ORDER

8110.6

Consolidated Reprint
Includes CHG 1
June 15 1982

TYPE CERTIFICATION - REVIEW CASES

6 JAN 71

DEPARTMENT OF TRANSPORTATION

FEDERAL AVIATION ADMINISTRATION

Distribution: WFS-3, RFS-4, FIA-0(Minimum) Initiated By: FS-103

RECORD OF CHANGES

DIRECTIVE NO. 8110.6

CHANGE TO BASIC	SUPPLEMENTS			OPTIONAL USE	CHANGE TO BASIC	SUPPLEMENTS			OPTIONAL USE
CHG 1									

FOREWORD

1. PURPOSE. This order revision converts the Type Certification - Review Case Handbook, FS P 8110.3, to the current four-digit agency directives numbering system and sets forth the policy and procedures for processing Review Cases.

2. DISTRIBUTION. This order is being distributed to the branch level in Washington Flight Standards offices; to the section level in regional Flight Standards offices; and to all International Aviation Field Offices. See also paragraph 5, chapter 1 regarding dissemination of Review Cases.

3. CANCELLATION. FS P 8110.3 and Changes 1 through 59 are cancelled.

4. REQUESTS FOR INFORMATION. Requests for information concerning this order should be transmitted to the Chief, Engineering and Manufacturing Division, Attention: FS-103.

Acting Director
Flight Standards Service

TABLE OF CONTENTS

Page No.

CHAPTER 1. GUIDELINES 1

1. Purpose 1
2. Regional Request 1
3. Manufacturer Request 1
4. Washington Action 1
5. Dissemination of Review Case Information 1
6.-20. Reserved

CHAPTER 2. PROCEDURES 3

21. General 3
22. Preparation of Review Case Requests 3
23. Preparation of Review Cases 3
24. Issuance of Review Cases 3
25.-30. Reserved

CHAPTER 3. REVIEW CASE ISSUANCES 5

31. General 5

Review Cases

No. 1. Bell Helicopter Company - Interpretation of CAR 6.11(e)(3) 7

No. 2. Boeing Model 707-100 Series Aircraft SR-422 versus SR-422B - Landing Climb Requirements 11

No. 3. Douglas Model DC-8 Aircraft - Rudder Pedal Nosewheel Steering 15

No. 4. North American Model NA-265 Aircraft - Type Certificate Limitations 23

No. 5. Douglas Model DC-8F Aircraft - Interpretation of CAR 4b.260 and 4b.350(e) 27

No. 6. Grumman Model G-150 Aircraft - Maximum Passenger Capacity Increase 31

No. 7. Sikorsky Model S-58 Helicopter - Blade Inspection Method 35

		Page No.
No. 8.	Beech Model H18 Aircraft - Proposed FAA Participation in Certification	41
No. 9.	Douglas Model DC-8F Aircraft - Dual Airspeed Limitations	45
No. 10.	Vertol Model 107-II Helicopter - CAR 7.382(a)	49
No. 11.	Cessna Models 310E through 310H, 320, and 320A Aircraft - Emergency Exit Provisions	51
No. 12.	Vertol Model 107-II - Water Certification and Emergency Evacuation	57
No. 13.	Boeing Model 727 Aircraft - Tail Light (Rear Position Light) Installation	61
No. 14.	Cessna Aircraft Company - Electric Clock Installation	63
No. 15.	Boeing 707-300B Series Aircraft - Antiskid Inoperative and Reverse Thrust Performance Credit	67
No. 16.	Cessna Model 336 Aircraft - Design Flap Speed and Intermediate Flap Settings	73
No. 17.	Piper Model PA-28-180 Aircraft - Normal Category Characteristics	77
No. 18.	Douglas Model DC-8-50 Aircraft - One or More Thrust Reversers Inoperative	85
No. 19.	Lockheed Model C-141A Aircraft - Maximum Allowable Speed Display	91
No. 20.	Hughes Model 269A Aircraft - High Altitude Tests	95
No. 21.	Champion Model 402 Aircraft - Sources of Power for Gyroscopic Indicators	105
No. 22.	Vertol Model 107-II Helicopter - Equivalent Safety Proposal	111
No. 23.	Removal of Pilot Chute	121

		Page No.
No. 24.	Lockheed Model 300 (C-141A) Aircraft - Hydraulic Fluid Quantity Gauges	125
No. 25.	Boeing Model 727 Aircraft - Longitudinal Static Stability	129
No. 26.	Bell Model 206 Helicopter - Interpretation of CAR 6.384 and 6.483	133
No. 27.	Boeing Model 727 Aircraft - Longitudinal Control During Flap Retraction	139
No. 28.	Boeing Model 727 Aircraft - Horizontal Stabilizer Stop Settings	149
No. 29.	Beech Model 65-90 Aircraft - Source of Power for Ejector/Instrument Vacuum System	153
No. 30.	Boeing-Vertol Model V107-II - Proposed 4000 Feet Extrapolation Method for CAR 7 Category A H-V Test Data	157
No. 31.	Mooney Models M20C and M20E Aircraft and Mitchell Model AK-123 Radio Coupler - Use of Communication/Navigation and Autopilot/Coupler Equipment in Part 3 Aircraft IFR Operations	165
No. 32.	Lockheed Model 300 (C-141A) Aircraft - Reverse Thrust Performance Credit	169
No. 33.	Piper Model PA-28 Aircraft - Power Adequacy Indication for Electric Turn and Bank Instrument	171
No. 34	Hughes Model 369 Helicopter - Powerplant Instruments	177
No. 35.	Douglas Model DC-9 Aircraft - Minimum Flight Crew Determination	181
No. 36.	Lear Jet Model 23 Aircraft - Oil Temperature and Pressure Gages	183
No. 37.	Bell Model 204B Helicopter - Maximum Rotorcraft - Load Combination Weight (Part 133)	191

		Page No.
No. 38.	Lear Jet Model 23 Aircraft - Stick Shaker-Pusher Installation	195
No. 39.	Boeing Model 707-353B Aircraft - ITT Model 3544 Distance Measuring Equipment	203
No. 40.	Fairchild C-82 Aircraft - Installation of an Auxiliary Jet Engine	209
No. 41.	Lockheed Model 382 (C-130E) Aircraft - Static Directional Stability	215
No. 42.	Aero Commander Model 1121 Aircraft - Effect of Engine Unbalance	221
No. 43.	Mooney Aircraft - Augmented Lateral Stability System	227
No. 44.	Douglas Model DC-9 Aircraft - Folding Armrests to Clear Type III Exit Area	233
No. 45.	Grumman G-159 Aircraft - Installation of the Sperry SP-40 Autopilot	237
No. 46.	Prue Super Standard Glider Visibility Requirements	243
No. 47.	deHavilland Dove DH-104 Aircraft - Applicability of CAR 3.381(b)	247
No. 48.	Lockheed Model 382 (C-130E) Aircraft - Design Landing Descent Velocity	251
No. 49.	Douglas Model DC-9 Aircraft - Amended Proposal for Folding Armrests to Clear Type III Exit Area	257
No. 50.	Boeing Models 707 and 720 Aircraft - Abbreviated Fire Detector System	259
No. 51.	Cessna Models 180 and 182 Aircraft - Cooling Test for Approval of Turbosupercharger Installation	267
No. 52.	Lockheed Model 382 (C-130E) Aircraft - Rotor Containment for Air Turbine Motor and Refrigeration Cooling Turbine	275

