is to provide a record of the crew communications preceding an accidental crash of the rotorcraft. Over the last several years, the National Transportation Safety Board (NTSB) has determined that CVR's are invaluable in determining probable cause of an accident. Because of this fact and acts of Congress, the use of CVR's is required on many rotorcraft involved in passenger-carrying operations.

b. <u>Procedures</u>. The following areas are of particular consideration in the approval of a CVR installation.

(1) <u>Equipment Qualifications</u>. The CVR must be approved. The most common way of obtaining an approval is to qualify the CVR (and associated control panel, if appropriate) to TSO C84.

(2) <u>Cockpit Area Microphone (CAM)</u>. The third channel of recorded information is specified to be from a cockpit area microphone or from voice activated lip microphones at the first and second pilot stations. It should be noted that a continuously recording or "hot" microphone at both the first and second pilot stations would satisfy this CAM requirement. In view of the fact that most helicopter pilots utilize headsets and boom microphones for all cockpit communications and that most current design CVR's have provisions to use hot microphones, it is recommended that hot microphones be used at both the first and second pilot stations instead of a cockpit area mounted microphone.

(3) <u>CVR Mechanical Installation</u>. The CVR or the portion thereof which contains the recording should be physically located to enhance the probability of the recording surviving a crash. Normally, such a location would be in the lower portion of the rotorcraft as far aft as possible.

(4) <u>Intelligibility of Recordings</u>. Tests should be accomplished to determine that the recording is intelligible enough to make a positive identification of the speaker and the words or phrases spoken. This is usually accomplished by a flight test which provides an operation to produce the maximum cockpit background noise. The operation should provide for the normal speech of all crew members to be recorded on the pertinent channels. Then, during playback, preferably using a different listener, the listener should be able to identify the different crew members, the words and phrases spoken by the crew, and the radio communications made by and to the crew. The use of special filters and multiple playbacks to improve intelligibility is acceptable.

(5) <u>Electrical Power Supply</u>. The rule requires that the CVR should be supplied with power from a reliable source which does not jeopardize essential or emergency loads. For Category A rotorcraft, the CVR is not an essential load as specified in § 29.1309(e). However, since the functioning of the CVR is required by operating rules for some operations, it should be given priority over other nonessential loads.

(6) <u>Self-Test Function</u>. The CVR should be provided with a means in the cockpit which will allow a test to ensure the CVR is functioning properly. This may be accomplished by a manual playback feature.

(7) Bulk erasure. If this function is provided, the installation should be as follows:

(i) Any probable malfunction will not cause erasure of the recording medium.

(ii) The crash impact forces will not cause activation of the bulk erasure function.

(iii) Inadvertent actuation of the bulk erasure function is minimized. Usually, this is accomplished by requiring two separate actions to operate the bulk erasure.

* 707. § 29.1459 (Amendment 29-25) FLIGHT RECORDERS.

a. Explanation. The function of the flight recorder, sometimes referred to as a flight data recorder, is to provide a record of various aircraft and air data parameters during the operation of the helicopter. This data is utilized by accident investigators to aid in determination of the probable cause of an accident. The problems associated with acquisition of this data in aircraft not equipped with flight recorders has been complicated by the use of advanced instrument systems such as EFIS, EICAS, and IDS. The very nature of the operation of these systems precludes the deduction of post-accident data, as was possible with mechanical and electromechanical instruments, annunciators and switches. The National Transportation Safety Board (NTSB) therefore made a recommendation to the FAA that aircraft should be required to have flight recorders. Subsequently Congress mandated that flight recorders be required on many rotorcraft involved in passenger-carrying operations in accordance with FAR 91 and FAR 135.

b. Procedures. The following areas are of particular consideration in the approval of a flight data recorder installation.

(1) Equipment Qualification. The recommended procedure to obtain an approval for the flight recorder (and associated control panel, if appropriate) is to qualify the flight recorder to TSO C-124. The required underwater locating device should be qualified to the provisions of TSO C-121.

(2) Recorded Parameters and Accuracy.

(i) Airspeed. The installed flight recorder for a Category A helicopter should record the airspeed with an accuracy of 3 percent or 5 knots (whichever is greater) from a speed of 80 percent of V_{TOSS} to V_{NE} in level flight, and an accuracy of 10 knots from a speed 10 knots less than V_{TOSS} to a speed of 10 knots more than V_Y in climb.

(ii) Pressure Altitude. The flight recorder should be capable of recording the pressure altitude of the helicopter with a range of -1,000 feet to the maximum certified altitude. The error of this recording at sea level, excluding instrument calibration error, should not exceed $\pm$ 30 feet or a value of $\pm$ 30 feet for each 100 knots of airspeed (whichever is greater).

(iii) Direction. The flight recorder should be capable of recording the magnetic heading of the helicopter within at least 10 degrees for any heading. *

(iv) Vertical Acceleration. The flight recorder should be capable of recording the normal acceleration within the center of gravity range of the helicopter. The recommended range of this recording is an envelope of -3 to +6 G with an accuracy of at least ± 0.2 G.

(v) Time Correlation. The flight recorder should provide a time scaled correlation between the data recorded and the time at which this information was presented to the first pilot via his required flight instruments. This correlation should normally be established before flight, and should have an accuracy rate that does not diverge by more than 4 minutes and 4 seconds in eight hours.

(vi) Caveat. It should be noted that even though the requirements outlined above provide for compliance with the specific provisions of § 29.1459 regarding the acquired data and its accuracy, a flight recorder certified to these minimum standards will not meet the requirements of Appendix F of FAR 91 or Appendix C of FAR 135. If the flight recorder is to be used to comply with these operating rules, it is recommended that the appropriate appendix be consulted prior to requesting certification. The approved configuration may then be certified as meeting the requirements of the appropriate appendix.

(3) Flight Recorder Mechanical Installation. The non-ejectable flight recorder or the portion thereof which contains the recorded data should be physically located to enhance the probability of the recording surviving a crash. Normally, such a location would be in the lower portion of the rotorcraft as far aft as possible. However other locations in the helicopter may be suitable to meet the requirement to "minimize the probability of container rupture resulting from crash impact and subsequent damage to the record from fire." The normal accelerometer should be located within the most restrictive center of gravity of the helicopter. The required underwater locator is usually mounted to the case of the flight recorder.

(4) Electrical Power Supply. The rule requires that the flight recorder should be supplied with power from a reliable source which does not jeopardize essential or emergency loads. For Category A rotorcraft, the flight recorder is not an essential load as specified in § 29.1309(e). However, since the functioning of the flight recorder is required by operating rules for some operations, it should be given priority over other nonessential loads.

(5) Self-Test Function. The flight recorder should be provided with a preflight test which will provide confirmation that the recorder and its recording medium are functioning properly.

(6) Data Erasure Feature. If this function is provided and the flight recorder is not powered solely by an engine or transmission driven generator, the installation should provide the following features:

(i) Any probable malfunction will not cause erasure of the recording medium.

(ii) The crash impact forces will not cause activation of the data erasure function.

(iii) Inadvertent actuation of the data erasure function is minimized. Usually, this is accomplished by requiring two separate actions to operate the data erasure. *

* 708. § 29.1461 (through Amendment 29-3) EQUIPMENT CONTAINING HIGH ENERGY ROTORS.

a. Explanation. This section contains requirements for the installation of equipment containing high energy rotors. A high energy rotor is any rotor which has sufficient kinetic energy to cause damage to surrounding structure, wiring, and equipment if a failure occurs. Turboshaft engine and APU rotors are not covered by this paragraph. One of the following requirements of § 29.1461 must be met.

(1) Paragraph (b) deals with damage tolerance, containment, and control devices.

(2) Paragraph (c) deals with containment and inoperative speed controls.

(3) Paragraph (d) deals primarily with equipment location.

b. Procedures.

(1) Compliance with § 29.1461(b) can be shown by a combination of analysis and test. A failure modes and effects and a stress analysis, together with a dynamic test, could be used to verify that the rotor would withstand the damage from environmental effects, and that the rotor case would contain any parts that may separate from the rotor shaft. The analysis and test should include a demonstration of the control device's ability to prevent limitations from being exceeded.

(2) If compliance with the requirements of § 29.1461(c) is chosen, a test must be conducted which demonstrates that all parts from any type failure of a high energy rotor will be contained when that rotor is operating at the highest speed obtainable, with all speed control devices inoperative. This containment must not damage any components, systems, or surrounding structures that are essential for continued safe flight.

(3) If compliance with § 29.1461(d) is chosen, the location of the high energy rotor must be in an area where uncontained failed parts will not damage other components, systems, or surrounding structure which are essential for continued safe flight. It must also be shown that there is no possibility for failed, uncontained parts to enter the cabin area and endanger any occupant. *

709.-717. RESERVED.

SECTION 40. OPERATING LIMITATIONS

718. § 29.1501 (through Amendment 29-15) GENERAL. This section simply requires specified operating limitations in addition to any other information necessary for the safe operation of the rotorcraft to be determined. Secondly, it requires that this pertinent information be made readily available to the crew members as required in the various sections of this subpart. *

719. § 29.1503 (through Amendment 29-24) AIRSPEED LIMITATIONS: GENERAL.

a. Explanation. This section requires that a safe operating speed range be established for all rotorcraft. If the safe operating speed range varies with operating conditions (rotor speed, power, etc.), ambient conditions (altitude and/or temperature), rotorcraft configuration (gross weight, center of gravity, and/or external equipment), or type of operation (in ground effect (IGE), instrument flight rules (IFR), etc.), airspeed limitations that correspond with the most critical combinations of these factors must be established.

b. Procedures.

(1) Airspeed Limitations. The airspeed limitations for each critical combination of factors are established by tests or analyses and verified by flight test. The following are airspeed limitations that are typically required depending on the particular rotorcraft design:

(i) V_{NE} (Power On). See paragraph 720.

(ii) V_{NE} (One Engine Inoperative (OEI)). See paragraph 720.

(iii) V_{NE} (Power Off). See paragraph 720.

(iv) V_{LO} (Maximum Airspeed for Landing Gear Operation). Compliance with structural, handling qualities, and controllability requirements should be demonstrated at the airspeed limit.

(v) V_{LE} (Maximum Airspeed Landing Gear Extended). If this airspeed limit differs from the maximum gear operation speed, compliance with the applicable structural, handling qualities, and controllability requirements should be demonstrated.

(vi) Low Speed Flight Limitation. It is permissible for the applicant to establish a minimum airspeed operating limitation as a function of weight, altitude, and temperature as long as there is still a practical flight envelope.

(vii) V_{MINI} (Minimum IFR Speed). The minimum speed for which compliance with the IFR handling qualities requirements has been demonstrated should be established as a limit for IFR operations.

(viii) Maximum Sideward and Rearward Flight Speed. The maximum demonstrated sideward flight or crosswind hover and rearward flight or tailwind hover airspeeds should be provided in the RFM. If these maximum speeds resulted from a control margin limitation, they should be included in the airspeed limitations section of the RFM. If adequate control margin remained for the critical combination of rotorcraft configuration and ambient conditions, the maximum demonstrated sideward or rearward flight airspeeds should be included in either the performance section or the limitations section of the RFM as the applicant desires.

(ix) Maximum Airspeeds for Special Configurations or Special Equipment. Standard configuration airspeed limits frequently have to be reduced for specific changes or external modifications. The following are examples of special equipment or configurations that have required additional airspeed limitations:

(A) Doors open or doors off.

(B) External hoist/cargo hook (stowed).

(C) Fixed or emergency flotation gear.

(D) External avionics equipment (large antennas, wires, etc.)

(E) External fuel tanks.

(F) Skid pad or ski equipment modifications to standard skid type landing gear.

(x) Maximum Airspeeds after Failure of Required Equipment. Rotorcraft that require auxiliary equipment such as stability augmentation systems to comply with FAR requirements throughout the approved operating envelope frequently require airspeed limitations following failure of part or all of this system in order to comply after the failure. The following are examples of auxiliary equipment that have required maximum airspeed limitations after failure of all or part of the system:

(A) Stability Augmentation Systems (SAS).

(B) Automatic Flight Control Systems (AFCS).

(C) Fly-by-Wire Elevator Systems (FBW).

(D) Air Data Computer Systems (ADC).

(2) Groundspeed Limitations. Although not specifically required by this "airspeed limitations" regulation, it may be necessary to establish "groundspeed" limitations for wheel-gear-equipped rotorcraft. These limitations are required to show compliance with the ground-handling characteristic requirements, structural strength requirements, or the ground-loads requirements. However because of the operational similarity of groundspeed limits to airspeed limits, it is a common practice to include groundspeed limitations under the airspeed limitations heading in the flight manual. For this reason, groundspeed limitations are included in this paragraph of the AC. Groundspeed limitations should be established with adequate safety margins to account for the possible inaccuracies associated with the necessity for the pilot to estimate groundspeed from indicated airspeed and available wind speed and direction information during actual operations. The following are examples of groundspeed limitations that have been required during past type certification programs:

(ii) Tests for static longitudinal stability during approach should include the steepest approach gradient for which approval is requested. Static stability tests may be simulated by initially establishing a trimmed rate of descent for maximum approach gradient assuming zero wind conditions. Actual approach tests at the maximum approved gradient should be conducted to evaluate tracking and maneuverability, including the capability to correct downward to a glide path when approaching in a slight (10 knot) tailwind condition.

(4) Static Lateral - Directional Stability.

(i) Tests for directional stability usually require instrumentation for lateral cyclic position, pedal position, and sideslip angle. Testing for compliance with the specific directional requirement is relatively simple; however, the pilot should look for significant longitudinal trim changes and short period dynamic modes which might occur only during sideslip conditions. Side force characteristics are indicated by the variation of bank angle with sideslip during steady heading sideslips. The number of ball widths of deflection is also indicative of the side force cue available to the pilot. A correlation between sideslip angle and ball widths of skid can be obtained at given speeds for use during later testing after sideslip instrumentation is removed. A simple yaw string can be calibrated in a similar manner. The TIA should define the maximum slideslip angles which should not be exceeded during the flight test program. These angles must not be greater than the structural sideslip envelope substantiated and are not required to be that sideslip angle obtained with full directional pedal deflection. Sufficient side force cues should accompany sideslip to alert the crew when approaching sideslip limits. This is needed to assure that structural sideslip limits will not be inadvertently exceeded in service. Although not stated in the requirement, flight conditions for demonstration of static longitudinal stability are also appropriate for demonstration of static lateral-directional stability.

(ii) Dihedral requirements may be more difficult to assess. For those helicopters which do not meet the position and force gradient requirements for the conventional, cross-controlled sideslips, there are alternative tests which may be used to determine acceptable characteristics. If directional pedals are utilized in steady sideslips, the resultant rolling tendency is the sum of (1) the aircraft's roll due to sideslip tendency (dihedral) and (2) the aircraft's roll due to directional control input. If the rotorcraft has a tail rotor which is excessively high or low in relation to the rotorcraft's vertical center of gravity (c.g.), application of tail rotor thrust will introduce a significant rolling moment. The basic intent of dihedral stability testing is to determine the rotorcraft response to sideslip exclusive of directional control input. In general, if a tail rotor configuration is involved, and the tail rotor is above the vertical c.g. of the rotorcraft, the effect of pedal input upon dihedral effect is destabilizing during conventional, control-induced sideslips.