		Page No.
No. 53.	Boeing Model 707-300C Aircraft - Access Aisleway at Type II Emergency Exits	279
No. 54.	Twin Beech Aircraft - Compliance with FAR 23, Section 23.735 - Brakes	281
No. 55.	Hiller Models UH-12L4, UH-12L and UH-12E-L Helicopters - Maximum Rotorcraft Load Combination Weight (FAR 133)	283
No. 56.	Lockheed Model 382 (C-130E) Aircraft - Deletion of Fire Shields from Aluminum Engine Mount Support Beam	287
No. 57.	Use of Autopilot as Stability Device When Pitch Trim Compensator is Inoperative - DC-8	293
No. 58.	Fire Resistant Fuel Lines in the Engine Compartment of the Cessna Model 188 Airplane	305
No. 59.	Fire Protection of Oil System of Swearingen SA-26T Airplane	311
No. 60.	Cessna Crafted Full-Flow Oil Filters Used on Cessna Aircraft	315
No. 61.	Review of CAR 6.328 With Particular Regard to Lockheed CL-286 Helicopter Power Control System	321
No. 62.	Labeling of Fire Panel - Boeing 707-321C	329
No. 63.	Application of CAR 4b.356(e) to the Aero Commander Model 1121	331
No. 64.	Boeing Request for Review Case on 737 APU Fuel Icing Protection	335
No. 65.	Piper PA-31 Aircraft/CAR 3.359 Amendment 3-7	337
No. 66.	Interpretation of FAR 25.857(e)(1) for the Boeing 747 Aircraft	341
No. 67.	Fire Resistant Requirements for Oil Cooler Located in Fire Zone of Windecker Model AC-7 Airplane	345

		Page No.
No. 68.	Cutout Switch for Elevator Trim Systems on Ted Smith Aircraft Company Models 600 and 601	351
No. 69.	Induction System Alternate Air Door Requirement for Windecker Model AC-7 Airplane with Fuel Injection Engine	355

CHAPTER 1. GUIDELINES

1. PURPOSE. This chapter sets forth the guidelines governing the initiating and applicability of a Review Case.

2. REGIONAL REQUEST. A Review Case should be requested by regional personnel whenever:

   a. The applicant requests a review, by Washington, of a determination of compliance made by the region in conjunction with a specific application for a type certificate or a supplemental type certificate, or

   b. When regional personnel first encounter a specific design feature compliance determination problem for one or more models, and for which the existing standards are considered inadequate or inappropriate.

3. MANUFACTURER REQUEST. A manufacturer may make a request for review to the region or to Washington. Requests received directly from a manufacturer will be referred to the appropriate region.

4. WASHINGTON ACTION. Upon receipt of a request for a Review Case the Washington Office will evaluate the facts in the matter and set forth its findings and applicability of the findings over the Director's signature. In those situations wherein the same problem arises and involves another aircraft type similar to that considered in a previous Review Case, the findings of the related Review Case may be applied at the discretion of the Regional Office. In each such instance the Region is to advise the Washington Office of such application and recommend as to the need for a regulatory change.

5. DISSEMINATION OF REVIEW CASE INFORMATION. Copies of Review Cases will be made available to the public upon request. All such requests should be transmitted to the Chief, Engineering and Manufacturing Division, Attention: FS-103. Certain portions of the Review Cases may be deleted prior to release. These include:

   a. Information furnished by any person that would not customarily be released to the public.

   b. Information furnished and accepted in confidence.

   c. Opinions, advice, deliberations or recommendations made in the course of developing the official action by the agency.

6-20. RESERVED.

21. GENERAL. The basic procedures set forth below should be followed in handling Review Cases. Exceptions may be allowed in cases or urgency or special importance.

22. PREPARATION OF REVIEW CASE REQUESTS. Requests prepared by the Regional Office must include adequate documentation. Such documentation must include, but not be limited to, the necessary regulation(s), the problem or differences of opinion, background material, analysis and conclusion by both the region and the applicant, etc. Such requests should be transmitted to the Chief, Engineering and Manufacturing Division, Washington, D. C.

23. PREPARATION OF REVIEW CASES. Washington will complete its analysis, assembly of additional pertinent information, prepare findings, and complete clearances and coordination with other pertinent offices. If necessary, a conference will be arranged in Washington, before actual Review Case issuance, between the representative(s) of the manufacturer concerned, the cognizant Regional Office, and the Engineering and Manufacturing Division. In such instances, advance notice will be given to the parties concerned.

24. ISSUANCE OF REVIEW CASES. Upon completion of the review and the findings made, the Review Case will be assigned a number and issued to the region affected for implementation, and to other holders of this Handbook for information purposes.

25.-30. RESERVED.

CHAPTER 3. REVIEW CASE ISSUANCES

31. GENERAL. This chapter contains each Review Case, including the findings by Washington, for which a request has been made for review of a compliance determination with one or more specific regulatory type certification requirements. Upon receipt of the issuance, the region affected is to take the action indicated therein.

REVIEW CASE NO. 1. BELL HELICOPTER COMPANY REQUEST FOR REVIEW OF A DECISION BY THE SOUTHWEST REGION RELATED TO INTERPRETATION OF SECTION 6.11(e)(3) OF THE CIVIL AIR REGULATIONS (Issued 11 June 1963)

1. INTRODUCTION

The Bell Helicopter Company has requested (through the medium of personal representations) review and reconsideration of a decision by the Southwest Region relating to interpretation of Civil Air Regulations, Section 6.11(e)(3), which was instrumental in establishing the certification basis applicable to a modification kit for Bell Model 47G-2 helicopters.

2. CHRONOLOGICAL HISTORY

a. By letter to the Southwest Region dated January 31, 1961, Bell Helicopter Company submitted for Federal Aviation Agency approval a modification kit identified as Bell Service Instruction No. 384. The kit modifications, when incorporated in a Bell Model 47G-2 helicopter, result in a helicopter having the gross weight, power, performance, dimensions, and altitude capability of the previously approved Model 47G-2A, except that certain fire protection details are omitted.

b. Because of the extensive changes in the kit modification, and the similarity of the resulting helicopter to the Model 47G-2A, the Southwest Region has decided that the kit should be certificated in accordance with the regulations applied to the Model 47G-2A. The Southwest Region advises that this decision was given verbally to Mr. Schroder of Bell.

c. Civil Air Regulations, Section 6.11(e)(3), provides that a new application for type certificate shall be required and the regulations, together with all amendments thereto, effective on the date of the new application shall be made applicable for the case where a change in design, configuration, power or weight which the Administrator finds is so extensive as to require a substantially complete investigation of compliance with the regulations.

d. It appears that Bell's request for reconsideration is based upon their opinion that the changes in design configuration, power, and weight which are involved in the modification kit for the Model 47G-2 helicopter are not so extensive as to require a substantially complete investigation of compliance with the regulations. Otherwise, if Bell agreed that the changes were so extensive as to require a substantially complete investigation of compliance with the regulations, the proper course of action would have been the submittal of a petition for exemption from the provisions of Section 6.11(e)(3). Since they have not done this, we must conclude that the issue at hand is whether or not the changes involved are such as to require a substantially complete investigation of compliance with the regulations.

3. FACTS IN THE CASE

The pertinent approved models of the Bell Model 47G series are as follows:

a. Model 47G-2. This helicopter was certificated under Type Certificate H-1, approved January 20, 1955, on the basis of CAR, Part 6, dated May 24, 1946.

b. Model 47G-3. This model incorporated a turbosupercharged Franklin Model 6VS-335 engine with higher power limits, changes in the airframe and rotor system, and increased gross weight. This model was certificated under Type Certificate 2H-3 approved March 17, 1960, on the basis of CAR, Part 6, dated December 1956, plus amendments through 6-4.

c. Model 47G-2A. This helicopter is identical to the Model 47G-3 except that it utilizes a Lycoming VO 435 engine. This model was certificated under Type Certificate 2H-3 approved December 10, 1960, on the same regulation basis as the Model 47G-3.

d. Bell Models 47G-3 and 47G-2A differ from each other only in respect to engine installation. Although power limits are the same for both, the Model 47G-3 is capable of higher altitude operation because of the turbosupercharging feature. Both models are growth versions of the Model 47G-2 and differ from it in the following major respects:

(1) Power increased from 200 to 240 horsepower for takeoff.

(2) Airspeed limit VNE, sea level, increased from 100 to 105 m.p.h.

(3) Maximum weight changed from 2450 to 2850 pounds.

(4) Installed metal rotor blades from Model 47J helicopter, with four feet increase in diameter.

(5) Lengthened fuselage and tail rotor drive shaft.

(6) Incorporated fire protection changes in engine compartment areas.

e. There are many and various changes in the airworthiness standards of CAR, Part 6, dated December 1956, plus amendments through 6-4 as compared with the standards of CAR, Part 6, dated May 1946. However, the only sections of interest in this issue are those relating to engine fire protective features, since the proposed kit would make the helicopter identical in all essential respects to a Model 47G-2A except for the fire protection changes in the engine compartment and certain minor production improvement items. Bell claims that the addition of the fire protection changes would increase the weight by 15 or 20 pounds. They further state that the fire protection changes do not materially contribute to safety because a fire is not likely to occur. It is claimed that no fires have ever occurred on the commercial Model 47 series, although a fire of minor consequence did occur on a military counterpart.

f. With respect to the fire protection changes which were required of both the 47G-2A and 47G-3 models in order to comply with CAR, Section 6.480, it is understood that fire-resistant plumbing, redesigned firewalls and seals, and the substitution of materials for certain parts were involved. The requirements under Section 6.480 prescribe certain features for protection against fire in the engine compartment and are intended to ensure that the main and auxiliary rotors and controls remain operable, the essential rotorcraft structure remains intact, and that the passengers and crew are otherwise protected at least five minutes after the start of an engine fire to permit a controlled autorotational landing.

g. The contention that the fire protective measures proposed for omission do not materially contribute to safety because of infrequent occurrence of powerplant fires is not considered valid. CAR, Part 6, requires only meager fire protection features as compared with CAR, Part 7, for the larger and more powerful transport helicopters. The small, low-power engine installations are generally less complicated, and experience shows there is less likelihood of fire occurring. Therefore, the standards recognize that the occurrence of fire is likely to be rare. If this were not so, the more extensive protective features, such as contained in CAR, Part 7, would have been prescribed.

h. The question as to whether or not the fire protection provisions under both the general and detailed sections of Section 6.480 do or do not materially contribute to safety is not one to be resolved by this review. If Bell Helicopter Company believes that a regulation is inappropriate and improper, they should take action through normal channels available to them to petition for an exemption or otherwise seek amendment to the requirements.

i. Since many of the parts and components comprising the kit were previously approved on either the 47G-2A, 47G-3 or earlier models, it is apparent that very little investigation of compliance with the regulations would be involved in the approval of these changes as now applied in kit form to the Model 47G-2.

j. The provisions of CAR, Section 6.11 (designation of applicable regulations) indicate that the intent is to make a judgment of the extent of changes made to the basic type design. Therefore, the provisions are seen to be applicable to the overall excursion from the basic type design, rather than to a series of design changes which, taken separately, might not be considered to be either extensive or require a substantially complete investigation; but when taken as a total group might be judged to be of that extent. The very existence of CAR, Section 6.11, is recognition that the airworthiness standards will undergo a continual process of revision and improvement as years pass. To ignore the effects of compounded design changes on basic type design would be contrary to the objectives of this section.

k. A review of the extent of investigation of compliance with regulations shows that the modifications to the rotor and drive system involved endurance testing as well as a complete vibratory stress investigation. Performance changes resulting from the increased power were cause for a substantially complete flight performance investigation. The weight changes involved required a substantially new structural substantiation program.

4. CONCLUSIONS

a. Based upon these facts, it is concluded that the original approval of the design changes included in the modification kit did entail a substantially complete investigation.

b. In consideration of the foregoing, it is found that Bell Helicopter Company has not shown that the provisions of the 6.11(e)(3) are not applicable, nor has it shown that the proper level of safety would be provided by an interpretation of Section 6.11(e)(3) which would permit the proposed kit of changes to be eligible for approval under the regulations originally applied to the Model 47G-2 helicopter.

REVIEW CASE NO. 2. BOEING 707-100 SERIES, SR-422 VERSUS SR-422B - LANDING CLIMB REQUIREMENTS (Issued 11 June 1963)

1. INTRODUCTION

The Boeing 707-100 series airplanes are certificated under the performance requirements of SR-422. In comparing the all-engine-operating landing climb requirement, 4T.119, of SR-422 with the corresponding but later requirements of SR-422B, The Boeing Company noted that the required climb gradient has been reduced from 4.0 percent in SR-422 to 3.2 percent in SR-422B. Boeing states that the higher climb gradient requirement of SR-422 results in a severe economic penalty for airline operations at high-altitude airports such as Denver, Colorado. In order to increase the maximum operating landing weight at high-altitude airports under SR-422, Boeing used an alternate reduced landing flap position of 30 degrees which gave higher climb performance but resulted in increased brake and tire wear, as well as longer landing distances. Boeing believes that the 4.0 percent climb requirement of SR-422 unjustly requires a higher level of safety than the later requirements of SR-422B, and results in a severe economic penalty when compared to other model jet transport airplanes which are certificated under SR-422B requirements. Boeing, therefore, requests that the landing climb gradient requirement for the Model 707-100 series airplanes be reduced to 3.2 percent as in SR-422B while remaining under the rest of the performance requirements of SR-422 in other respects.

2. CHRONOLOGICAL HISTORY

a. Boeing letter of December 28, 1961 to the Western Region

This letter introduced Boeing's request and presented the reasons and justification.

b. WE-210 letter of January 8, 1962, to Boeing

This letter acknowledged Boeing's letter of December 28, 1961, and stated that Boeing's request was being evaluated.

c. WE-210 memorandum of January 31, 1962, to FS-100

This memorandum contains a repetition of Boeing's request and presentation, together with copies of the pertinent correspondence between Boeing and the Western Region. WE-210 concluded that Boeing's request could not be granted without the issuance of an FAA exemption, and requested our early concurrence with their stand and whatever comments we had on the subject

d. WE-210 memorandum of February 27, 1962, to FS-100

This memorandum directed our attention to WE-210's memorandum of January 31, 1962, (item C above), for which WE-210 desired an early reply due to a Boeing request.

e. FS-100 memorandum of March 15, 1962, to WE-210

This memorandum acknowledged WE-210's memorandums of January 31, 1962, and February 27, 1962, and informed the Regional Office that Boeing's request would be processed as an Engineering and Manufacturing Division Review Case over FS-1's signature.

3. FACTS IN THE CASE

a. A comparison of the required landing climb gradient in the SR-422 series of regulations is as follows:

Regulation	Required Landing Climb Gradient
SR-422	4.0 percent
SR-422A	3.2 percent
SR-422B	3.2 percent

b. The above table shows that the required landing climb gradient for SR-422A and SR-422B is the same. This point is emphasized in order to establish the intent of the regulations regarding mixing of regulations for type certification purposes. The preamble of SR-422A in the fourth paragraph of the first page clearly states, with respect to the use of portions of SR-422A instead of the entire SR-422 regulations, that it is intended that compliance be shown with all the provisions of SR-422A if used, and it is not intended to permit a showing of compliance with some portions of SR-422A and different portions of SR-422 simultaneously. The same principle would apply to mixing of SR-422B and SR-422 regulations.

c. The following is a direct quotation from the preamble of SR-422A of the portion concerning the mixing of SR-422A and SR-422:

> ".... it is intended that compliance be shown with all the provisions of this regulation and it is not intended to permit a showing of compliance with portions of this regulation and portions of SR-422."