(iii) There are two alternate methods which, for small angles of sideslip, can give an indication of the basic dihedral stability of the rotorcraft. Both methods involve freezing directional controls while artificially creating sideslip by other means.

(iv) The first method is only applicable for rotorcraft with single main rotor systems. To utilize this method, the rotorcraft is stabilized in a given flight condition and small collective (torque) changes are applied in each direction (e.g., ± 5% & ± 10%) while holding pedals fixed. Sideslip angle, lateral control position, and lateral control force may be measured and plotted for small torque changes from trim. This technique will not work for aircraft which have collective to pedal or collective to lateral control couplings.

(v) In the second method, the rotorcraft is stabilized in a trimmed flight condition with a small amount of bank (5-10°). The rotorcraft is then rolled to an approximately equal angle of bank in the opposite direction holding the pedals fixed. The change in direction of bank results in a small change in sideslip angle and again sideslip angle may be plotted versus lateral control position and/or force. This test should be conducted in both directions and the results averaged. This method can give reasonably accurate results for small perturbations. Other factors contribute to the results of either of these two methods. It is always important to assess the roll due to sideslip tendency with pedal induced sideslips to assure lateral control forces are reasonable and in a proper direction for directional out-of-trim conditions, and to assure the pilot has adequate sideslip cues.

(vi) Wording of the dihedral requirement is intended to allow slightly negative dihedral stability at critical loading conditions. This will ordinarily result in positive dihedral stability throughout a great majority of the approved loading envelope. The test for maximum allowable negative dihedral effect would involve stabilization at a required flight condition, inducing a sideslip up to ± 10° from trim, then assessing lateral cyclic friction/deadband to determine if roll is restrained while remaining in the control system friction/deadband so that the control may be released without resulting in the aircraft rolling in the adverse direction. When testing for this condition, lateral cyclic friction should be adjusted to the minimum value.

(vii) The intent of the dihedral rule is to allow small amounts of control system friction and deadband to mask small values of negative dihedral. Where slope of the negative dihedral versus sideslip exceeds these small values, the negative dihedral shall not be approved. The operational pilot must not be presented with opposite cyclic sensing for similar sideslip conditions as loadings and flight conditions change. In general, large values of control system friction and deadband are undesirable. The addition of friction or deadband into the control system for the purpose of satisfying the dihedral requirement is not acceptable.

(viii) In approving small negative dihedral values, the pilot should insure that other positive flight cues, such as suitable side force, accompany sideslip. This will aid the pilot in determining direction of sideslip so that no reverse sensing or confusion accompanies sideslip conditions.

(5) Dynamic Stability.

(i) Dynamic characteristics are defined in quantitative terms; however, some areas of interpretation and technique need special consideration:

(A) Unlike fixed-wing aircraft where the size of the input has no effect on damping ratio, rotorcraft can be sensitive to the type and size of input used to excite each dynamic mode. For instance, it has been found that for the phugoid-type dynamic oscillation, damping ratio is inversely proportional to the size of the input. It therefore becomes important that dynamic excitations be sized to approximate the response of the rotorcraft in a moderate turbulent gust. Also, the dynamic input should be made with the control(s) which most accurately simulates the typical aircraft gust response. Obviously, for this evaluation some flying of the rotorcraft in turbulence is necessary to obtain knowledge of the rotorcraft's gust response. Pulses and doublets may be used to generate disturbances similar to a gust. To assist returning the control(s) to the trim position a hand held jig may be used. Use of attitude and rate instrumentation is desirable. The pilot may find that collective excitation, or collective in conjunction with cyclic, is most appropriate for gust simulation.

(B) The second area of concern in evaluating dynamic response is whether to let only one axis respond to an excitation or to let the rotorcraft respond in two or more axes. When it can be done safely, the rotorcraft should be allowed to follow its dynamic response in all axes. In other words, if pitch oscillations feed into roll, the pilot should attempt to observe and record the total aircraft dynamic response in both pitch and roll.

(C) The third area concerns strict compliance with the exact wording of the dynamic requirement. In this regard, a neutrally damped oscillation with a period of 19 seconds would not be acceptable; however, a very divergent oscillation that doubles in amplitude in 21 seconds would be acceptable. The 19-second oscillation is much less severe than the 21-second oscillation and yet is unacceptable by the "letter of the law." Figure 775-1 below is a graphical display of the dynamic requirement. The 19- and 21-second oscillations are shown as points (1) and (2). Point No. 1 is positioned much more toward the acceptable portion of the graph and yet by the "letter of the law" is unacceptable. The intent of the dynamic requirement is roughly approximated by the dashed-curved line. Areas to the right of that line may be considered for findings of equivalent safety.

(D) A fourth area requiring special care in testing is the aperiodic requirement. The most common aperiodic motion is the spiral characteristic which results when aircraft attitude is displaced in roll. The preferred method for testing this requirement is to stabilize precisely on a trimmed condition in straight flight, then displace the helicopter to 10° of bank, stabilize momentarily, set the controls as they were positioned for straight flight, and release them. Time and bank angles are then recorded. Recovery is initiated when bank angle or roll rate becomes excessive. Of particular interest is the time for bank angle to pass 20° and this time should not be so short as to cause the aircraft to have objectionable flight characteristics in the IFR environment. The time period to double amplitude (20°) should be at least 9 seconds. It is vitally important that controls (particularly lateral cyclic) is positioned exactly as it was for the straight flight condition. If a high resolution force trim system is not incorporated, an alternative method may be used. In this second method, the helicopter is trimmed for straight flight as described above and controls are released. Roll attitude may simply be allowed to vary naturally with time or small pulse input may be made with pedals. It is important that controls are positioned precisely as they were for the trimmed, straight flight condition and a plot of bank angle versus time is obtained. This plot is then compared against a divergent roll condition which doubles in amplitude every 9 seconds. Of particular interest is again the rate passing 20° of bank. If airspeed changes as the aircraft rolls or if roll/pitch coupling occurs, these changes should be allowed to interact naturally until recovery is necessary. Due to the sensitive nature of this test, smooth air is essential. Repeatability may be a problem. At least two test points in each direction should be obtained at each trim condition. Results may be averaged if they show reasonable repeatability. The same procedures may be utilized for an aperiodic pitch response; however, a displacement of 5° from trim should be used and of particular importance is the pitch rate passing 10°. Again, at least two test points in each direction should be obtained for each trim condition. Although not stated in the requirement, the flight conditions for demonstration of static longitudinal stability are also appropriate for demonstration of dynamic stability. The degree of testing referred to here represents that which might be required of a marginally stable helicopter. For those configurations which provide good aerodynamic stability or use varying degrees of SAS, the scope of the demonstration program would be decreased significantly.

(ii) Control system dynamics should also be evaluated. This may be accomplished by lightly bumping each control in flight and observing its free response. Any resulting control motion must dampen quickly and should not be driven by aircraft/control system interaction. This will assure safe flight in the event a control is inadvertently bumped or released from an out-of-trim condition.

(6) Stability Augmentation System (SAS).

(i) If a SAS installation stabilizes the helicopter by allowing the pilot to "fly through" and perceive a stable, well-behaved vehicle, it qualifies as a SAS, and if reliable, receives credit under Sections III through VII of Appendix B for use in complying with all-handling qualities requirements. If a conventional autopilot does not provide "fly through" capability or allow the pilot to perceive a stable, well behaved vehicle through his manipulation of primary flight controls and feedback from those controls, then it tends to remove him from active involvement in flying and is eligible primarily as a workload reliever.

(ii) If handling qualities credit is given for a SAS then it must be shown to be reliable. If a reliable SAS is incorporated, it should be operational during handling qualities testing for trim and stability. Reasonable single failures of the SAS must be evaluated and the resultant handling qualities must be evaluated to assure that in this degraded configuration, (1) handling qualities have not been degraded below "VFR" levels defined in FAR Part 29, Subpart B, (2) the helicopter is free from any tendency to diverge rapidly from stabilized flight conditions, and (3) the helicopter can be flown IFR throughout its endurance capability without undue difficulty by the minimum flight crew. Compliance with a majority of the IFR handling qualities requirements is desired and the degraded characteristics should be documented and explained. Revised flight envelope boundaries for the failed condition may be considered if they are controllable by the pilot, e.g., altitude and airspeed. When loss of a SAS results in a need for minor adjustment of a flight condition then a system can be accepted that allows failures during the life of each helicopter. If loss of the system will prevent continuation of safe flight and landing, the reliability of the system must be high enough to assure that failure of the system will not be expected to occur during the life of the helicopter fleet. When evaluating the reliability of a system, the installation of the system should be considered as part of the design. The total system including inputs, outputs, environment, isolation features, and exposure times is a pertinent consideration.

(iii) Stability augmentation system reliability is evaluated by Systems and Equipment personnel. If credit is to be given for system reliability, freedom from malfunction, hardover and oscillatory conditions (limited to critical frequencies determined during autopilot failiure analysis), a thorough system evaluation is needed. Flight test personnel should coordinate closely with the systems and equipment personnel whenever credit is given for advanced design and system reliability because the hardover/malfunction condition may not require in-flight testing. The decision is made on the basis of system design, failure analysis, and overall probability of malfunction. If flight testing is required, appropriate delay times as shown below, are required. If the system is to be approved without flight restrictions (operating at all times), malfunctions should be demonstrated to be satisfactory during takeoff, climb, cruising, landing, maneuvering, and hovering. If a flight restriction is provided, it should be determined to be an appropriate and relevant operating limitation, and it should be specified in the rotorcraft flight manual. Significant information regarding the restriction should be made available to the pilot in the operating procedures section of the rotorcraft flight manual. If the restriction excludes operation under any of the flight conditions listed above, flight testing of the condition is not required.

Flight Condition	Time Delay
Hover, takeoff, and landing	Normal pilot recognition and reaction time
Maneuvering and approach	Normal pilot recognition plus 1 second Note: Recovery from simulated malfunctions of any SAS axis occurring while the pilot is applying control inputs to cause rotation about that axis may be initiated with normal pilot reaction; the 1-second delay in maneuvering flight pertains to established turns (level, climbing, and descending) only.
Climb, cruise, and descent	Normal pilot recognition plus 3 seconds

For helicopters requiring a minimum crew of two pilots and with stability systems that do not have coupling capability such as vertical speed hold, altitude hold, or navigation tracking, a time delay of 1 second may be used in climb, cruise, and descent. Reference to visual cues is assumed only in hover, takeoff, and landing. For other flight conditions, the pilot is assumed to recognize the malfunction condition without reference to outside visual cues. If the stability system has not previously been certified as a part of the aircraft for VFR flight, malfunctions should also be conducted throughout the VFR envelope utilizing the appropriate delay times in Advisory Circular 29-1. Pickup to a hover, landing, sideward, rearward, * and forward hovering flight must be considered, because of the visual cues available to the pilot operating VFR, shorter delay times following stability system malfunctions may be appropriate. These delay times are:

(A) One to three seconds delay for cruising flight. (The time delay selected should be based upon the degree of stability provided and the amount of alertness required of the pilot. For example, three seconds are required for a fixed-wing transport aircraft in cruising flight).

NOTE: If the improved stability and the resultant higher degree of relaxation by the pilot has justified time delays greater than one-second minimum in cruise, then a reexamination is in order of the engine failure time delays used during the original type certification prior to the SAS installation.

(B) One second delay for climbing flight.

(C) Zero second delay for takeoff, landing, hovering, and maneuvering flight. *

(iv) A good method to accurately determine pilot recognition and reaction time is to establish typical climb, cruise, descent, and approach conditions and instruct a subject pilot to react as soon as he recognizes individual hardover conditions in pitch, roll, yaw, and heave (if installed). Several pilot subjects may be used. Sensitive recording instrumentation is needed to show the hardover input to the actuator and the pilot's initial control movement. This procedure is usually conducted prior to the critical hardover tests so that the total necessary time delay (recognition plus 3 seconds, etc.) can be established. This procedure actually determines recognition plus reaction time, although reaction time has been shown in hardover testing to be a relatively constant 0.5 seconds. Different recognition times for various axes are not unusual. During one recent program, recognition time for directional hardovers was 0.3 second, but for roll hardovers was 0.9 second. There is typically 0.1 second or less scatter among properly briefed pilots. Recognition time is then added to delay time to determine total necessary delay for hardover testing. As an example, for the above roll condition, a single pilot configuration would require a total 3.9 second duration from signal input to initial control actuation for recovery. Allowable attitude excursions must also be considered. Although allowable attitude excursions during hardover testing probably depend more upon acceleration and rate of acceleration than on attitude, a general rule of 30° pitch and 60° bank may be used. For some designs, maximum safe attitudes may be lower. Certain responses with rapid initial motion, but self-correcting characteristics thereafter have been allowed to diverge as much as 55° in pitch and 80° in roll as long as no rotor system or control difficulties result during malfunction or recovery. The key is: Can a safe, reasonable recovery be made without exceeding aircraft limits? During high speed malfunction testing, the maximum speed allowable during malfunction or during recovery is 1.11 V_{NE} (V_{df}). The maximum allowable speed for SAS operation must be adjusted to prevent exceeding V_{df} during malfunction testing at any altitude.

(v) Applicable procedures and techniques for conduct of hardover tests are contained in Paragraph 637 of this AC. All cockpit emergency controls including emergency quick disconnects should be "red." The quick disconnect may be actuated at initiation of recovery. Other disconnects should only be actuated after full aircraft control has been achieved following recovery. Aircraft limits may not be exceeded during malfunction or recovery. If a monitor device automatically disconnects the SAS, it must be clearly annunciated to the crew. *

(vi) Series actuator hardover conditions in some rotorcraft can seriously degrade control margin. Critical loadings, power settings, rpm, and altitudes in conjunction with a SAS actuator hardover in an adverse direction can result in reduction of control travel requiring flight envelope constraints. Flight testing is usually necessary to determine the appropriate flight envelope reductions.

(vii) Subsequent failures and unrelated probable combinations of failures must be considered, including subsequent SAS failures. Systems and equipment section analysis should provide necessary SAS malfunction combinations for flight testing as a result of their system analysis. Minimum requirements for dispatch and procedures following failure should be included in the malfunction analysis. Results of the probability analysis and the resultant malfunction configurations are primarily the responsibility of the systems and equipment section.