4. CONCLUSIONS

a. We have concluded, as a result of our review, that Boeing's request cannot be granted. The reason for not allowing mixing of these different sets of regulations is that, while certain performance requirements in the later regulations are lower than those in SR-422, the reverse is true with certain other performance requirements, the net result being that the overall level of safety remains about the same for each set of regulations. If the lowest of the respective performance requirements in SR-422, SR-422A, and SR-422B were allowed to be selected as a basis for certification, the resultant level of safety would be greatly reduced in comparison to any one of these special regulations taken in its entirety.

b. In summary, the Boeing proposal cannot be considered as meeting the intent of the pertinent regulations for the above reasons. It is clearly evident, of course, that Boeing has the option of using all of the SR-422B requirements for certification of the 707-100 series airplanes, if the company desires to do so.

REVIEW CASE NO. 3. REQUEST BY DOUGLAS AIRCRAFT COMPANY FOR CREDIT FOR RUDDER PEDAL NOSEWHEEL STEERING ON DC-8 AIRPLANES (Issued 11 June 1963)

1. INTRODUCTION

a. The Douglas Aircraft Company has incorporated a rudder pedal steering system on the DC-8 series aircraft which provides nosewheel steering through the rudder pedals. This is a desirable design feature, not incorporated at this time on any other transport aircraft, in that it improves directional control with no additional effort on the part of the pilot for all ground operations including takeoffs, landings, and taxiing on dry, wet, or slippery runways, and in high winds.

b. Douglas has requested that credit, in the form of lower critical engine failure speeds, be given the DC-8 airplanes incorporating this design feature.

2. CHRONOLOGICAL HISTORY

a. Douglas Aircraft Company letter dated October 21, 1960, to the Director, Flight Standards Service, requesting exemption for DC-8 aircraft from that portion of SR-422B, paragraph 4T.114(a), which requires that the critical engine failure speed V_1 be determined with primary aerodynamic controls alone.

b. Meeting held in Washington on October 27, 1960, by representatives of the FAA Washington Safety Regulations Division, Engineering and Manufacturing Division, FAA Western Region Flight Test Section, and the Douglas Aircraft Company to discuss the Douglas petition for exemption.

c. Letter from the Chief of the Regulations Staff, FAA, Washington, dated December 1, 1960, to the Douglas Aircraft Company advised that their petition for exemption had been reviewed and suggested that the Douglas proposal may be approved under the equivalent safety provisions of CAR 4b.10. This letter also advised Douglas that the request for petition had been referred to the Washington Office of the Engineering and Manufacturing Division for technical evaluation under Section 4b.10.

d. FAA Western Region Flight Test Section memorandum to FAA Washington Flight Test Branch dated June 14, 1961, advised that Douglas was currently submitting a revised proposal to demonstrate critical engine failure speeds V_1 with active rudder pedal steering.

e. Douglas letter dated June 20, 1961, to the FAA Engineering and Manufacturing Division submitted the following proposal for determination of speeds for the DC-8 series 50 airplanes under equivalent safety provisions of CAR 4b.10:

"Douglas Aircraft Company hereby requests that the V_{mcg} for the DC-8 series 50 airplanes be certified under the equivalent level of safety provisions of CAR 4b.10. It is proposed to certify the V_{mcg} with rudder pedal nosegear steering connected under wet runway conditions using elevator control up to the limit of one-hand control capability. It has been shown during these demonstrations that the resulting V_{mcg} provides a level of safety equal to or greater than that attained without the use of the rudder pedal nosegear steering system on both wet and dry runways. It is proposed to use this demonstrated V_{mcg} for all takeoff conditions when no ice, snow, or slugh exists on the runway; at ambient air temperatures above 40°F. with or without precipitation and at any temperature when no ice, snow, or slush exists on the runway and no precipitation is present. For takeoff conditions with ice, snow, or slush on the runway, or with visible precipitation and temperatures below 40°F., it is proposed to use V_{mcg} as demonstrated under existing SR-422B regulations and interpretations."

f. FAA Washington Engineering and Manufacturing Division letter dated June 30, 1961, to Douglas Aircraft Company advised that their proposal had been reviewed and that the FAA Western Region would advise Douglas of the FAA decision.

g. FAA Washington Engineering and Manufacturing Division memorandum dated July 7, 1961, to the FAA Western Region Flight Standards Field Division advised the criteria under which the Douglas proposal would be acceptable. These criteria are as follows:

(1) <u>The minimum V_1 speed tested with rudder pedal nosegear steering connected with nosewheel noticeably light on wet runway</u>.

These minimum V_1 speeds may be used operationally for takeoff conditions as follows:

(a) At all ambient air temperatures on a dry runway

(b) At ambient air temperatures above 40 degrees Fahrenheit on a dry or wet runway (which means no ice, snow, or slush)

(2) <u>The minimum V_1 speed tested with primary aerodynamic controls alone</u>.

These minimum V_1 speeds must be used operationally for takeoff conditions as follows:

(a) When there is ice, snow, or slush on the runway

(b) At ambient air temperatures below 40 degrees Fahrenheit with precipitation

(3) <u>Airplane Flight Manual</u>

The airplane flight manual should clearly describe to the pilot when, and under what circumstances, the various ground minimum control speeds are applicable. In addition, the manual material should indicate very plainly that all of the accelerate-stop distances are still based on dry conditions in accordance with past practice.

h. FAA Western Region Engineering and Manufacturing Branch letter dated July 1, 1961, which transmitted to Douglas the criteria for approval of the rudder pedal steering credit for the DC-8 aircraft.

i. Douglas letter dated January 2, 1962, to FAA Western Region Engineering Branch submitted the following revised proposal for rudder pedal steering credit in determining ground minimum control speeds for DC-8 aircraft:

Runway Surface Condition	Ambient Air Temperature	Applicable V_{mcg} Curve of DC8-A12, 516B	Elevator Control Force Required
Dry	All Temperatures	B	0
Wet	Above 40 Degrees Fahrenheit	B	20 lbs. to 25 lbs. (push)
Wet	40 Degrees Fahrenheit or less	A	0
Snow, slush or ice	All temperatures	A	0

j. FAA Western Region Flight Test Section memorandum dated January 8, 1962, transmitted the revised Douglas proposal (i above) together with Douglas substantiating data to FAA Washington Flight Test Branch.

3. FACTS IN THE CASE

a. The current airworthiness requirements, Special Regulation 422B, Section 4T.114(a), requires that the critical engine failure V_1 be "not less than the minimum speed at which controllability by primary aerodynamic controls alone is demonstrated during the takeoff run to be adequate to permit proceeding safely with the takeoff using average piloting skill, when the critical engine is suddenly inoperative."

b. The currently approved critical engine failure speeds for the DC-8 series aircraft were established by test with the rudder pedal steering disconnected and the nosewheel noticeably light on the runway, to simulate the slippery runway conditions envisioned by SR-422B, Section 4T.114(a). Douglas proposes in their January 2 proposal, to retain those speeds for wet runways when temperatures are at or below 40 degrees Fahrenheit (4.5C) and on snow, slush, and ice-covered runways at all temperatures.

c. The January 2 Douglas proposal also requests approval of additional critical engine failure speeds, V_1, for wet runways when temperatures are above 40 degrees Fahrenheit (4.5C) and for dry runways at all temperatures. These speeds were obtained by testing on a wet runway with rudder pedal steering connected and with 20-25 pounds of forward pressure on the elevator control. The proposed speeds correspond to those obtained by tests on a dry runway with zero elevator force and with rudder pedal steering connected.

d. The DC-8 rudder pedal steering is controlled by the rudder pedals and is, therefore, always active whenever the pilot applies rudder (primary directional aerodynamic control) for directional control. Full rudder deflection and rudder pedal steering are attainable with approximately 70 pounds of rudder pedal force.

e. The 20-25 pounds of elevator force results in a nosegear strut condition which is quite light as shown by report DC-8 A12.525. A 20-pound push force results in a nosegear shock strut compression from one to five inches. The total strut travel for full compression is 16 inches. Therefore, the nosewheel is noticeably light on the runway when the 20-25 pounds of elevator force is applied.