(viii) No reasonably probable failure should result in a worse condition than that tested for hardovers. For example, if a magnetic brake force trim system is employed, failure of electrical power to the magnetic brake circuit may cause the cyclic control to fall which may result in a more dangerous flight condition than individual SAS hardovers. The overall control system is to be evaluated for all probable failures to preclude hazardous failure conditions. Other areas for investigation include beep trim and auto trim failures. The delay

The location of the generator field protective devices has been a problem in some helicopters. The protective devices that can result in the loss of a required power system source should be capable of being reset or replaced in the cockpit while in flight. This position is further supported by the occurrence of nuisance opening of circuit protective devices in rotorcraft. Further discussion on this issue is included in paragraph 655b(4) of this advisory circular.

(F) Intercommunication System. All audio for the entire helicopter comes together at this system. An evaluation should be made to assure that no single failure will result in the loss of all audio for the helicopter. Check for common grounds, common connectors, etc. Power inputs should also be disabled.

(14) Rotorcraft Flight Manual Material.

(i) In addition to other required information, the limitations section of the Rotorcraft Flight Manual (RFM) or RFM Supplement must include the approved IFR flight envelope, minimum IFR crew requirements, the minimum required equipment for dispatch into IFR conditions that is not covered by the operating regulations, and the maximum approach gradient which has been approved. If a significant loss of altitude is experienced in any flight regime or maneuver during certification analysis or testing, the emergency operating procedures should include a statement of this altitude loss along with any other appropriate information.

(ii) The limitations section of the RFM should not include restrictions prohibiting external cargo operations. These operations are covered by FAR Parts 91 and 133 and all external load operations conducted under these parts must be approved by the controlling operations inspector. It is the responsibility of the operator to demonstrate and the operations inspector to confirm that any external load operation, including en route IFR, can be safely conducted.

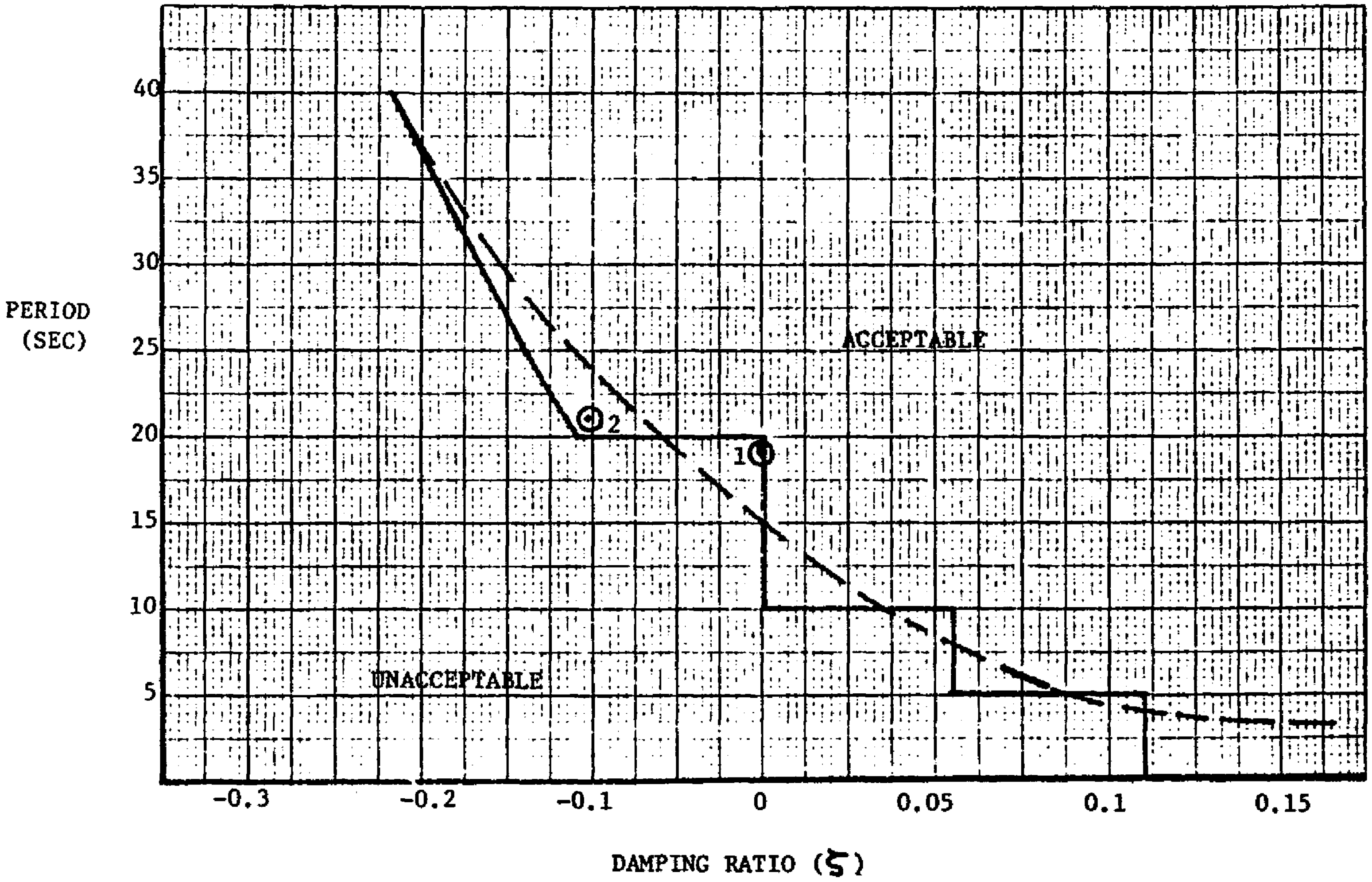

FIGURE 775-1. HELICOPTER DYNAMIC STABILITY REQUIREMENTS FOR IFR

(C) Malfunction Indicator Circuit Loading. It should be determined that the malfunction indicator source impedance is matched by its loads and that the source capacity is not exceeded. When the system is capable of transfer, the transfer loads should also be considered (§ 29.1301).

(D) Synchro Signal Loading. When parallel loads are added to Synchro's, the manufacturers' specifications should be reviewed to assure that the additional loads do not result in an overloaded synchro.

(iv) Interface. In many cases, the mating units of a system are designed by different manufacturers. For example, a brand-X gyro may be designed for operation with a brand-X flight director, but later a modifier decides to operate a brand-Y autopilot with the brand-X gyro. This applies just as well to NAV receivers, AREA NAV units, course indicators, omni bearing selectors, tachometer indicators, transmitters and many other equipment items. When this is the case, the applicant should provide data, in summarized form, describing those characteristics such as impedance, volts, etc., that are necessary to assure a compatible and reliable system. The data should also reference the source of the interface data (§ 29.1301).

(v) Flight Tests. An FAA engineering flight test is required during type certification or after modification that changes the established limitations, flight characteristics or performance of a helicopter or any of its required systems or operating procedures. New installations of equipment in the cockpit or modifications that affect existing equipment in the cockpit should be evaluated by appropriate flight test personnel, if it is necessary to evaluate operational aspects of the change. Where possible, cockpit arrangement, placards, markings, instrument visibility, and light reflections can be evaluated on the ground if the applicant opts to darken the windows. Electromagnetic compatibility functional checks, windshield glare, and pilot workload evaluations may be conducted in flight at the FAA flight test pilot's option.

(vi) Radio Master Switches. Some installations incorporate radio master switches to control special busses for the avionics systems. If this capability is provided it should be evaluated to assure failure modes are not introduced that will result in excessive or even total loss of all required avionics. One switch that controls all required avionics is not considered acceptable for IFR installations. The evaluation should include an assessment of the loss of the systems to be included on the radio master switch(es), and the subsequent effect on continued safe flight.

b. Test Procedures. Where the airworthiness or operating regulations require a system to perform its intended function, and/or not create a hazard to other required systems, sufficient testing should be accomplished to assure satisfactory performance. When ground testing is not sufficient to properly evaluate a system's performance, flight testing should be accomplished. Acceptable flight test criteria for specific navigation and communication equipment are contained herein. If the rotorcraft is to be approved for IFR operations, the additional criteria of paragraph 775 of this advisory circular should be satisfied.

(1) VHF Systems.

(i) General. Intelligible communications should be provided between the rotorcraft and ground facilities throughout the airspace within 100 NM of an FAA ground facility from radio line of sight altitude to the maximum altitude for which the rotorcraft is certificated. Communication should be provided with the rotorcraft at or above line of sight altitude in right and left bank up to 10 degrees and on all headings. Radio line of sight can be computed from the formula $d_L = .87 (\sqrt{2H_1} + \sqrt{2H_2})$ where d_L is the distance in nautical miles, H_1 is the ground antenna height in feet, and H_2 is the airborne antenna height in feet.

(ii) Electromagnetic Compatibility (EMC). With all systems operating in flight, verify by observation that no adverse effects are present in the flight systems.

(iii) Antenna Measurement. If satisfactory antenna measurement data are provided, the following flight test may be reduced to checks in right and left turns in the vicinity of the predicted bearings of worst performance. If antenna locations are symmetrical, tests may be conducted using only one direction of turn.

(A) Long Range Reception. Starting at a distance of at least 100 NM from the ground facility antenna, perform a right and/or left 360 degree turn at a bank angle of at least 10 degrees. Communicate with the ground facility every 10 degrees of turn to test the intelligibility of the signals received at the ground station and in the rotorcraft. For 100 NM, the minimum line of sight altitude is approximately 7,000 feet.

(B) Approach Configuration. With the landing gear down and with the rotorcraft in the approach configuration (at a distance of 10 NM from the ground station and in an idle power descent toward the station), demonstrate intelligible communications between the rotorcraft and the ground facility.

(2) HF Systems.

(i) Acceptable communications should be demonstrated by contacting a ground facility at a distance of at least 100 nautical miles. Single sideband equipment should also perform acceptably in the amplitude modulation mode of operation.

(ii) It should be demonstrated that precipitation static is not excessive when the aircraft is flying at cruise speed (in areas of high electrical activity, including clouds and rain if possible). Use the minimum amount of installed dischargers for which approval is sought.

(3) VOR Systems.

(i) These flight tests may be reduced if adequate antenna radiation pattern studies have been made and these studies show the patterns to be without significant holes (with the rotorcraft configurations used in flight, i.e., landing gear retracted en route and extended for approach). Particular note should be made in recognition that certain rotor rpm settings may cause modulation of the course deviation indication (rotor modulation). VOR performance should be checked for rotor modulation in both approach and en route operation while varying rotor rpm throughout its normal range.

(ii) The airborne VOR system should operate normally with warning flags out of view at all headings of the rotorcraft (in level flight) throughout the airspace within 100 NM of the VOR facility while flying above the radio line of sight altitude to within 90 to 100 percent of the maximum altitude for which the rotorcraft is certified.

(20) Night Vision Goggles (NVG).

(i) Background. Night vision goggles (NVG) have been used by U.S. military pilots since the early 1970's. The first units (first generation or GEN I) were constructed from the rifle "Sniper-Scopes." These units did not provide much light amplification. The second generation (GEN II) were still primarily designed for ground use. Second generation high performance units (military designation AN/PVS-5C) had some consideration for flight use but were still lacking in several aspects. A light level of at least a quarter moon well above the horizon was required for operation of these NVG. At first the normally helmet-mounted units covered the pilot's entire upper face and the pilot could only see through the NVG. In order to protect the light amplification system these NVG had an automatic shutoff feature when brighter than relatively low levels of light were encountered. Normal incandescent and especially red incandescent lights would cause these NVG to shut down. Aircraft cockpit lights, especially the red warning lights, would cause "blooming" (an increased brightness of all or portions of the NVG field of view with the disappearance of the "picture" in that area) or a total shutdown of the NVG. Military aircraft cockpits and lighting systems were significantly modified to avoid this problem. In the late 1980's the military pushed technology for better and aircraft compatible NVG. Third generation (GEN III, military designation ANVIS or AN/AVS-6) NVG systems became available about 1988. These systems require only star light for satisfactory operation.

(ii) Procedure. As of January 1990, no approvals for civil helicopter operations with NVG have been issued. Since NVG are not installed in the helicopter, they are not required to be approved as part of the type design. However, since an operational approval would be required for use of NVG, they should meet some acceptable performance standard. The minimum standard recommended is the GEN III NVG. The performance of these NVG are rated as their spectral response to irradiated light sources, measured as density of incident photons per square meter. Figure 776-1 shows a comparison of the spectral performance of GEN II and GEN III NVG. Third generation, AN/AVS-6, NVG have been evaluated for compatibility with a limited number of helicopters and were generally found to be usable during en route operations with no cockpit lighting systems modifications. It is anticipated, however, that some aircraft may require significant modifications to the existing cockpit lighting system. The FAA policy is that modification of the cockpit to a non-compliant configuration to accommodate NVG use is not acceptable. For instance, alteration of the required red warning annunciators to some other color is not acceptable. Since individual helicopters may have been modified with additional lights or systems, each helicopter being considered for use with NVG should be evaluated by an FAA representative during a night flight. If it is anticipated that cockpit lighting system modifications will be required to achieve an adequate level *

* of NVG compatibility FAA involvement should be arranged as soon as possible. Preferably this evaluation flight would be made with two pilots or a pilot and safety observer, over a known area, where all the aircraft and cockpit lights are operated and their effect on the NVG determined. Reflections of landing or searchlights on windshields or other glass during approach or landing may affect NVG and may impose a minimum altitude restriction for use of NVG. Failure of the NVG should be evaluated during any critical flight phase.

Note that the above discussion is purposely limited in scope. Issues such as crew training and operating limitations would have to be addressed in detail to obtain an operational approval.

(21) Rotorcraft Health and Usage Monitoring Systems (HUMS).

(i) General. HUMS can be divided into two major categories: Health Monitoring Systems and Usage Monitoring Systems. The provisions of § 29.1301 are used to determine that the system performs its intended function. The provisions of § 29.1309(a) and (b) are used to look at the impact of environmental conditions and malfunctions. To date (mid-1990) HUMS have not been approved to replace service life or other specific physical limits but several systems are now in the process of seeking approval. Health monitoring systems are considered to be the serious applications of this technology, and it will probably be some time before the necessary data base to allow full reliance on this technology is available. There have been numerous approvals of usage monitoring systems as optional equipment, and a good example of this technology is a condition monitoring system described in 776b(19) above.

(ii) Health Monitoring Systems.

(A) It is anticipated these systems will begin as "optional" systems in order to build a data base to support expansion of the approval to achieve credit for extension of maintenance intervals, and so forth. Some of these applications may require system redundancy, and some may require DO178A Level I or equivalent software.

(B) Some systems that are being considered will utilize off aircraft processing of data. If this is to be pursued it should be assumed that the aircraft data will be lost or misplaced at the processing center, and the aircraft system design should consider this possibility. Some on-board data storage is one way to account for this lost data. The integrity of the processing center's software should be equal to that of the aircraft software. In addition the intervals for processing the data from each flight should be specified as part of the approval.