During FAA and Douglas testing with instrumented test aircraft, it has been shown that a push force of approximately 15 pounds was a normal pilot reaction in controlling the airplane following an engine failure. During these tests, the pilot was not aware that he was applying a push force. The nosegear strut compression varied between zero to six inches.

f. The Douglas Aircraft Company has been training all operators of the DC-8 aircraft to apply a push force to the elevator control for all takeoffs. It has been verified that airlines are training their DC-3 pilots to apply a push force during all takeoffs.

g. It has been found that the critical engine failure speeds, V_1, proposed by Douglas in their letter dated January 2, 1962, to the FAA Western Region Engineering and Manufacturing Branch, were determined in accordance with the criteria contained in the letter dated July 7, 1961, from the Washington Chief of the Engineering and Manufacturing Division to the Chief of the Flight Standards Field Division, Western Region, and FAA test pilots have found that the speeds can be realized in service by pilots of average skill. FAA test pilots also feel that the rudder pedal steering is a very desirable design feature and that credit should be given when incorporated on any transport design.

h. The proposed airplane flight manual procedures clearly describe conditions under which the various critical engine failure speeds are applicable. Although all manuals state that takeoff and landing performance is based on dry runways, the proposed manual emphasizes that accelerate-stop distances are based on a dry runway condition. The airplane flight manual procedures also inform the pilot that increased forward pressure on the elevator control will provide increasingly effective directional control on the ground. The applicable portions of the flight manual are quoted below:

IN THE LIMITATIONS SECTION:

Engine Failure During Takeoff

"During takeoff, monitor desired takeoff EPR and observe V_1, V_r, and V_2 speeds. The nosewheel should remain firmly in contact with the runway until V_r is obtained...."

"The required takeoff field length is based on stopping on a dry hard surface runway....."

There are two types of V_{mc} ground speeds depending on the runway surface condition presented in this manual.

(1) V_{mcg} (WET-DRY)

V_{mcg} (WET-DRY) is applicable at all temperatures if the runway is dry, and above 4.5C if the runway is wet but free from ice, snow and slush.

(2) V_{mcg} (COLD-WET-ICE)

V_{mcg} (COLD-WET-ICE) is applicable if the runway is wet at temperatures below 4.5C and at all temperatures if there is snow, slush or ice on the runway. It is also conservative for all conditions since V_{mcg} (COLD-WET-ICE) is faster than V_{mcg} (WET-DRY).

IN THE PERFORMANCE SECTION:

Effect of Rudder Pedal Nosewheel Steering on V_{mcg}

"The rudder pedal nosewheel steering feature on the DC-8 provides a reduction in V_{mcg} for all runway surface conditions from the V_{mcg} available with aerodynamic rudder control only. The effectiveness of rudder pedal nosewheel steering can be improved by applying a push force on the control column."

Performance data is shown in this manual for two levels of V_{mcg}. These are:

(1) V_{mcg} (WET-DRY) for use in determining takeoff performance on wet runways which are free from ice, snow and slush at temperatures above 4.5°C and on dry runways at all temperatures. The V_{mcg}'s presented for these conditions, V_{mcg} (WET-DRY), are those obtained with rudder pedal nosewheel steering operating on wet runways, using normal pilot technique, with a positive push force on the control column.

(2) V_{mcg} (COLD-WET-ICE) for use in determining takeoff performance on wet runways at temperatures of 4.5°C and below and on snow, slush and ice-covered runways at all temperatures. The V_{mcg} (COLD-WET-ICE) shown for snow, slush and ice conditions have not incorporated the benefit available due to rudder pedal nosewheel steering.

4. CONCLUSIONS

In consideration of the foregoing, it has been found that, under the conditions for which Douglas is requesting approval, the rudder pedal steering is a compensating feature which results in a level of safety equivalent to that required by Special Regulation No. SR-422B, Section 4T.114(a). Therefore, the Douglas request is granted under the equivalent safety provisions of CAR 4b.10.

REVIEW CASE NO. 4 NORTH AMERICAN AVIATION REQUEST FOR ISSUANCE OF NA-265 TYPE CERTIFICATE WITH DATA SHEET LIMITATIONS OR AUTHORIZATION FOR HORIZONTAL STABILIZER SHORT-TIME REPLACEMENT (Issued 17 July 1963)

1. INTRODUCTION.

North American Aviation has requested the Western Region to issue the type certificate for the NA-265 with special inspection and repair procedure limitations indicated on the type certificate data sheet. The request stems from failures of the horizontal stabilizer skin and ribs which have occurred during flight testing. The Western Region contends that the type design should contain no known adverse or undesirable feature at the time of issuance of the type certificate. As an alternate request, North American may, under the fatigue strength requirements of Part 4b of the Civil Air Regulations, Section 4b.270(a), propose replacement of the skin and ribs after 100-300 hours of flight. The Western Region contends that such periods appear unreasonably low for the airplane and the user in question.

2. CHRONOLOGICAL HISTORY.

a. The Western Region is evaluating the type design for the North American Model 265. Procurement of this aircraft by the United States Air Force is contingent on Federal Aviation Agency type certification under Part 4b. In showing compliance with CAR, Part 4b, the applicant elected to conduct a fatigue evaluation of this aircraft in accordance with Section 4b.270(a). He selected a target aircraft service life of 10,000 hours.

b. The Western Region reported, in a telegram, WE-210 January 301955, that cracks had been found in the horizontal stabilizer skin and ribs. These cracks occurred in all five test aircraft during the flight test program conducted to date. Special inspections and repairs were imposed to keep cracks within reasonable safe limits during the remainder of the FAA certification program. The cause of the cracking has not been identified but acoustical fatigue is considered one possible contributing factor.

c. The following additional information was provided in a telephone conversation with the Western Region on February 7:

(1) Thirty-six aircraft have been delivered to date to the Air Force. The Air Force has instituted a mandatory special inspection of the stabilizers beginning after the first 40 hours of flight, and after each subsequent 100 hours of flight. The results of this inspection are not yet available.

(2) Among four flight test aircraft on which stabilizer cracks have been reported, the installation of new stabilizers has been necessary after 100-200 hours on two separate occasions. One aircraft has been found to have two stabilizer cracks after 430 hours, another has been found to have nine stabilizer cracks after 256 hours, and a third has been found to have 17 stabilizer cracks after 260 hours.

3. FACTS IN THE CASE.

a. The current airworthiness requirements, CAR, Part 4b.300, state that "The airplane shall not incorporate design features or details which experience has shown to be hazardous or unreliable." This requirement dictates that a type certificate should not be issued under the proposed conditions until a thorough evaluation has been completed on the modified horizontal stabilizer design. It is equally unreasonable to issue a type certificate for a design where findings prior to the issuance establish that an FAA airworthiness directive (AD) will be needed shortly after type certification, or where a limitation on the type certificate data sheet is to be incorporated, which is tantamount to an AD.

b. In showing compliance with CAR, Part 4b.270(a), it is incumbent on the applicant to have selected a reasonable target life early in the design program, and evaluate the design against this figure, taking into account the provisions of CAR, Part 4b.270(a)(1), and the recommendations of Section 1 of Appendix H to CAR 4b. Based on the adverse experience to date, it is highly doubtful the applicant has established proper correlation with the typical loading spectra expected in service, particularly if he now can only substantiate a 100-300 hour replacement period - an unreasonably low period compared with the target life of 10,000 hours originally selected.

c. CAR, Part 4b.270(a), requires that "The structure shall be shown by analysis and/or tests to be capable of withstanding the repeated loads of variable magnitude expected in service." Our oral understanding is that the applicant will contend that the loading spectrum encountered in flight tests conducted to date is more severe than that expected in normal transport use, and therefore that the 100-300 hour point of severe cracking is unduly conservative. However, the intended Air Force usage includes training missions. The severity and frequency of loads experienced in these training missions is expected to equal or exceed those encountered in flight testing.

d. Section 306 of the Federal Aviation Act of 1958 states, "In exercising the authority granted in and discharging the duties imposed by this Act, the Administrator shall give full consideration to the requirements of national defense, and of commercial and general aviation, and to the public right of freedom of transit through the navigable airspace." Since the United States Air Force is procuring these aircraft, it is not in the best interest of national defense for the FAA to certificate this aircraft as proposed.