(C) Due to the limited experience with these systems it is suggested the issue paper process be utilized to record the progress of the approval, and to provide information for later updating of this AC material. *

(iii) Section 29.939, turbine engine operating characteristics, intends a flight investigation to ensure that no adverse characteristics are present to a hazardous degree during normal and emergency operation in the allowed flight envelope. The evaluation should include assessment of the minimum FADEC system certification configuration; i.e., the minimum proposed by the applicant to meet Part 29 requirements. Reduced capabilities (e.g., restrictions on normal collective movements, limited aircraft maneuvers, etc.) may be acceptable for degraded FADEC modes or backup systems not required to meet Part 29 requirements if those degraded capabilities are reasonable and not hazardous as determined by flight evaluation. The restrictions should be specified in the flight manual.

(iv) The rotorcraft with FADEC engines must of course meet all of the Part 29 requirements, but the areas described herein are those which deserve special attention.

785. AGRICULTURAL DISPENSING EQUIPMENT INSTALLATION.

NOTE: This paragraph has been extensively revised and expanded to clarify the restricted category certification of agricultural dispensing equipment installations on rotorcraft.

a. Explanation. In the early development of the helicopter one of its primary usages was agricultural operation. The FAA recognized that the existing requirements, which were designed primarily to establish an appropriate level of safety for passenger-carrying aircraft, imposed an unnecessary economic burden and were unduly restrictive for the manufacture and operation of aircraft intended only for use in rural, sparsely settled areas. Therefore, a special document that established new standards for agricultural dispensing equipment and other special purposes was developed. Restricted Category CAM 8 became effective October 11, 1950.

(1) During the recodification of 1965, CAR 8 ceased to exist as a regulatory basis and selected portions addressing certification were incorporated into FAR 21. While the specific standards in CAR 8 were not changed substantially when adopted into FAR 21, the less restrictive philosophy of CAM 8 and the policy material that was stated in the preamble to CAM 8 were not clearly conveyed.

(2) Advisory material published in 1965 and revised in 1975, summarized the information contained in the advisory portions of CAM 8. This new advisory material indicated that the CAM advisory material would be applicable to the related FAR's. Unfortunately, this document specified that CAM 8 could be used in conjunction with certain FAR's for restricted category certification of small agricultural airplanes only. Rotorcraft were omitted. *

(3) A survey of restricted category rotorcraft projects related to agricultural modifications indicates that the CAM 8 philosophy was interpreted to allow the use of AC 43.13-2A structural criteria for most STC's issued through the early 1980's. Since then more restrictive guidance based on CAR 6 and FAR 27 requirements has been applied by some ACO's to several STC applications. Since the more restrictive guidance imposed a significant economic burden on the industry, the HAI requested a meeting with the FAA during the 1990 annual convention in Dallas. As a result of the meeting, an Action Notice to clarify the interpretation of FAR 21.25(a)(1) for restricted category aircraft has been issued.

(4) The following advisory material is a result of a reassessment of past and present policy.

b. Procedures. The certification basis for agricultural dispensing equipment in the restricted category is FAR 21.25(a)(1) as interpreted by Action Notice 8110.22. The accountable Directorate guidance for the substantiation requirements for helicopters is as follows:

(1) Substantiation of the agricultural dispensing system hoppers or spray tanks to the load factors provided in Figure 785-1 provides for proof of structure. The load factors of Figure 785-1 address the critical structural load conditions of dispensing equipment mounted in or near the fuselage and provide adequate margins of safety.

FIGURE 785-1
ACCEPTABLE ULTIMATE LOAD FACTOR FOR
AGRICULTURAL DISPENSING EQUIPMENT DESIGN

	UP	DOWN	SIDE	FORWARD	AFT
Tanks & Equipment Mounted In Or Near The Fuselage	1.5g	4.0g	2.0g	4.0g Note 1	----
Spray Booms	1.5g	2.5g	----	Note 1	2.5g Note 2

Note 1: An ultimate load factor of 2 G's is acceptable for externally side or under fuselage mounted tank and forward mounted spray booms where failure in a minor crash landing will not create a hazard to occupants or prevent exit from the helicopter.

Note 2: The aft loads for spray booms may be developed by the applicant based on the 111 percent of V_{ne} for which certification is requested or the load factors of Figure 785-1, whichever is greater.

(2) The applicant may elect to substantiate his/her product by either static or dynamic testing, by analysis, or any combination thereof. *

(3) Lower load factors may be used only when justified by manufacturer's data, rational analysis, or actual rotorcraft flight and ground load demonstrations.

(4) Tank pressure test, while not mandated, is recommended for safety reasons. An acceptable procedure is included in paragraph c(4).

(5) Dispensing equipment installation attach points. If attach points exist which are an integral part of the helicopter and these attach points have been certified to the standard category requirements no further substantiation of the attach point is required if an analysis indicates the dispensing system does not impose loads which exceed those for standard category certification.

(6) Ground clearance for dispensing equipment installation. A 5-inch ground clearance has typically been used for skid gear equipped helicopters which incorporate belly mounted supply tanks/hoppers or systems which have dual side mounted supply tanks/hoppers and the design incorporates cross tubes or other system components which are located beneath the bottom of the fuselage when these components are rigidly attached to the airframe structure. The 5-inch dimension is measured vertically from the ground to the lowest point of the installed system, with the helicopter in its operational configuration and gross weight (including disposable load) and while resting on a smooth, level asphalt surface. For helicopters equipped with wheels and/or landing gear struts, the maximum system deflections should be considered when determining the 5 inches of acceptable static ground clearance. The 5-inch ground clearance would only apply to original configuration of newly manufactured helicopters. However, a 3-inch ground clearance has been found acceptable and may be approved for skid gear equipped helicopters to account for the in-service permanent set allowed for skid gear members, (i.e., cross tube deflections allowed per the maintenance manual). Cable supported systems, (i.e., cargo hook installations) or dispensing systems utilizing flexible ducts (certain types of dry material dispensing equipment which may or may not be retractable) have been approved even though portions of the system may contact the surface during a normal landing.

(7) A number of rotorcraft are approved for external cargo operations that allow a gross weight higher than the approved internal gross weight limit. This difference is usually due to the allowable weight limit restriction of the landing gear. (The gear is not approved for the higher weight.) Those types of dispensing equipment, that can be loaded in flight to a weight that exceeds the allowable limit of the landing gear should incorporate a reliable means that rapidly reduces the total aircraft gross weight to within allowable landing gear limits. In most cases, this will involve jettison of the disposable load. The time interval for this operation should be demonstrated, and should not exceed a recommended 3 seconds from a level flight condition. *

(8) A flight check or demonstration of the agricultural dispensing equipment installation is normally conducted. This flight check should also qualitatively determine that no hazardous deflection or resonance in the helicopter or dispensing system exists. This flight check should be conducted in accordance with the requirements of FAR 133.41.

(9) For rotorcraft certificated in dual categories, the inspection requirements of FAR 21.187(b) must be observed when converting from restricted to normal category.

c. Acceptable Means of Compliance.

(1) Analysis Method. Structural analysis (static) may be used if the structure is of a configuration for which experience has shown the method to be reliable. Structural substantiation of tanks that are designed to contain liquid materials may be accomplished by pressure testing. For tanks or hoppers designed to contain dry material, (e.g., dust or fertilizer) static load tests may be used to verify structural integrity. The tank/hopper, mounting hardware, and support structure should all be substantiated to the load conditions specified by this paragraph considering the effects of internal fluid pressures when applicable.

(2) Static Tests. Static tests of tank/hoppers, mounting hardware, and support structure for each critical load condition may be accomplished using conventional techniques; such as, dead weight loading, whiffletree systems, and hydraulic rams. If tests of the tank and its mounting hardware are conducted using a test fixture representing the helicopter, the helicopter support structure may be substantiated independently by means of test and/or analysis. Static test loads should be applied in combination with associated internal fluid pressure loadings. The ultimate loads specified in Paragraph 785 should be sustained for at least 3 seconds without failure.

(3) Dynamic Tests.

(i) If the applicant elects to test to the load factors noted herein, the maneuvering and gust loadings will be considered to be adequately substantiated. For each condition, the critical volume and density of fluid should be used.

(ii) The tank and mounting hardware should support ultimate loads without detrimental permanent set or failure, respectively. The helicopter support structure may be included in the dynamic tests, or it may be substantiated separately via static test and/or analysis for each condition specified by this paragraph.

(4) Pressure Testing. Internal pressure loads may be applied using the water standpipe technique. Standpipe water height should be accurately computed for each critical spray tank static test loading. Pressure testing of spray tanks is not absolutely essential but is recommended for safety reasons. This testing will also determine whether the joints and connections are tight and will not leak in addition to determining any weak spots in the construction. Where spraying is done with highly volatile and flammable liquids, or where the tank has a return line, such as in an engine oil tank where the fluid is pumped back into the tank, it is recommended that the tank be tested for a pressure of 5 pounds per square inch. For other liquids, and where no fluid return line is used, testing to 3 1/2 pounds per square inch should be satisfactory. There are many ways of pressure testing a tank, however, it is believed that the simplest and easiest method is to fill the tank with water and use a standpipe filled with water. A 1 1/8-inch pipe can be connected to the venting tube or one adapted to the filler opening. In either case the height of the pipe would be the same. For a 3 1/2 psi test of the tank the height of the water in the pipe would only need to be 8 feet and for a 5 psi test only an 11 1/2-foot height of water will be needed.

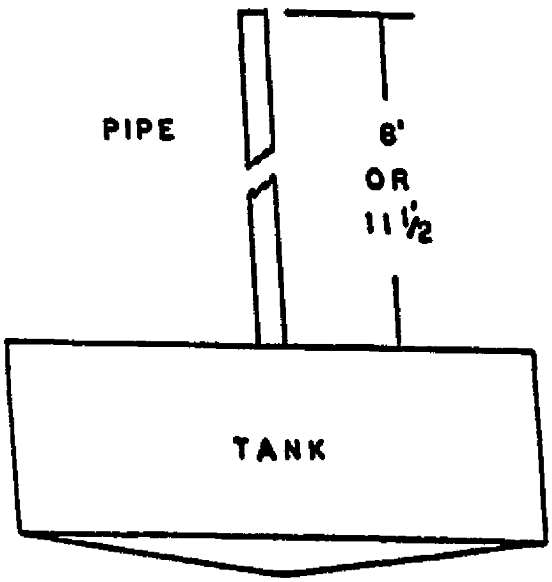

Figure 785-2. Sketch of Tank Pressure Test *

786. EMERGENCY MEDICAL SERVICE (EMS) SYSTEMS, INSTALLATIONS, INTERIOR ARRANGEMENTS, AND EQUIPMENT.

a. Explanation. This paragraph pertains to EMS configurations and associated rotorcraft airworthiness standards. EMS configurations are usually unique interior arrangements that are subject to the appropriate airworthiness standards, FAR Part 29 or its predecessor CAR Part 7, to which the aircraft was certificated. No relief from the standards is intended except by § 21.21(b)(1) and exemption. EMS configurations are seldom, if ever, done by the original manufacturer.

(1) The FAA has not specified in the airworthiness or operating rules the minimum equipment for an EMS configuration. Whatever equipment is presented for evaluation and approval is subject to compliance with the airworthiness standards. Any equipment that is not essential to safe operation of the aircraft is evaluated for a "no hazard approval;" i.e., it is optional equipment and may be approved provided the use, operation, and possible failure modes of the equipment are not hazardous to the rotorcraft. Safe flight, safe landing, and prompt evacuation of the rotorcraft, in the event of a minor crash landing for any reason, are the objectives of the FAA evaluation of interiors and equipment unique to EMS.

(i) For example, a rotorcraft equipped only for transportation of a nonambulatory person (a police helicopter with one litter) as well as a rotorcraft equipped with multiple litters and complete life support systems and two or more trained attendants/medical personnel may be submitted for approval. These configurations will be evaluated to the airworthiness standards appropriate to the aircraft certification basis.

(ii) Transport rotorcraft should comply with many flightcrew and passenger safety standards which dictate features of the basic certified rotorcraft which are related to the interior arrangement, to the doors and emergency exits and, to occupant protection. Compliance with the airworthiness standards results in an emergency interior lighting system, placards or markings for doors and exits, exit size, exit quantity and location, exit access, safety belts, and possibly shoulder harnesses or other restraint or passenger protection means as a part of a rotorcraft type design. The features, placards, markings, and "emergency" systems which are required as a part of the type design should be retained unless specific replacements or alternate designs are necessary for the EMS configuration to comply with the airworthiness standards.

(2) Many EMS configurations of transport rotorcraft are equipped with the following:

(i) Attendant/medical personnel seats which may swivel.

(ii) Multiple litters, some of which may tilt.

(iii) Medical equipment stowage compartments.

(iv) Life support and other complex medical equipment.

(v) Incubators for infants.

(iv) In prominent place when filler box is open, preferably near liquid oxygen drain valve:

DO NOT OPEN DRAIN VALVE UNTIL DRAIN HOSE AND DRAIN TANK ARE CONNECTED

(v) Other placards, such as one at the converter cautioning about the presence of liquid oxygen, may also be appropriate.

(7) Other installation criteria are given in Chapter 6, AC 43.13-2A, Acceptable Methods, Techniques, and Practices--Aircraft Alterations, dated June 9, 1977, and should be given full consideration.

(D) Precautions. The referenced SAE report contains precautions peculiar to a liquid oxygen installation, and this material should be reviewed. It should also be emphasized that liquid oxygen equipment and the aircraft being serviced must be electrically grounded during servicing to prevent an accumulation of static electricity and discharge. The following considerations are included for special emphasis:

(1) System Cleanliness. The completed installation should be free of oil, grease, fuels, water, dust, dirt, objectionable odors, or any other foreign matter, both internally and externally prior to introducing oxygen in the system.

(2) Closures. Lines which are required to be disconnected, due to the location of the converter within the rotorcraft during rotorcraft maintenance checks or overhaul, should be capped to prevent materials which are incompatible with oxygen from entering the system when the system integrity is broken. Caps which introduce moisture and tapes that leave adhesive deposits shall not be used for these purposes. All openings of lines and fittings shall be kept securely capped until closed within the installation.

(3) Degreasing. All components of the oxygen system should be procured for oxygen service use in an "oxygen clean" condition. Parts of the oxygen system, such as tubing, not specifically covered by cleaning procedures should be degreased using a vapor phase trichloroethane degreaser. Ultrasonics may be used in conjunction with vapor phase degreasing for the cleaning of components.

(4) Purging. The system should be purged with hot, dry 99.5 percent pure oxygen gas in accordance with the manufacturers recommendations after:

(i) Initial assembly of the oxygen system; and

(ii) After system closure whenever the oxygen system pressures have been depleted to zero, or the system has been left open to atmospheric conditions for a period of time or is opened for repairs.

(5) Maintenance and Replacement. All parts of the oxygen system should be installed to permit ready removal and replacement without the use of special tools. All tubing connections and fittings should be readily accessible for leak testing with a leak test compound formulated for leak testing oxygen systems and for tightening of fittings without removal of surrounding parts.