4. CONCLUSIONS.

In consideration of the foregoing, it is concluded under the provisions of Part 4b of the Civil Air Regulations, Section 4b.300, that:

a. issuance of a type certificate under Part 4b of the Civil Air Regulations shall be withheld for the North American NA-265 aircraft incorporating the present horizontal stabilizer design, and

b. prior to granting a type certificate under Part 4b of the Civil Air Regulations for the North American NA-265, the applicant must substantiate the reliability of the redesigned horizontal stabilizer.

The substantiation of the redesigned horizontal stabilizer shall include:

a. accurate identification of the cause of the cracking and incorporation of this factor in the loading spectrum,

b. conduct of the complete NA-265 functioning and reliability test program with the redesigned horizontal stabilizer installed,

c. in lieu of item "b," conduct of a ground test on the redesigned horizontal stabilizer wherein the loading spectrum determined in item "a" is simulated for flight time corresponding to the total flight time utilized by the applicant and the Federal Aviation Agency in showing compliance with the flight requirements of subpart B of the Civil Air Regulations plus the flight time originally specified for the NA-265 functioning and reliability test program; in addition, at least 25 hours of actual flight time with the redesigned horizontal stabilizer installed, shall be satisfactorily completed.

REVIEW CASE NO. 5 DOUGLAS AIRCRAFT COMPANY REQUEST FOR AN INTERPRETATION OF CIVIL AIR REGULATIONS 4b.260 AND 4b.350(e) RELATIVE TO TYPE CERTIFICATION OF THE DC-8F (Issued 17 July 1963)

1. INTRODUCTION.

The Douglas Aircraft Company, Incorporated, has requested a ruling from the Flight Standards Service for their DC-8F combination cargo-passenger configuration as to whether or not an aisle must be maintained from the flight compartment to the passenger compartment after the aircraft has experienced the crash loading conditions specified in Civil Air Regulation 4b.260, Emergency Landing Conditions - General. Their request also asks for confirmation that accessibility to the door specified by CAR 4b.350(e), Pilot Compartment - General, is only required under normal flight and ground loading conditions but not under the emergency landing conditions specified in CAR 4b.260. The Western Region Engineering and Manufacturing Branch concurs with the statement of the problem and also has requested policy guidance on the case.

2. CHRONOLOGICAL HISTORY.

a. Mr. George Castle, Douglas Aircraft Company, FAA Liaison Engineer, outlined the problem to representatives of the Washington Engineering and Manufacturing Division on December 6, 1961. He pointed out that the question has arisen because of the unique nature of the Douglas DC-8F configuration wherein the cargo compartment separates the pilot compartment from the aft located passenger compartment.

b. Western Region representatives confirmed the need for a policy ruling on the matter during visits to the Washington Office on December 6, 1961, and again in February 1962.

c. Mr. George Castle, Douglas Aircraft Company, Incorporated, requested confirmation of the FAA ruling on the matter in a wire to the Director, Flight Standards Service, on February 21, 1962.

d. The Director, Flight Standards Service, wired Douglas Aircraft Company, Incorporated, on February 23, 1962, that the matter was under study and a reply would be forthcoming by February 28, 1962.

e. Western Region representatives confirmed on February 27, 1962, that:

(1) The tiedown means for cargo retention are designed to withstand the 1.5g side loading condition specified in CAR 4b.260.

(2) The crash net separating the pilot compartment and the cargo compartment would be designed to account for the 9g forward crash load condition specified in CAR 4b.260, and that the dynamic effects associated with cargo movement would be suitably accounted for.

(3) In normal flight and landing the crash net is slack and could be unfastened to gain access to the cargo area through an aisleway consisting of the outer fuselage shell and cargo loading restrictions.

(4) Under a crash condition the net is loaded, thus precluding unfastening of the net.

(5) The dislocation of the cargo and resulting structural deformation during a survivable crash would be such that the aisleway provided and maintained under normal flight and landing to gain access to the passenger compartment, would be blocked, thus precluding access by a flight crew member to the passenger compartment.

f. Mr. L. J. Devlin, Vice President - Director, Engineering and Product Development, Douglas Aircraft Company, Incorporated, wrote to the Director, Flight Standards Service, on March 29, 1962. Mr. Devlin expressed concern about the problem of a ruling being established regarding the aisle. He requested an opportunity to discuss the subject with the Director, should an unfavorable ruling be made.

g. The Director, Flight Standards Service, wired the Western Region on April 30, 1962, that an aisle between the flight compartment and passenger compartment must be maintained subsequent to load factor conditions of CAR 4b.260.

h. The Chief, Engineering and Manufacturing Branch, Western Region, transmitted the conclusion requiring an aisle be maintained after crash loads to the Chief Engineer, Douglas Aircraft Company, Inc. in a letter dated May 8, 1962.

i. Messrs. Strang, Castle, and Adams met with personnel of the Flight Standards Service on May 23, 1962, and presented and discussed their objection to the conclusion that an aisle must be maintained. They were requested to resubmit their case, including technical, economic, and other aspects discussed during the meeting. It was agreed that a resubmittal of all factors would be forwarded from Douglas Aircraft Company in the immediate future.

j. Mr. L. J. Devlin, forwarded to the Director, Flight Standards Service, on July 17, 1962, a report entitled "Post-Crash Crew-Passenger Compartment Aisle Probability Study" for Model DC-8F. A reevaluation of all aspects of adequate provisions for passenger evacuation was carefully considered.

3. FACTS IN THE CASE.

a. The crash barrier between the pilot compartment and the cargo compartment, and the restraint provisions for cargo carried in the compartment will be designed to comply with applicable strength provisions of CAR 4b.260, Emergency Landing Conditions - General, and CAR 4b.359, Cargo and Baggage Compartments, respectively, as outlined in the FS-100 letter of October 24, 1961, to the Douglas Aircraft Company and the FS-100 memorandum of October 12, 1961, to the Western Region.

b. Means exist, under normal flight and ground conditions, whereby the flight crew can gain access through the crash barrier and cargo compartment to the passenger compartment. This is accomplished by unfastening detachable portions of the barrier.

c. The passenger compartment will comply with the provisions of CAR 4b.362, Emergency Evacuation.

d. The CAR 40 operating rules, and in particular, CAR 40.265, Flight Attendant, require that at least one flight attendant be provided by the air carrier on all flights carrying passengers in airplanes of ten-passenger capacity or more.

e. The CAR 41, 42, and 43 operating rules, do not contain a provision similar to that provided in CAR 40.265.

f. Douglas Aircraft Company, Inc., Report Number SM-22611, "DC-8 Flotation Study," revised July 18, 1961, was checked and approved by the Western Region and submitted as requested to the Washington Office for additional review. This review has indicated no fallacies in the ditching analysis.

4. CONCLUSIONS.

In consideration of this request, it is unnecessary that the Douglas Aircraft Company provide flight crew access to the passenger compartment on the DC-8F as implied in CAR 4b.350(e) after the airplane has experienced the emergency landing conditions of CAR 4b.260 providing:

a. Means exist, under normal flight conditions, whereby the flight crew can gain access to the passenger compartment.

b. The crew and passenger areas comply with the provisions of CAR 4b.362 with respect to emergency evacuation.

c. At least one flight attendant be required for CAR 40, 41, 42, and 43 operations. The attendant should be trained and have demonstrated ability to perform all emergency functions, including ditching. The Airplane Flight Manual is to include complete information pertaining to these procedures.

REVIEW CASE NO. 6 GRUMMAN AIRCRAFT ENGINEERING CORPORATION REQUEST TO INCREASE THE MAXIMUM PASSENGER CAPACITY OF THE MODEL G-159 FROM 19 TO 24 PASSENGERS (Issued 17 July 1963)

1. INTRODUCTION.

The Grumman Aircraft Engineering Corporation has requested the Eastern Region to approve an increase in the maximum passenger capacity of their Model G-159 from 19 to 24 passengers. As compensation under the provision of Civil Air Regulations 4b.362(c)(4) for an increase of five passengers, Grumman requests approval to activate the 20-inch by 36-inch floor level cargo door located in the aft right side of the fuselage as a passenger exit. Grumman also proposes to install an evacuation slide on this exit. The Eastern Region contends this exit (Type III dimensions) and the presence of the left forward entrance door (air stair) are sufficient compensation to allow an increase of five passengers, and requests Washington Office concurrence.

2. CHRONOLOGICAL HISTORY.

a. The Grumman G-159 was type certificated by the Eastern Region as a 19-passenger aircraft. The emergency exit aspects were approved on the basis of CAR 4b.10, Eligibility for Type Certificates, as being equivalent to the provisions of CAR 4b.362, Emergency Evacuation, as amended by Amendment 4b-5, effective April 9, 1957. The emergency exit provisions included two pairs of 19 inches by 26 inches elliptically shaped overwing exits and an overhead hatch aft of the crew compartment. Additional openings provided but not considered as emergency exits are: a main entrance air stair door on the left forward side and a rectangular floor level cargo door (20 inches by 36 inches) opening on the right rear side.

b. Under CAR 4b, Amendment 4b-5, for 19 passengers, one pair of Type III exits, plus crew escapement means, was needed. Under the provisions of CAR 4b.362(c)(3) Grumman elected to substitute two pairs of Type IV exits in lieu of the required one pair of Type III exits.

These Type IV exits were elliptical with a major horizontal axis of 26 inches, and a minor vertical axis of 19 inches. Under the provisions of CAR 4b.362(b)(4) Type IV openings are required to be rectangular and not less than 19 inches wide and 26 inches high.

c. Under the provisions of CAR 4b.10, Grumman was required to conduct an evacuation test to establish if the two elliptical exits on each side were reasonably equivalent to one Type III exit on each side. The tests conducted on July 10, 1957, and duly witnessed by the Civil Aeronautics Administration at that time, demonstrated this.

The installation was subsequently approved by the Eastern Region and formed the basis for showing equivallence under CAR 4b.10 with CAR 4b.362(c)(1) in the type certification of the Grumman G-159 as a 19-passenger configuration.

d. Early in 1961, Grumman requested approval from the Eastern Region to increase the capacity of the G-159 from 19 to 29 passengers. Under CAR 4b.362(c)(4), Grumman requested consideration of the presence of the right rear aft cargo access door and the left forward air stair door as compensating factors.

The Eastern Region requested a Washington ruling on this in their memorandum of May 15, 1961. A refusal of the Grumman request was forwarded to the Eastern Region in the Washington reply of June 13, 1961, and subsequently conveyed to Grumman by the Eastern Region. The basis was as follows:

> The table in CAR 4b.362(c)(1) requires for 20 to 39 passengers at least one Type II and one Type IV exit per side. The G-159 has on the left side two exits which Grumman has shown to be equivalent to Type IV exits; one more such exit than required is provided. The main entrance door, however, which contains the air stair was not considered to qualify as an emergency exit due to the mechanical, hydraulic, and electrical complexity of the stair mechanism. On the right side of the fuselage, the 20 inches by 36 inches cargo door fell short of the Type II exit dimension required by CAR 4b.362(c)(1). Therefore, it was concluded that insufficient compensating factors existed to authorize an increase in passenger seating capacity to the maximum of ten permitted under CAR 4b.362(c)(4).

e. Under CAR 4b.362(c)(4), Grumman has recently reapplied to the Eastern Region for approval to increase the passenger seating capacity from 19 to 24 passengers as stated in the introduction. The Eastern Region believes the request is reasonable and in their memorandum of January 29, 1962, has asked for Washington approval.

3. <u>FACTS IN THE CASE.</u>

a. The 24-passenger version complies with CAR 4b.362(a) with respect to a top hatch for crew escapement.

b. The 24-passenger version exceeds CAR 4b.362(c)(1) with respect to the required one pair of Type IV exits on each side as two pairs are provided on each side.

c. The 24-passenger version requires a pair of Type II emergency exits as specified in CAR 4b.362(c)(1). These do not exist, but the authority vested to the Administrator in CAR 4b.362(c)(4) clearly permits approval of an increase in passenger seating capacity up to ten passengers irrespective of CAR 4b.362(c)(1), providing compensating factors in the emergency evacuation means exist. Each side of the aircraft, right and left, must be considered before concluding what, if any, compensating factors are present.

For the right side of the aircraft, the presence of a third opening, heretofore not considered for emergency evacuation, is a compensating factor. As herein considered, it is intended that as many as ten additional occupants may be authorized with the addition of an exit of reasonably high effectiveness and that a lesser number of occupants would be authorized with the addition of a less effective exit. The effectiveness of the additional exit varies with parameters such as: the type, location, and number. The presence of an aft opening, at floor level, and of Type III dimensions (20 inches by 36 inches); the proximity of the last two rows of seats to this exit; the presence of an unobstructed passageway at least 20 inches wide; and the fact that this opening is a third means of egress on the right side of the aircraft, or 50 percent more than the number required, clearly establishes that the exit is an effective means of evacuation. Assuming evacuation is through the right side exits, it is reasonable to increase the passenger seating capacity by five additional persons.

For the left side, we do not consider there are compensating factors present in the emergency evacuation means now provided. There is an additional exit of Type I dimensions which incorporates an air stair door, but this door is not considered acceptable for emergency evacuation (Reference - Item 4, Chronological History). One alternative is for the applicant to qualify the present air stair door at least as a Type II emergency exit as defined in CAR 4b.362(b)(2). This would entail removal of the air stair door and installation of a conventional side hinged door. With this modification, the left side of the airplane would exceed the present minimum requirements specified in CAR 4b.362(c)(1) such that it would be reasonable to increase the passenger seating capacity by five additional persons.

d. The evacuation slide as proposed by Grumman at the 20-inch by 36-inch aft cargo opening is not required under the provisions of CAR 4b.362(e)(7) as the exit is less than six feet from the ground.

4. <u>CONCLUSIONS.</u>

In consideration of the foregoing, it is concluded that the Grumman Aircraft Company's request to increase the passenger capacity on their

Model G-159 from 19 to 24 passengers, is acceptable under the compensating factor provisions of 4b.362(c)(4), providing:

a. The door located in the aft rear side of the fuselage complies with the emergency exit arrangement, marking, and access provisions of CAR 4b.362(e), (f), and (g), respectively.

b. The cargo and baggage compartment in the aft portion of the fuselage and immediately adjacent to the right rear exit complies with the provisions of CAR 4b.260, Emergency Landing Conditions, and CAR 4b.359, Cargo and Baggage Compartments.

c. The passageway leading to the rear exit on the right side is unobstructed and not less than 20 inches wide.

d. The forward main entrance door on the left side of the fuselage is suitably modified to qualify at least as a Type II emergency exit as defined in CAR 4b.362(b)(2).

REVIEW CASE NO. 7. SIKORSKY AIRCRAFT REQUEST TO DELETE THE CURRENT SERVICE LIFE LIMITATION ON THE MODEL S-58 MAIN ROTOR BLADE AND CUFF, TO USE A BLADE INSPECTION METHOD (BIM) TO INDICATE SERVICEABILITY OF THE BLADES, AND TO HAVE UNLIMITED SERVICE LIFE ON THE CUFF, PREDICATED ON VISUAL INSPECTION (Issued 17 July 1963)

1. INTRODUCTION.

Sikorsky Aircraft has requested approval from the Eastern Region to eliminate the mandatory 1000-hour life limitation on the Model S-58 main rotor blade when the blade is equipped with a blade inspection method (BIM), which consists of pressurizing the hollow spar. With BIM incorporated, Sikorsky contends the main rotor blades may be used indefinitely and only blades found unserviceable for further use need be discarded. Sikorsky has also requested approval from the Eastern Region to eliminate the mandatory service life of the Model S-58 main rotor blade cuff, which constitutes the blade attachment fitting, predicated on visual inspection only.