(ii) Gaseous Oxygen.

(A) General. This guidance is intended to supplement the existing guidance in AC 43.13-2A, Chapter 6. If there are any differences within the two AC's, this guidance shall prevail since it pertains specifically to Part 29 requirements.

(B) System Components.

(1) High Pressure Cylinders. Many installations utilize hospital type cylinders rather than aviation type cylinders. A concern with the hospital type cylinders is the yoke and the hard plastic washer that is commonly used with these cylinders. It is very difficult to properly attach these yokes since the helicopter provides a high vibration environment and no positive lock is provided. Leaks are a continuous problem with this configuration. Yokes * are available for these bottles that provide for a positive lock. Improved washers that provide for a good elastometric seal and include a metal ring to limit crushing the washer are also available. If the hospital type bottles are to be used, only the modified yokes and improved seals should be considered for future installations. The preferred cylinder is the aviation type cylinder with the integral shut-off valve and regulator. All cylinders should be DOT approved.

(2) Lines.

(i) General. Any lines that pass through potential fire zones should be stainless steel.

* (ii) High Pressure. Use of high pressure lines may be necessitated by the use of a pressure regulator that is remote from the cylinder. The intent is to locate the regulator as close as physically possible to the cylinder, and to minimize the use of fittings. Lines of 6-inch lengths are encouraged with 18-inch lengths being the maximum in unusual circumstances. Lines made of stainless steel are recommended. *

(iii) Low Pressure. Although lines may only be subjected to low pressures, if they are located behind upholstery or for any reason are not 100 percent visible during normal operation, they should be solid metal lines or high pressure flexible lines such as Aeroquip 300 series hose, or Stratoflex 124, or 170 series hose assemblies. The so called "green lines" should only be used in locations that are 100 percent visible during normal operation. This would restrict their use to the run between the mask and the bulkhead disconnect in the aircraft cabin. Synthetic lines such as plastic, nylon, or rubber cannot be recommended for applications that will be exposed to continuous pressure (i.e., as opposed to pressurized when needed). These materials can cold flow.

(3) Fittings.

(i) High Pressure. Intercylinder connections are made with regular flared or flareless tube fittings with stainless steel. Usually fittings are of the same material as the lines. Mild steel or aluminum alloy fittings with stainless steel lines are discouraged. Titanium fittings should never be used because of a possible chemical reaction and resulting fire. An example of a series of fittings that has been accepted is the "SS" series Swagelok tube fittings (flareless).

787. RESERVED.

788. SUBSTANTIATION OF COMPOSITE ROTORCRAFT STRUCTURE

a. Reference FAR Sections §§ 29.305, .307, .571, .603, .605, .609, .610, .611, .613, .629, .923, .927, .931, .1529 and Appendix A.

b. Purpose. These substantiation procedures provide a more specialized supplement to the general procedures outlined by AC 20-107A, "Composite Aircraft Structure." These procedures address substantiation requirements for composite material system constituents, composite material systems, and composite structures common to rotorcraft. A uniform approach to composite structural substantiation is desirable, but it is recognized that in a continually developing technical area which has diverse industrial roots, both in aerospace and in other industries, some variations and deviations from the procedures described herein will be both necessary and acceptable. Significant deviations from this material should be coordinated in advance with the Rotorcraft Directorate.

c. Special Considerations. Since rotorcraft structure is configured uniquely and is inherently subjected to severe cyclic stresses, special consideration is required for the substantiation of all rotorcraft structure, including composites. This special consideration is necessary to ensure that the level of safety intended by the current regulations is attained during the type certification process for all structure with special emphasis on composite structure because of its unique structural characteristics, manufacturing quality and operational considerations, and failure mechanisms.

d. Background.

(1) Historically, rotorcraft have required unique, conservative structural substantiation because of unique configuration effects, unique loading considerations, severe fatigue spectrum effects, and the specialized comprehensive fatigue testing required by these effects. Rotorcraft structural static strength substantiation for both metal and composite structure is essentially identical to that for fixed-wing structure once basic loads have been determined. However, rotorcraft structural fatigue substantiation for metals is significantly different from fixed-wing fatigue substantiation. Since AC 20-107A, as developed, applies to both fixed-wing aircraft and rotorcraft; it, of necessity, was finalized in a broad generic form. Accordingly, a need to supplement AC 20-107A for rotorcraft was recognized during type certification programs. One significant difference in traditional rotorcraft fatigue substantiation programs and fixed-wing fatigue programs is the use of multiple full-scale specimen fatigue tests for rotorcraft programs rather than just one full-scale specimen test. Also, constant amplitude, accelerated load tests are typically used rather than spectrum tests because of the high frequency loads common to rotorcraft operations. These rotorcraft fatigue tests have traditionally involved the generation of stress versus life or cycle (S-N) curves for each critical part (most of which are subjected to the cyclic loading of the main or tail rotor system) using a monotonic (sinusoidal) fatigue spectrum based on maximum and minimum service stress values. Unless configuration differences or flight usage data dictate otherwise, the monotonic fatigue spectrum's period is typically based on six ground-air-ground (GAG) cycles for each flight hour of operation. The S-N curves for the substantiation of each detailed part are typically generated by plotting a curved line through three data points (reference *

* draft AC 29-571-X, "Fatigue Evaluation of Transport Category Rotorcraft Structure (Including Flaw Tolerance)"). The three data points selected are a short specimen life (low cycle fatigue), an intermediate specimen life and a long specimen life (high cycle fatigue). Each raw data point is generated by monotonically fatigue testing at least two full-scale specimens (parts) to failure or run out for each data point on the S-N curve. The raw data point values are then reduced by an acceptable statistical method to a single value for plotting to ensure proper reliability of the associated S-N curve. FAA Order 8110.9, "Handbook on Vibration Substantiation and Fatigue Evaluation of Helicopter and Other Power Transmission Systems" and AC 29.571 contain comprehensive discussions of the S-N curve generation process. The rotorcraft S-N curve process contrasts sharply with the fixed-wing process of using a single full-scale fatigue article (usually an entire wing or airframe, which constitutes a single full-scale assembly data point), generic material or full-scale assembly S-N data (e.g., MIL-HDBK-5D for metals, MIL-HDBK-17B for composites, or AFS-120-73-2 for full-scale assemblies), a non-monotonic spectrum and relatively large scatter factors to verify or determine the design fatigue life of the full-scale airplane.

(2) Also, rotorcraft have employed and mass produced composite designs in primary structure (typically main and tail rotor blades) since the early 1950's. This was 10 or more years before composites were type certificated for primary fixed-wing structure in either military or civil aircraft applications (with some notable limited production exceptions, such as the Windecker fixed-wing aircraft). In any case, the early 1950 period was well before a clear, detailed understanding of composite structural behavior (especially in the areas of macroscopic and microscopic failure mechanisms and modes) was relatively common and readily available in a usable format for the average engineer working in this field. It also predated the initial issuance of AC 20-107. Currently, much composite design information is proprietary, either to government, industry or both, and many data gathering methods have not been completely standardized. Consequently, a significant variation from laboratory to laboratory in material property value determination methods and results can exist. The early rotor blade designs (as well as current designs) are by nature relatively low strain, tension structure designs. Also, by nature, these designs are not damage or flaw critical. Thus by circumstance as much as design, early composite rotor blade and other composite rotorcraft designs incorporated an acceptable fatigue tolerance level of safety. In the 1980's, more test data, analytical knowledge, and analytical methodology became available to more completely substantiate a composite design. Current FAR's 27 and 29 contain many sections (reference paragraph a.) to be considered in substantiating composite rotorcraft structure, but this advisory material is needed to supplement the general guidance of AC 20-107A by providing specific rotorcraft guidance for obtaining consistent compliance with FAR sections applicable to rotorcraft.

e. Definitions. The following basic definitions are provided as a convenient reading reference. MIL-HDBK-17, and other sources, contain more complete glossaries of definitions.

(1) AUTOCLAVE. A closed apparatus usually equipped with variable conditions of vacuum, pressure and temperature. Used for bonding, compressing or curing materials.

(2) ALLOWABLES. Both A- basis and B- basis values statistically derived and used for a particular composite design *

(3) BALANCED LAMINATE. A composite laminate in which all laminae at angles other than 0 degrees occur only in ± pairs (not necessarily adjacent).

(4) A-BASIS ALLOWABLE. The "A" mechanical property value is the value above which at least 99 percent of the population of values is expected to fall, with a confidence of 95 percent.

(5) B-BASIS ALLOWABLE. The "B" mechanical property value is the value above which at least 90 percent of the population of values is expected to fall, with a confidence of 95 percent.

(6) BOND. The adhesion of one surface to another, with or without the use of an adhesive as a bonding agent.

(7) COCURE. The process of curing several different materials in a single step. Examples include the curing of various compatible resin system pre-pregs, using the same cure cycle, to produce hybrid composite structure or the curing of compatible composite materials and structural adhesives, using the same cure cycle, to produce sandwich structure or skins with integrally molded fittings.

(8) CURE. To change the properties of a thermosetting resin irreversibly by chemical reaction; i.e., condensation, ring closure, or addition. Cure may be accomplished by addition of curing (crosslinking) agents, with or without catalyst, and with or without heat.

(9) DELAMINATION. The separation of the layers of material in a laminate.

(10) DISBOND. A lack of proper adhesion in a bonded joint. This may be local or may cover a majority of the bond area. It may occur at any time in the cure or subsequent life of the bond area and may arise from a wide variety of causes.

(11) FIBER. A single homogeneous strand of material, essentially one-dimensional in the macro-behavior sense, used as a principal constituent in advanced composites because of its high axial strength and modulus.

(12) FIBER VOLUME. The volume of fiber present in the composite. This is usually expressed as a percentage volume fraction or weight fraction of the composite.

(13) FILL. The 90 degree yarns in a fabric, also called the woof or weft.

(14) GLASS TRANSITION. The reversible change in an amorphous polymer or in amorphous regions of a partially crystalline polymer from (or to) a viscous or rubbery condition to (or from) a hard and relatively brittle one.

(15) GLASS TRANSITION TEMPERATURE. The approximate midpoint of the temperature range over which the glass transition takes place.

(16) HYBRID. Any mixture of fiber types (i.e., graphite and glass).

(17) IMPREGNATE. An application of resin onto fibers or fabrics by several processes: hot melt, solution coat, or hand lay-up.

* (18) LAMINA. A single ply or layer in a laminate in which all fibers have the same fiber orientation.

(19) LAMINATE. A product made by bonding together two or more layers or laminae of material or materials.

(20) LOW STRAIN LEVEL. As used herein, is defined as a principal, elastic axial gross strain level, that for a given composite structure provides for no flaw growth and thus provides damage tolerance of the maximum defects allowed during the certification process using the approved design fatigue spectrum.

(21) MATERIAL SYSTEM CONSTITUENT. A single constituent (ingredient) chosen for a material system (e.g., a fiber, a resin).

(22) MATERIAL SYSTEM. The combination of single constituents chosen (e.g., fiber and resin).

(23) MATRIX. The essentially homogeneous material in which the fibers or filaments of a composite are embedded. The resins used in most aircraft structure are thermoset polymers.

(24) MAXIMUM STRUCTURAL TEMPERATURE. The temperature of a part, panel or structural element due to service parameters such as incident heat fluxes, temperature, and air flow at the time of occurrence of any critical load case, (i.e., each critical load case has an associated maximum structural temperature). This term is synonymous with the term "maximum panel temperature."

(25) POROSITY. A condition of trapped pockets of air, gas, or void within a solid materials, usually expressed as a percentage of the total nonsolid volume to the total volume (solid + nonsolid) of a unit quantity of material.

(26) PRE-PREG, PREIMPREGNATED. A combination of mat, fabric, nonwoven material, tape, or roving already impregnated with resin, usually partially cured, and ready for manufacturing use in a final product which will involve complete curing. Pre-preg is usually drapable, tacky and can be easily handled.

(27) RESIN. An organic material with indefinite and usually high molecular weight and no sharp melting point.

(28) RESIN CONTENT. The amount of matrix present in a composite either by percent weight or percent volume.

(29) SECONDARY BONDING. The joining together, by the process of adhesive bonding, of two or more already-cured composite parts, during which the only chemical or thermal reaction occurring is the curing of the adhesive itself. The joining together of one already-cured composite part to an uncured composite part, through the curing of the resin of the uncured part, is also considered for the purposes of this advisory circular to be a secondary bonding operation. (See COCURING).

(30) SHELF LIFE. The length of time a material, substance, product, or reagent can be stored under specified environmental conditions and continue to meet all applicable specification requirements and/or remain suitable for its intended function.

(31) STRAIN LEVEL. As used herein, is defined as the principal axial gross strain of a part or component due to the principal load or combinations of loads applied by a critical load case considered in the structural analysis (e.g., tension, bending, bending-tension, etc.). Strain level is generally measured in thousandths of an inch per unit inch of part or microinches/per inch (e.g., .003 in/in equals 3000 microinches/inch).

(32) SYMMETRICAL LAMINATE. A composite laminate in which the ply orientation is symmetrical about the laminate midplane.

(33) TAPE. Hot melt impregnated fibers forming unidirectional pre-preg.

(34) THERMOPLASTIC. A plastic that repeatedly can be softened by heating and hardened by cooling through a temperature range characteristic of the plastic, and when in the softened stage, can be shaped by flow into articles by molding or extrusion.

(35) THERMOSET (OR CHEMSET). A plastic that once set or molded cannot be re-set or remolded because it undergoes a chemical change; (i.e., it is substantially infusible and insoluble after having been cured by heat or other means).

(36) WARP. Yarns extended along the length of the fabric (in the 0 degree direction) and being crossed by the fill yarns (90 degree fibers).

(37) WORK LIFE. The period during which a compound, after mixing with a catalyst, solvent, or other compounding constituents, remains suitable for its intended use.

f. RELATED REGULATORY AND GUIDANCE MATERIAL.

	Document	Title
(1)	AC 20-95	"Fatigue Evaluation of Rotorcraft Structure
(2)	AC 20-107	"Composite Aircraft Structure"
(3)	AC 21-26	"Quality Control for the Manufacture of Composite Materials"
(4)	MIL-HDBK-17B (28 Feb 88)	"Polymer Matrix Composites Volume 1: Guidelines"

g. PROCEDURES FOR SUBSTANTIATION OF ROTORCRAFT COMPOSITE STRUCTURE. The composite structures evaluation has been divided into eight basic regulatory areas to provide focus on relevant regulatory requirements. These eight areas are: (1) fabrication requirements; (2) basic constituent, pre-preg and laminate material acceptance requirements and material property determination requirements; (3) protection of structure; (4) lightning protection; (5) static strength evaluation; (6) damage tolerance and fatigue evaluation; (7) dynamic loading and response evaluation; and (8) special repair and continued airworthiness requirements. Original as well as alternate or substitute material system constituents (e.g., fibers, resins, etc.), material systems (combinations of constituents and adhesives), and composite designs (laminates, cocured assemblies, bonded assemblies, etc.) should be qualified in accordance with the methodology presented in the following paragraphs. Each regulatory area will be addressed in turn. It is important to remember that proper certification of a composite structure is an incremental, building block process which involves phased FAA involvement and incremental approval in each of the various areas outlined herein. It is strongly recommended that a FAA certification team approach be used for composite structural substantiation. The team should consist of FAA engineering, the MIDO inspector(s), the associated Designated Engineering Representatives (DER's), the associated Designated Manufacturing Inspection Representatives (DMIR's), and cognizant members of the applicant's organization. Personnel who are composites specialists (or are otherwise knowledgeable in the subject) should be primary team member candidates. Once selected, it is recommended that team meetings be held periodically (possibly in conjunction with type boards) during certification to ensure the building block certification process is accomplished as intended.

(1) The <u>first</u> area is the <u>fabrication requirements</u> of § 29.605:

(i) The quality control system should be developed considering the critical engineering, manufacturing, and quality requirements and a guidance standard such as AC 21-26, "Quality Control For the Manufacture of Composite Materials." This ensures that all special engineering, or manufacturing quality instructions for composites are presented, evaluated, documented, and approved, using drawings, process and manufacturing specifications, standards, or other equivalent means. This should be one of the early phases of a composite structure certification program, since this represents a major building block for sequential substantiation work.

(ii) Specific allowable defect limits on, for example, fiber waviness, warp defects, fill defects, porosity, hole edge effects, edge defects, resin content, large area debonds, and delaminations, etc., for a particular material system component, laminate design, detailed part, or assembly should be jointly established by engineering, manufacturing, and quality and the associated inspection programs for defect detection created, validated, and approved. Each critical engineering design should consider the worse-case effects of the manufacturing process (maximum waviness, disbonds, delaminations, and other critical defects) allowed by the reliability limitations of the approved inspection program.

(iii) If bonds or bond lines such as those typical of helicopter rotor blade structure are used, special inspection methods, special fabrication methods or other approved verification methods (e.g., engineering proof tests, reference paragraph g(5)) should be provided to detect and limit disbonds or understrength bonds.

(iv) Structurally critical composite construction fabrication process and procurement specifications, for fabricating reproducible and reliable structure, must be provided and FAA approved early during the certification process and should, as a minimum, cover the following:

(A) <u>Vendor and Qualified Parts List (QPL) Control</u>. Applicants should be able to demonstrate to FAA certification team members (both the manufacturing and inspection district office (MIDO) and FAA engineering) at any * time, that their quality control systems ensure on a continuous basis, that only qualified suppliers provide the basic material constituents or material systems (e.g., pre-pregs) that meet approved material specifications. Recommended guidelines for qualification of alternate material systems and suppliers are contained in MIL-HDBK-17B, Volume I, Section 2.3.2. These methods can also be used, periodically for qualification status renewals of existing material systems and suppliers.

(B) Receiving Inspection and In Process Inspection. Applicants should be able to demonstrate to FAA certification team members (both MIDO and engineering), at any time, that their receiving and in process quality control system provides products which continuously meet approved material and process specifications. Quality systems should be designed with appropriate checks and balances, such that the necessary statistical reliability and confidence levels for the items being inspected (that are specified by engineering) are continuously maintained. This will require periodic standard inspections and engineering characterization tests on basic constituent and material system samples which should be conducted, as a minimum, on a batch-to-batch basis. The periodic testing necessary to maintain the quality standard should be conducted by the applicant on conformed samples and should be FAA-witnessed.

(C) Material System Component Storage and Handling. Applicants should be able to demonstrate to FAA certification team members (both MIDO and engineering), at any time, that their composite material system (or constituent) storage and handling procedures and specifications provide products which continuously meet approved material and process specifications. Quality systems should be designed with appropriate checks and balances, such that the necessary statistical reliability and confidence levels for the items being inspected (which are specified by engineering) are continuously maintained. This should require, as a minimum, periodic inspections to ensure that proper records are kept on critical parameters (e.g., room temperature "bench" exposure, shelf life, etc.) and that periodic basic constituent and material system characterization tests are conducted, on a batch-to-batch basis. The periodic testing necessary to maintain the quality standard should be conducted by the applicant on conformed samples and should be FAA-witnessed.

(D) Statistical Validation Level. It is necessary to maintain the minimum required statistical validation level of the quality control system (which should be specified for each critical item or constituent by the approved quality and engineering specifications). The statistical validation level should be defined and approved early in certification. Also, approval and proper usage should be continuously maintained during the entire procurement and manufacturing cycles.

(v) Alternate fabrication and process techniques should be approved and should comply with § 29.605. Any alternate techniques should provide at least the same level of quality and safety as the original technique. Any changes should be presented and FAA-approved well in advance of the change's production effectivity.

(2) The second area is the basic raw constituent, pre-preg, and laminate material acceptance requirements and material property determination requirements of §§ 29.603 and 29.613. These criteria require application of the critical environmental limits such as temperature, humidity, and exposure to aircraft fluids (such as fuel, oils, and hydraulic fluids), to determine their effect on the performance of each composite material system. Temperature and humidity effects are commonly considered by coupon and component tests utilizing preconditioned test *

specimens for each material system selected. Material "A" & "B" basis allowable strength values and other basic material properties (based on MIL-HDBK-17B, or equivalent) are typically determined by small scale tests, such as coupon tests, foruse in certification work. In the case of composites, determination of these basic constituent and material system properties will almost invariably involve the submittal, acceptance and use of company standards. This is currently necessary because MIL-HDBK-17B has not completed development of "B" basis allowables for inclusion in the handbook. Also, test methods vary somewhat from manufacturer to manufacturer; therefore, individual company results will exhibit some scatter in final material property values. Any company standard which is approved and used should meet or exceed related MIL-HDBK-17B requirements. Material structural acceptance criteria and property determination should, as a minimum, include the following:

(i) Property characterization requirements of all material systems (e.g., pre-pregs, adhesives, etc.) and constituents (e.g., fibers, resins, etc.) should be identified, documented, and approved. These requirements, once approved, should be placed in all appropriate procedures and specifications (such as those in g(1) above).

(ii) Moisture conditioning of test coupons, parts, subassemblies, or assemblies should be accomplished in accordance with MIL-HDBK-17B, other similar approved methods or per FAA approved programs.

(iii) The maximum and minimum temperatures expected in service (as derived from test measurements, thermal analyses on panels and other parts, experience, or a combination) should be determined and accounted for in static and fatigue strength (including damage tolerance) substantiation programs considering associated humidity induced effects.

(iv) The glass transition temperature, Tg, is an important characteristic parameter of amorphous polymers, such as epoxies. It is the temperature below which the polymer behaves like a "glassy" solid and above which it behaves like a "rubbery" solid, i.e., it is the temperature at which there is a very rapid change in physical properties. In actuality, the change from a hard polymeric material to a rubbery material takes place over a narrow temperature range. A composite material will experience a drastic reduction in matrix controlled mechanical material properties when loaded in this temperature range. Since the resin (matrix) is the critical structural constituent in a composite and since Tg exceedance is critical to structural integrity; Tg determination is necessary. The Tg margin methodology of MIL-HDBK-17B, Section 2.2.2.1, should be implemented, i.e., the wet glass transition temperature (Tg) should be 50 degrees Fahrenheit higher than the maximum structural temperature (see definition). For any type of resin or adhesive, an acceptable temperature margin using MIL-HDBK-17B techniques (e.g., consideration of limited high temperature excursions) or equivalent methodologies based on tests and/or experience should be established and approved early in the certification process. In no case should structural strength be degraded below limit load capability on a maximum world wide high temperature day.

(v) Local design values should be established by analysis and characterization tests and approved for specific structural configurations (point designs) which include the effects of stress risers (e.g., holes, notches, etc.) and structural discontinuities (e.g., joints, splices, etc.). Proper determination of these values for full-scale design and test should be considered one of the most * critical building blocks in substantiating and evaluating a composite structure. These transitional load transfer areas typically produce the highest stresses (and strains) and serve as the nucleation sites for many of the failures (including those due to the relatively low interlaminar strength of composites) that occur in service in a full-scale part or assembly. Small scales tests (such as coupon, element, and subcomponent tests), or equivalent approved testing programs, and analytical techniques should be carefully designed, prepared, and approved to evaluate potential "hot spots" and provide accurate simulations and representations of full-scale article stresses and strains in the critical transition areas. Proper certification work in this area will ensure initial safety and continued airworthiness in full-scale production articles.

(vi) The design strain level for each major component and material system should be established and approved such that specified impact damage considerations are defined and properly limited. The effects of the approved strain levels should be established for each composite material using small scale characterization tests and the results should be used to establish or verify the maximum allowable design strain level for each full-scale article. The maximum allowable design strain values selected should also take into account the reliability and confidence levels established for the relevant portions of the quality control system. This methodology is necessary because the amount and size of flaws in the production article may restrict the allowable level of design strain. In a no-flaw-growth design, the maximum specified impact damage and manufacturing flaw size at the most critical location on the part will be a major factor in determining the maximum allowable elastic strain. This design approach is currently selected for nearly all civil and most military applications; since, under normal conditions, only visual inspections are required in the field (unless unusual external damage circumstances such as a hail storm occur) to maintain the initial level of airworthiness (safety). However, many military applications because of their demanding missions, employ scheduled field non-destructive inspection (NDI) maintenance, (such as comparative ultrasonics) to ensure that flaw growth either does not occur, is controlled by approved structural repair, or by replacement of affected parts. To date, civil applications have not been presented that desire a flaw growth, phased NDI approach. Therefore, selection of the full-scale article's design strain limit based on small scale tests for a no flaw growth design is seen to be extremely important.

(vii) Composite and adhesive properties should be determined such that detrimental structural creep does not occur under the sustained loads and environments expected in service. Small scale characterization tests (such as coupon, element, and subcomponent tests) and analysis, which verify and establish the full-scale design criteria and parameters necessary to ensure that detrimental structural creep in full-scale structure does not occur in service, should be conducted early in certification and should be FAA-approved.

(viii) Material allowable strength values for full-scale design and testing should be developed using the coupon procedures presented in MIL-HDBK-17B or equivalent. At least three batches of material samples should be used in material allowable strength testing. Company standards should be prepared, evaluated and FAA approved early in certification (as part of the building block process), that reflect the material property determination considerations recommended in MIL-HDBK-17B on a equal to or better than basis. *

* (3) The <u>third</u> area is the <u>protection of structure</u> as required by § 29.609. Protection against thermal and humidity effects and other environmental effects (e.g., weathering, abrasion, fretting, hail, ultraviolet radiation, chemical effects, accidental damage, etc.) should be provided, or the structural substantiation should consider the results of those effects for which total protection is impractical. Determination and approval of worst-case or most conservative operating limits, and damage scenarios should be accomplished. Appropriate flammability and fire resistance requirements should also be considered in selecting and protecting composite structure. Usually a hazard analysis is conducted early in certification which identifies the various threats and threat levels for which protection must be provided. This data is then used to construct and submit for approval the methods-of-compliance necessary to provide proper structural protection.

(4) The <u>fourth</u> area is the <u>lightning protection</u> requirements of § 29.610. Protection should be provided and substantiated in accordance with analysis and with tests such as those of AC 20-53A and FAA Report DOT/FAA/CT-86/8. For composite structure projects involving rotorcraft certified to earlier certification bases (which do not automatically include the lightning protection requirements of § 29.610), these requirements should be imposed as special conditions. The design should be reviewed early in certification to ensure proper protection is present. The substantiation test program should also be established, reviewed and approved early to ensure proper substantiation.

(5) The <u>fifth</u> area is <u>the static strength evaluation</u> requirements of §§ 29.305 and 29.307 for composite structure. Only conservative proven methods of static analysis and failure criteria should be employed. The material stress-strain curve should be clearly established, at least through the ultimate design load, for each composite design. Composite structure should be statistically demonstrated, incrementally, through a program of analysis, coupon tests, minor component ultimate load tests and major component ultimate load tests. The static strength substantiation program should consider all critical loading conditions for all critical structure including residual strength and stiffness requirements after a predetermined length of service, e.g., end of life (EOL) (which takes into account damage and other degradation due to the service period). Analytical reports and tests should consider all possible failure modes and should include the critical, allowable effects of:

(i) Environment (reference paragraphs 2 and 3.)

(ii) Service Life (residual limit strength and stiffness demonstration.)

(iii) Load path loss (fail-safe analysis and limit strength demonstration.)

(iv) The standard fabrication process and its variability.

(v) Impact damage expected during service up to the established threshold of detectability of the field inspection methods to be employed.

(vi) Point design and structural discontinuity considerations (e.g., stress risers, joints, etc.) *

(vii) Unless the ultimate strength of each critical bonded joint can be reliably substantiated in production by NDI techniques (or other equivalent, approved techniques), then limit load capability must be guaranteed by either of the following or a combination thereof:

(A) The maximum disbond of each critical bonded joint which will carry limit load is established by test, analysis, or both. Disbonds greater than these values are typically prevented by design features. (B) Each critical bonded joint on each production article should be proof tested to the critical limit load.

(viii) For static strength analysis laminae and laminate "A" and "B" basis allowables (determined in accordance with paragraph (2)) should be used subject to the following conditions unless lower material properties are required by point design considerations (e.g., stress risers, joints, etc.) stiffness requirements (e.g., flutter or vibration margins), fatigue strength (including damage tolerance), or other overriding considerations.

(A) When applied loads are distributed through a single load path or single member within an assembly, the failure of which would result in the loss of the structural integrity of the component involved or inability of the rotorcraft structure to carry limit load, the part should be designed, analyzed, and tested using "A" basis allowables.

(B) Redundant (fail-safe) structures in which the failure of individual elements would result in applied loads being safely redistributed to other load carrying members without exceeding the limit load capability of the rotorcraft structure may be designed, analyzed, and tested using "B" basis allowables.

(6) The <u>sixth</u> area is the <u>fatigue evaluation</u> requirements of § 29.571. The fatigue evaluation method for the rotorcraft being certified should consider damage tolerance in accordance with AC 20-107A.

(i) The safe-life method for composite structure as defined in AC 20-107A is a flaw tolerant safe-life method (e.g., the test specimens consider inherent production flaws and impact damage (reference paragraph (7)(ii)).

(ii) Large area disbonds, weak bonds, delaminations, or other defects should be considered in tests or be prevented or be limited by appropriate flaw tolerant special design features and by special manufacturing, maintenance, and inspection procedures. Special attention should be assigned to all pure bond lines (reference paragraph (5)).

(iii) Non-fail-safe or partially fail-safe dynamic component structure, which may employ bond lines as the only load path, should be designed to relatively small previously approved values of elastic, ultimate strain for the material system utilized, and should be subjected to full-scale S-N curve testing. Six or more specimens are recommended, as part of the substantiation process. Where practical, flight-by-flight spectrum testing should be used.

(iv) All critical safety of flight composite structure must be designed to be flaw (damage) tolerant. Environment degradation and in-service damage critical values are typically included in the flaw tolerance evaluation. All other key factors, such as material selection, manufacturing, and quality assurance controls, and in-service inspection and maintenance, as noted previously, are also to be accounted for.

(v) The fail-safe design features of the rotor heads and blade retention systems, other critical primary composite structure, and point design features (e.g., bonded metal-to-composite joints) should be assessed and appropriate inspection programs provided to prevent catastrophic failure from flaw/damage propagation.

(vi) The method of generating S-N curves using approved raw data should be demonstrated, evaluated, and approved.

(vii) Any limited life items must be identified and placed in the Airworthiness Limitations section of the maintenance manual in accordance with § 29.571.

(viii) Load spectra, load truncation methods and all other major aspects of the fatigue evaluation are documented in test proposals and approved.

(ix) Flaw growth rates (from initial detectability to the established value for residual strength) must be previously established and closely monitored during substantiation. This data should be used to establish special phased inspections and maintenance intervals for critical structure, as required.

(7) The <u>seventh</u> major area is the <u>dynamic loading and response</u> requirements of § 29.629 for vibration and resonance frequency determination and separation for aeroelastic stability and stability margin determination for flutter critical flight structure. Critical parts, locations, excitation modes, and separations are to be identified and substantiated. This substantiation should consist of analysis supported by tests and tests which account for repeated loading effects and environment exposure effects on critical properties, such as stiffness, mass, and damping. Initial stiffness, residual stiffness, proper critical frequency design, and structural damping are provided as necessary to prevent vibration, resonance, and flutter problems.

(i) All vibration and resonance critical composite structure are identified and properly substantiated.

(ii) All flutter-critical composite structure are identified and properly substantiated. This structure must be shown by analysis to be flutter free to 1.1 Vne (or any other critical operating limit, such as V_d, for a VSTOL aircraft) with the extent of damage for which residual strength and stiffness are demonstrated.

(iii) Where appropriate, crash impact dynamics considerations should be taken into account to ensure proper crash resistance and a proper level of occupant safety for an otherwise survivable impact. *

(8) The eighth area is the special repair and continued airworthiness requirements of §§ 29.611, 29.1529, and FAR Part 29 Appendix A for composite structures. When repair and continued airworthiness procedures are provided in service documents (including approved sections of the maintenance manual or instructions for continued airworthiness) the resulting repairs and maintenance provisions must be shown to provide structure which continually meets the guidance of paragraphs (1) through (7) of this AC paragraph. All certification based repair and continued airworthiness standards, limits, and inspections must be clearly stated and their provisions and limitations defined and documented to ensure continued airworthiness. In general, no composite repair should be attempted which is out of scope to repairs stated in an approved Structural Repair Manual (SRM) without an engineering design approval by a qualified FAA representative (DER or staff engineer). The following minimum criteria should be met in any acceptable composite repair:

(i) The repair should be permanent.

(ii) The repair should restore the structure to the required strength and stiffness.

(iii) The repair should restore all functional requirements.

(iv) The repair should have a negligible weight penalty.

(v) The repair should be aerodynamically compatible.

(vi) The repair materials should be compatible in all essential aspects with the parent materials.

In summary, primary composite structure is an especially critical structure that requires a clearly defined, phased approval (building block) certification process. This process should involve the entire project certification team from a project's start to its finish so that proper certification is continuously and ultimately achieved. Also, in some special cases, involving new advanced state-of-the-art composite technology, an issue paper may be necessary. However, in the majority of cases (using current composite materials and design philosophy) the applicant's acknowledged use of this advisory material (as recorded in the type board minutes) should eliminate the need for a separate issue paper. *

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722. <u>§ 29.1517 (through Amendment 29-21) LIMITING HEIGHT-SPEED ENVELOPE</u>.

a. <u>Explanation</u>.

(1) This section requires that the height-velocity (HV) envelope developed in compliance with § 29.79 of the performance requirements be established as an operating limitation for Category A rotorcraft.

(2) For rotorcraft with FAR Part 29 and CAR Part 7 certification bases prior to Amendment 29-21, this section requires that the HV envelope be established as an operating limitation for Category B rotorcraft as well as Category A. The rule was revised by Amendment 29-21 to allow the HV envelope to be provided as performance information rather than as a limitation for rotorcraft meeting the revised § 29.1 Category B requirements. In addition, supplemental type certificates have been approved which allow Category B rotorcraft meeting the revised § 29.1(f) requirements to move the HV envelope from the limitations section to the performance section of the Rotorcraft Flight Manual (RFM). (See paragraph 763 of this AC.)

b. <u>Procedure</u>. The limiting height-speed envelope developed in accordance with § 29.79 should be established as an operating limitation or as performance information to be included in the RFM in accordance with §§ 29.1583(f) and 29.1587(b)(6). (See paragraphs 72, 763, and 765 of this AC for additional information.)

723. <u>§ 29.1519 (through Amendment 29-19) WEIGHT AND CENTER OF GRAVITY</u>.

a. <u>Explanation</u>. This rule requires that weight and center of gravity (c.g.) combinations which are substantiated structurally and also found satisfactory during flight tests (per §§ 29.25 and 29.27) must be established as operating limits. A related portion in § 29.1583(c) further requires that weight and c.g. limitations be entered in the Rotorcraft Flight Manual Limitations Section. Both maximum and minimum weight must be established as operating limitations along with the corresponding longitudinal and lateral centers of gravity for each condition. Weight and c.g. limits are discussed in more detail in paragraphs 43 and 44 of this advisory circular.

b. <u>Procedures</u>.

(1) The results of shifts in center of gravity with fuel burn should be evaluated. If it is possible to take off within the approved loading envelope and subsequently burn fuel to a condition which is significantly beyond the approved weight/c.g. envelope, then there should be appropriate instructions in the loading and/or operating procedures of the RFM to avoid this condition.

(2) Typical loading conditions should not result in weight/c.g. combinations outside of approved limits. A minimum of two loadings, appropriate to the helicopter configuration, should be evaluated. These should include critical combinations of maximum/minimum variables for fuel, passengers, and crew. If this results in loading outside approved limits, special interior placarding or cautionary information should be provided in appropriate sections of the Rotorcraft Flight Manual.

724. §29.1521 (through Amendment 29-24) POWERPLANT LIMITATIONS.

a. Explanation.

(1) This rule requires that the various parameters and operating conditions listed under each type of operation be evaluated and established as operating limitations. The procedures for establishing and verifying each powerplant limitation are discussed in the powerplant section of this AC. This rule requires that powerplant limitations be established for four specific types of operation or power ratings: takeoff, continuous, 2 1/2-minute, and 30-minute. Additional limitations are required to account for engine and transmission cooling and minimum required fuel grade. The 2 1/2-minute and 30-minute limitations are optional requirements intended for use only on multiengine rotorcraft after failure of one engine. These limits are generally referred to as one-engine-inoperative (OEI) limitations.

(2) It is important to differentiate between the rotorcraft powerplant limitations and the engine limitations as established under Part 33. For some parameters, these two limits may be identical, but frequently the engines will be capable of exceeding the maximum limitations substantiated for the combined powerplant installation. Limitations established according to this rule may not exceed the engine limitations established in accordance with Part 33 but may be less than the Part 33 limits as desired by the applicant.

b. Procedures.

(1) Determine the limiting parameters for each required power rating according to the requirements of Part 29, Subpart E, Powerplant. (See applicable paragraphs of this AC for detailed procedures.)

(2) Provide the limitations established according to this rule to the rotorcraft crew through placards in accordance with § 29.1541, instrument markings in accordance with § 29.1549, and in the Rotorcraft Flight Manual Limitations Section in accordance with § 29.1583(b). (See paragraphs 763 and 781 of this AC.)

725. § 29.1522 (through Amendment 29-17) AUXILIARY POWER UNIT LIMITATIONS.

a. Explanation.

(1) Any APU installed in a rotorcraft will have operating limitations which have been developed by design and testing. These APU operating limitations become part of the operating limitations for the rotorcraft.

(2) TSO-C77 establishes the minimum performance standards and limitations which gas turbine APU's should meet in order to be identified with the TSO marking.

b. Procedure.

(1) Limitations for APU's which meet the requirements of TSO-C77 will be contained in the APU model specification and in one or more manuals containing instructions for the installation, operation, servicing, maintenance, repair and overhaul of the APU. Data from these documents which are required by the TSO, should be included in the rotorcraft flight manual (RFM) and in maintenance manuals, as appropriate.

(2) APU's which do not meet the requirements of TSO-C77 should have the design and operating limitations defined and included in the operating limitations section of appropriate rotorcraft manuals. TSO-C77 can be used as a guide to identify and develop the detailed data which will be included in the rotorcraft flight and maintenance manuals.

726. § 29.1523 MINIMUM FLIGHTCREW.

a. Explanation.

(1) This rule requires that the minimum crew necessary to show compliance with the requirements of Part 29 or for safe operation of the rotorcraft be established as an operating limitation.

(2) The determination of minimum crew requirements is typically based on a subjective pilot assessment of the crew requirements for safe operation of each rotorcraft design. Certain regulations, such as the requirements for instrument flight rules (IFR), have specific quantitative differences between single-pilot and two-pilot requirements. However, most often the minimum crew requirement will be based on more subjective considerations such as location of necessary controls, pilot workload to accomplish required tasks, type of operation, and overall complexity of the rotorcraft design.

(3) Minimum crew requirements for the same type design may vary with the kind of operation. Many rotorcraft have been approved for a single-pilot crew for visual flight rules (VFR) operations but require a two-pilot crew for IFR operations. Other kinds of operations that may require more than one crewmember to meet type certification requirements are night operations, operations into known icing conditions, operations in falling and blowing snow, extended overwater operations, and external load operations.

(4) It is important to distinguish between the minimum crew requirements for compliance with Part 29 type certification regulations and the minimum crew requirements of the various operating regulations (Parts 61, 91, 121, 133, 135, and 137). A rotorcraft may be type certified for a minimum crew of one and still be required to have a crew of two or more by the operating regulations for certain types of operation or by the workload associated with an operating environment. Therefore, an applicant should carefully consider the possible operational uses of any rotorcraft design and become familiar with the applicable operating regulations as well as the type certification requirements early in the design process.

(5) The applicant is encouraged to contact the responsible FAA type certification office as early in the design phase as possible to initiate the qualitative assessment process. Cockpit layout drawings, instrument panel mockups, and full-scale cockpit mockups can be used to determine if required controls are accessible and to begin the pilot workload assessment for certain operations.

b. Procedures.

(1) General.

(i) A systematic evaluation and test plan is required for any new or modified rotorcraft. The methods for showing compliance should emphasize the use of acceptable analytical, simulation, and flight test techniques. The crew complement should be studied through a logical process of estimating, measuring, and then demonstrating the workload imposed by a particular cockpit design. When the minimum crew requirements have been determined, they should be included in the limitations section of the Rotorcraft Flight Manual in accordance with § 29.1583(d).

(ii) Appropriate analysis should be conducted by the applicant early in the design process. The specific method(s) of analysis should be selected on the basis of its predictive validity, sensitivity, reliability, applicability to the particular cockpit configuration, and availability of a suitable reference for comparison.

(2) Analytical Approach.

(i) One analytical approach defines workload as a percentage of the time available to perform tasks (Time Line Analysis). This process may be applied to an appropriate set of flight segments in which operationally important time constraints can be identified. This method is useful for evaluation of cockpit changes relating to overt pilot work such as control movements and data inputs. The generally accepted practice involves careful selection of the limited set of flight scenarios and time segments that represent the range of operational requirements (including the range of normal and nonnormal procedures.) Time line analysis yields useful data when tasks must be performed within operationally significant time constraints. The adequacy of this method is very much dependent on an accurate determination of the time available. Absolute standards are not available for interpretation of obtained time required scores, but such records can be used to identify high or simultaneous workload demands for later testing in a simulator or aircraft, and comparisons can be made with overt workload demands in proven aircraft. However, the impact of cockpit changes on planning and decisionmaking is difficult to quantify by this method. *

(ii) The most frequently used basis for deciding that a new design is acceptable is a comparison of a new design with previous designs proven in operational service. By making specific evaluations using the acceptable human factors techniques, and comparing new designs to a known baseline, it is possible to proceed with confidence that the changes incorporated in the new designs accomplish the intended result. When the new cockpit is considered, certain components may be proposed as replacements for conventional items, and some degree of rearrangement may be contemplated. New avionics systems may need to be fitted into existing panels, and newly automated systems may replace current indicators and controls. As a result of this evolutionary characteristic of the cockpit design process, there is frequently a reference cockpit design, which is usually a conventional aircraft that has been through the test of operational usage. If the new design represents an evolution, improvement attempt, or other deviation from this reference cockpit, the potential exists to make direct comparisons. Service experience should be researched to assure that any existing problems are understood and not perpetuated.

(iii) If preliminary analysis by the certification team identify potential problem areas, these areas should receive more extensive evaluation and data collection in order to verify compliance with § 29.1523. These concerns should be adequately addressed in the manufacturer's demonstration plan when submitted to the FAA.

(iv) If the new design represents a significant change in level of automation or pilot duties, analytic comparison to a reference design may have lessened value. Without a firm data base on the time required to accomplish both normally required and contingency duties, more complete and realistic simulation and flight testing will be required.

(3) Testing.

(i) In the case of the minimum crew determination, the final decision is reserved until the rotorcraft has been flown by experienced flight test pilots trained and current in the aircraft. More assurance is derived from actual flight tests than from earlier simulator tests or other synthetic or computer model procedures.

(ii) The test program should address the workload functions and factors listed below. For example, an evaluation of communications workload should include the basic workload required to properly operate the aircraft in the environment for which approval is sought. The goal is to evaluate workload with the proposed crew complement during realistic operating conditions, including representative air traffic and weather.

(A) Basic workload functions. The following basic workload functions are considered:

(1) Flight path control.

(2) Collision avoidance.

(3) Navigation.

(4) Communications. *

(5) Operation and monitoring of aircraft engines and systems.

(6) Command decisions.

(B) Workload factors. The following workload factors are considered significant when analyzing and demonstrating workload for minimum flight crew determination:

(1) The accessibility, ease, and simplicity of operation of all necessary flight, power, and equipment controls, including emergency fuel shutoff valves, electrical controls, electronic controls, and engine controls.

(2) The accessibility and conspicuity of all necessary instruments and failure warning devices such as fire warning, electrical system malfunction, and other failure or caution indicators. The extent to which such instruments or devices direct the proper corrective action is also considered.

(3) The number, urgency, and complexity of operating procedures with particular consideration given to the specific fuel management schedule imposed by center of gravity, structural or other considerations of an airworthiness nature, and to the ability of each engine to operate at all times from a single tank or source which is automatically replenished if fuel is also stored in other tanks.

(4) The degree and duration of concentrated mental and physical effort involved in normal operation and in diagnosing and coping with malfunctions and emergencies.

(5) The extent of required monitoring of the fuel, hydraulic, electrical, electronic, deicing, and other systems while en route.

(6) The actions requiring a crewmember to be unavailable at his assigned duty station, including: observation of systems, emergency operation of any control, and emergencies in any compartment.

(7) The degree of automation provided in the aircraft systems to afford (after failures or malfunctions) automatic crossover or isolation of difficulties to minimize the need for any flight crew action to guard against loss of hydraulic or electric power to flight controls or to other essential systems.

(8) The communications and navigation workload.

(9) The possibility of increased workload associated with any emergency that may lead to other emergencies.

727. <u>§ 29.1525 (through Amendment 29-24) KINDS OF OPERATION</u>.

This rule states that the kinds of operation to which the rotorcraft is limited are established by demonstrated compliance with applicable certification requirements (primarily flight) and the equipment requirements established for that kind of operation. The basic flight characteristics requirements of Part 29 are suitable for day VFR approval. Additional night considerations appear in § 29.141(c) and in the operating rules. IFR requirements are addressed in § 29.141(c) and Appendix B to Part 29. Additional IFR equipment requirements are contained in the operating rules. Icing certification criteria are contained in paragraph 386 of this AC. External load requirements for certification may be found in §§ 29.25(c) and 29.865(c) in addition to Part 133. Related § 29.1525(d) further requires that the approved kinds of operation must be listed in the operating limitations section of the Rotorcraft Flight Manual. The equipment that is necessary for a specific kind of operation other than basic day VFR operation should also be listed in the limitations section of the RFM.

728. § 29.1527 (through Amendment 29-19) MAXIMUM OPERATING ALTITUDE.

a. Explanation. This rule requires that the maximum altitude for operation of the helicopter must be established as an operating limitation. The rule is intended to establish en route altitude as an operating limit. The requirements for maximum takeoff and landing altitude are contained in other portions of the rule. (See discussion in paragraph 81a(2)(ii) of this advisory circular.) The en route limit may be established by any of the preceding subparts of the rule involving flight, structural, powerplant, equipment or related functional requirements of those subparts. Maximum operating altitude is ordinarily specified initially by the manufacturer and substantiated throughout the type certification program by each engineering discipline. Maximum operating altitude must be established in terms of pressure altitude unless the pilot is provided with some equally functional means of observing specified altitude limits (e.g., a density altitude indicator if maximum altitude is specified in terms of density altitude). A related requirement in § 29.1583 specifies that maximum operating altitude must be established as an operating limitation in the Rotorcraft Flight Manual and further that any limiting factors must be identified and explained.

b. Procedures. Each FAA engineering discipline must assure that data and testing are adequate to properly substantiate and qualify all critical components to the maximum operating altitude of the helicopter. The design or maximum substantiated altitude should be specified in the Type Inspection Authorization. The flight test program must include at least one test flight to the maximum approved altitude and this flight must include functional testing of all critical aircraft components. Due to specific requirements in § 29.21(b), no extrapolation of these results is allowed.

(4) Markings of essential visual position indicators must be obvious and within view of required crewmembers. Landing gear markings normally include indications for down, intermediate/unsafe, and up. Accepted symbology has included arrows for up/down indications, crosshatching for intermediate/unsafe, various combinations of colored lights, and combinations of all of the above. Cockpit presentation is further discussed in paragraph 301. Emergency controls which should be marked in red include those used for firewall/emergency fuel shutoff, landing gear blowdown/emergency release, fire extinguishers, float activation, cargo hook release and fuel dump. The method of operation of emergency controls must be clearly marked. In the case of switches and buttons, the method of operation is often inherently obvious without dedicated labeling.

(5) The two most obvious means of displaying landing gear operating speed are use of a placard or an appropriate mark in the airspeed indicator.

748. § 29.1557 (through Amendment 29-26) MISCELLANEOUS MARKINGS AND PLACARDS.

a. Explanation.

(1) This section specifies the markings and placards associated with baggage, cargo, ballast, seats, fuel, oil, and emergency exits.

(2) The data contained in these markings and placards must conform to the approved type design of the rotorcraft.

b. Procedure.

(1) The placard for baggage and cargo compartment limitations should clearly state all limitations which apply to that compartment. The limitations may apply to what is carried, the dimensions, exact location, and maximum weight allowed. The placard should be located in a place where it cannot be obstructed and is clearly visible before or after opening the compartment. For ballast, the placard should state the type of ballast permitted (lead plate, shot bags, etc.), the exact placement, if applicable, and the maximum allowable weight. If there are other limitations which are applicable to these compartments, they should be clearly stated.

(2) Seats in rotorcraft are designed to meet vertical descent loads which have been established to insure a certain level of occupant survivability in the event of a hard landing or crash. To meet these load requirements, 170 pounds was established as the minimum occupant design weight. If the seat was designed and certified to an occupant weight lower than 170 pounds, the seat must carry a placard in a conspicuous place, which limits the weight of the seat occupant to the certified weight.

(3) The fuel and oil filler opening markings are self-explanatory.

(4) Emergency exit placards must be so distinctive and clear that they are easily identified and understood under extreme and intense circumstances by individuals who have little or no familiarity with aircraft escape procedures. *

749. § 29.1559 (through Amendment 29-24) LIMITATIONS PLACARD.

a. Explanation.

(1) The content and location requirements on the placard are specified in the standard. The content and information in the placard has changed significantly as a result of associated and complementary changes in the airworthiness rules and the maintenance and operating rules.

(2) By adoption of FAR Part 29 in 1965, the standard (and its predecessor CAR Part 7) required compliance with the operating limitations in the approved Rotorcraft Flight Manual.

(3) With the adoption of an Airworthiness Limitations Section for the maintenance manual as stated in § 29.1529 of Amdt. 29-4, the content of the placard was changed significantly to require compliance with the requirements in that section.

(4) Amendment 29-20, issued in 1980, adopted standards requiring "Instructions for Continued Airworthiness" (maintenance manual). This manual may include an Airworthiness Limitations section which is segregated and an approved part of the manual. The maintenance and operating rules, §§ 43.16, 91.163(c), and other operating rules require compliance with the Airworthiness Limitations Section. Other airworthiness standards were adopted for airplanes, engines, and propellers to similarly require Instructions for Continued Airworthiness and an Airworthiness Limitations Section. See paragraph 729 of this AC for further information. The limitations placard standard was not changed by this amendment.

(5) Amendment 29-24 adopted a significant change for the placard. The placard must be in clear view of the pilot and must provide a convenient cockpit presentation of the approved types of operation for each aircraft. Other operating and maintenance rules referenced in the previous paragraph provided the basis for much of the change in the placard content.

b. Procedures.

(1) A placard (or durable decal) must be legible to the pilot and located in clear view of the pilot. If two pilots are required, a single placard may satisfy the standard. This aspect will be evaluated by a test pilot. The TIR should contain a compliance check entry.

(2) The placard must specify the kinds of operations such as VFR, IFR, day, night, or icing for which the particular rotorcraft is equipped and approved if Amdt. 29-24 applies.

(3) The placard content for older designs is related to the rotorcraft certification basis. If the rotorcraft type design has an "FAA-approved" and segregated Airworthiness Limitations Section of the maintenance manual, the limitations placard may be revised to comply with the new standard. The certification basis should be changed in conjunction with the placard change.

766. § 29.1589 (through Amendment 29-19) LOADING INFORMATION.

a. Explanation. Control of the rotorcraft weight and balance is an operational function, and is the responsibility of the operator. However, instructions necessary to enable loading of the rotorcraft within the established limits of weight and center of gravity, and to maintain the loading within such limits are required by the operating regulations, and inclusion of such loading instructions in the Rotorcraft Flight Manual is required by § 29.1583(c). Approved loading instructions, therefore, must be presented in the Rotorcraft Flight Manual, and at the option of the applicant, may be included in the approved portion or may be included in the unapproved portion.

b. Procedures.

(1) For the purpose of the flight manual, distinction is made here between the loading instructions required by the certification requirements of Part 29, and the weight and balance data required by the operating requirements. The former prescribed information is applicable to the rotorcraft type, and is subject to FAA approval as flight manual material.

(2) For compliance with the noted requirements, it is necessary for the applicant to develop weight and balance data and loading instructions as necessary to satisfy the needs of both certification and operation. In order to consolidate in one document information on rotorcraft loading, however, it is recommended that the weight and balance data be developed to include appropriate loading instructions, and that both be included in the Rotorcraft Flight Manual as an "unapproved" section entitled, "Weight and Balance." Such a section should include the following statement as a note: "In accordance with FAA procedures, the detail weight and balance data of this section are not subject to FAA approval. The loading instructions of this section, however, have been approved by FAA as satisfying all requirements for instructions on loading of the rotorcraft within approved limits of weight and center of gravity, and on maintaining the loading within such limits."

(3) An actual or specimen weight and balance section should be included in the initial submittal of the manual. Weight and balance data for each particular rotorcraft need not be submitted as flight manual material.

(4) The weight and balance material outlined below is believed to be adequate for rotorcraft with conventional loading and fuel-management techniques. For rotorcraft which necessitate redistribution of fuel (other than normal consumption) to maintain loading within prescribed limits, the material should be amplified as necessary.

(i) Weight Limits. Contained in limitations section of the flight manual.

(ii) Center of Gravity Limits. Contained in the limitations section of the flight manual.

(iii) Dimensions and Datum Line Locations. The dimensions and relative location of rotorcraft features associated with weighing and loading of the rotorcraft and with weight and balance computations should be described and/or illustrated.

(iv) Equipment List. The rotorcraft should be defined or described sufficiently to identify the presence or absence of optional systems, features, or installations that are not readily apparent. In addition, all other items of fixed and removable equipment included in the empty weight should be listed.

(v) Fuel and Other Liquids. Fuel and other liquids, including passenger-service liquids that are included in the empty weight, should be identified and listed together with information necessary to enable ready duplication of the particular condition.

(vi) Weight Computations. Computations of the empty weight and empty-weight c.g. location should be included.

(vii) Empty Weight and Empty-Weight Center of Gravity Location. Statement of these values should be included.

(viii) Loading Schedule. Loading schedule should be included, if appropriate.

(ix) Loading Instructions. Complete instructions relative to the loading procedure, or to use the loading schedule, must be included.

(x) Special Consideration. Consideration should be given to the lateral center-of-gravity loading instructions when various kits such as a side mounted hoist are installed.

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MISCELLANEOUS AIRWORTHINESS

775. IFR CERTIFICATION.

a. Explanation. Requirements for Instrument Flight Rules (IFR) have been incorporated into Part 29, Appendix B, utilizing a regulatory format. Various information from previous interim standards, procedures, test techniques, and acceptable means of compliance for helicopter IFR flight are included in the following sections.

b. Procedures.

(1) General.

(i) The certified instrument flight envelope may be more restrictive than the Visual Flight Rules (VFR) envelope in terms of weight, center of gravity, speed, altitude, or rate of climb and descent. The approved envelope should be operationally practical such that it does not impose constraints with which the crew has difficulty complying.

(ii) Controllability requirements are to be met from 0.9 V_{MINI} to 1.1 V_{NE}. Stability requirements must be met where specified. Stability devices are to be designed to allow safe flight following a failure. The evaluating pilot should assure that all equipment and devices installed for IFR, including reasonable failures of that equipment, do not compromise the VFR approval for that rotorcraft. An example of this would be a stability system failure that caused loss of swashplate or tail rotor control travel when failed in a hardover condition. If the device remains in the hardover position after the stability system is turned off, control capability may be compromised. Cyclic controllability tests at high speed and at the limiting rearward flight condition, or tail rotor tests in sideward flight at high altitude, may reveal a lower control capability and a more restrictive envelope. In addition, controllability testing should be accomplished with the control rigging set at the most adverse production tolerance for the test condition; e.g., minimum forward swashplate for high speed testing.

(2) Trim. Compliance with the IFR trim requirement may be met by use of a magnetic brake with a recentering button, an electrically driven trim system activated by a "beeper" type control, or other means, so long as the system does not introduce any objectionable discontinuities in the force gradient or otherwise result in objectionable flight characteristics. Trim release devices should be free of objectional stick jump. Electrically driven trim systems should have a smooth change in force with a rate compatible with the normal helicopter maneuvers. Only the cyclic trim control must exhibit positive self-centering characteristics. Collective and directional controls are not required to incorporate positive self-centering characteristics, but these controls should not move when released by the pilot (adjustable friction devices are satisfactory); however, for systems which use hydraulic or pneumatic dampers, control motion following release by the pilot is permitted during the time interval when the damper is bleeding off. Movement of the trim controls should produce a similar effect on the rotorcraft in a plane parallel to that of the control motion. The control system free play and breakout force must

be evaluated to assure a close and direct correlation between control input (force and deflection) and rotorcraft response (pitch, roll, yaw, and heave (vertical motion)), and to permit small, precise changes in flight path. If trim control is provided in a stability augmentation system (SAS), the control should be of such design and so installed that any failure will not create a hazardous condition. If an inadvertent out-of-trim condition can be developed, its effect on the rotorcraft should be investigated. These failures or malfunctions should be investigated as outlined in (6) "Stability Augmentation Systems" which follows. The controls for this trim function should be installed such that, the controls should operate in the plane and with the sense of motion of the rotorcraft. Each control means should have the direction of motion plainly marked thereon or adjacent to the control. *

(3) Static Longitudinal Stability.

(i) Positive static longitudinal stability is a key IFR requirement which assures a self correcting airspeed response and allows a pilot to recognize any substantial change in speed. Very shallow force gradients can be approved for systems with low deadband and low friction. Systems with significant friction and deadband require much steeper force gradients to be acceptable. The longitudinal force gradient can be determined by either one of two methods. The most commonly used method measures the forces on the ground (with hydraulic and electric ground power units if required). The force applied to the cyclic stick and the cyclic stick displacement are measured and a plot of stick force verses displacement in each direction is obtained. The longitudinal static stability tests are conducted in the air as described in paragraph 85. The trim system should be on during the test and trimmed at the trim speed. After each end point, the cylic should be allowed to slowly return to the trim position. When all the force is released from the cyclic stick and the airspeed has stabilized, note the airspeed. The airspeed should return to within 10 percent or 10 knots, whichever is less, of the trim speed. An alternate method of determining the longitudinal stick force stability is to measure the force on the cyclic stick in flight using a hand held force gage or other force measuring instrumentation. The in-flight technique is the same as the first method. Testing should be accomplished at a minimum of two altitudes. One altitude should be low enough to assure limiting power is attained. Another should be at or near the maximum approved altitude. Reasonable interpolation is allowed. If no marginal areas are apparent interpolation over a 10,000-foot altitude range is considered reasonable.

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