The Eastern Region is of the opinion that the BIM installation on the main rotor blade will provide a level of safety equivalent to that obtained under CAR, Part 6.250, Main Rotor Structure, provided that the inspection interval and the reliability of the method in service are satisfactorily substantiated.

With respect to elimination of the mandatory service life on the cuff attachment, the Eastern Region contends that the present service life should be retained since visual inspection alone will not suffice.

2. CHRONOLOGICAL HISTORY.

a. July 27, 1960 - A fatal accident of the S-58 occurred in civil operation at Chicago, Illinois.

b. July 29, 1960 - A telegraphic airworthiness directive was issued reducing the service life to 1400 hours on the blade and requiring daily X-ray inspections of all blades with more than 1000 hours' time in service.

c. August 2, 1960 - At Fort Rucker, Alabama, a blade fracture was discovered during ground inspection of an Army H-34 after approximately 830 hours' time in service.

d. August 3, 1960 - A telegraphic airworthiness directive was issued amending the directive dated July 29, 1960. This latter airworthiness directive further reduced the service life to 1000 hours on the blade and required a one-time X-ray of the rotor blade spar. This directive was subsequently printed as AD 60-17-3.

e. October 18, 1960 - Sikorsky Aircraft Corporation, by letter SE-9813, to FAA, FS-1120, requested approval of the BIM installation in the S-58 helicopter.

f. December 23, 1960 - The FAA (FS-1120 letter) granted approval of the BIM installation. This approval did not alter the 1000-hour retirement life established by AD 60-17-3.

g. June 30, 1961 - Sikorsky (letter SE-2442) submitted Sikorsky Engineering Report No. SER-58331, Structural Reliability of the S-58 Main Rotor Blade to FS-1120. A copy of the report was subsequently forwarded to FS-120 by FS-1120 memorandum dated August 15, 1961.

h. November 22, 1961 - A meeting was held at Sikorsky Aircraft. Representatives of the Washington FAA Airframe Branch, Engineering and Manufacturing Division of FAA Eastern Region Airframe and Equipment Branch and Sikorsky attended. The discussions pertained in part to structural reliability.

i. January 10, 1962 - EA-212 memorandum to FS-120 requested our comments and concurrence regarding approval of the BIM installation. They concluded the current 1000-hour limitation could be deleted and that the blade could be retired on condition with the BIM installed following complete substantiation of the inspection interval and gage reliability. They also concluded that the service life of the blade cuff attachment could not be predicated on visual inspections as proposed by Sikorsky and the present service life of the cuff would remain in effect.

j. February 19, 1962 - In discussions held with the Navy Department Bureau of Naval Weapons, it was established that the Navy has initiated action to approve the installation of BIM on the military version of the S-58 and considers the main rotor blades to have a life of 3000 hours with BIM installed.

3. FACTS IN THE CASE.

a. The BIM blade inspection method consists of pressurizing the hollow spar of each main rotor blade to ten pounds per square inch. The area pressurized includes the blade attachment to the cuff, but excludes a small portion of the blade tip. BIM is designed to permit inspection personnel to ascertain, through a gage at the root end of the blade spar, that pressure is being maintained and thus no crack exists in the spar and its attachment. Inspection of the blade spar pressure is proposed to be accomplished on a preflight basis.

b. The original certificated service life of the S-58 main rotor blade was 2450 hours based on the procedures outlined in Appendix A to Civil Aeronautics Manual 6.

c. Investigation of the Chicago accident revealed that the main rotor blade failed as result of fatigue. To determine the cause of this fatigue failure, Sikorsky conducted an evaluation of the effects of preloads (quick starts), various finishes, corrosion, adequacy of original flight strain survey, and manufacturing processes. Upon conclusion of this investigation, no positive cause of the failure was found.

d. The Army report of the investigation of the Fort Rucker H-34 incident concluded that this fracture was caused by an undetected nonmetallic inclusion in the spar. To preclude further incidents of this type, refined manufacturing inspection methods were introduced, both at the material supplier and at Sikorsky.

e. Following the investigation as to the cause of the catastrophic failure at Chicago, Illinois, Sikorsky requested approval of the BIM installation on the basis that it would render the main rotor spar a "fail-safe" structure, and thus eliminate the need for the present safe-life limitation of 1000 hours. Substantiation of the BIM installation was provided by Sikorsky Report SER-58331. In this report, probabilistic and statistical concepts were applied to the results of laboratory fatigue tests and flight stress surveys. Factors considered in the analysis included fatigue crack initiation, crack propagation, inspection interval, and reliability of the BIM. Sikorsky concluded that, on the basis of this analysis, installation of the BIM offered an improvement of 20 to 1 in reliability.

Sikorsky further noted that low occurrence fatigue fractures are caused by the random variability of many factors, and therefore contended that installation of the BIM is required to eliminate fractures which cannot be controlled without inspection. Samples of such factors include variability in the operating environment and variability in maintenance and overhaul procedures.

On the basis of the above report, Sikorsky concludes that main rotor blades equipped with BIM are fail-safe and can be considered serviceable until a crack is detected.

f. CAR Part 1.24(a), Service Experience Changes, states in part "when the Administrator finds as a result of service experience an unsafe condition exists ... the product shall not be operated until the unsafe condition has been corrected ... unless otherwise authorized by the Administrator under specified conditions and limitations, including inspections ...." The current main rotor blade retirement life of 1000 hours was imposed as a result of an unsafe condition. The cause of the fatigue failure has not been determined. The installation of BIM cannot correct the unsafe condition, but can establish the basis for permitting operation beyond 1000 hours by mandatory application of reliable inspection procedures.

g. CAR, Part 6.250(a), Main Rotor Structure, requires that "The service life of such parts (i.e., blades, blade attachments, etc.) shall be established by the applicant on the basis of fatigue tests or by other methods found acceptable to the Administrator." The requirement for the establishment of a service life for the main rotor blades is unequivocal. The Sikorsky proposal for, in effect, a fail-safe design and, more importantly, for unlimited service life, is incompatible with this portion of the requirement.

h. The portion of CAR, Part 6.250(a), that states "...by other methods found acceptable ..." could permit the use of BIM. An acceptable method must be one of unquestioned reliability. Since the December 23, 1960, approval of BIM, service experience with the method has been limited to several sets of blades being flown by one operator. We have been informally advised by Sikorsky that these sets of blades were handmade, and that difficulty has been encountered in sealing the blade during attempts to put the BIM design into production. The adequacy of BIM as a safe indication of blade failure can be evaluated following its use to a more extensive and widespread degree by operators. Until this use is acquired, complete acknowledgment that the method is acceptable from a reliability standpoint cannot be validated.

i. Among the BIM reliability substantiations required from the applicant must be included the approval of a process specification, in accordance with the provisions of CAR, Part 6.302, Fabrication Methods, the clear establishment and definition of inspection intervals and procedures, as required by CAR, Part 6.305, Inspection Provisions, and the demonstration of the gage installation reliability, necessitated by CAR, Part 6.601, Functional and Installational Requirements.

j. Sikorsky proposes that the rotor blade cuff life limitation be relieved and that the cuff also be retired "on condition" based on visual inspections. As previously indicated in CAR, Part 6.250(a), a service life must be established. Under this provision this proposal is not acceptable. Considering this proposal as a means to permit an extension of service life, on the basis of visual inspections only, would not be considered adequate. An inspection program similar in capability of crack detection to that of the BIM would be necessary.

4. CONCLUSIONS.

In consideration of the foregoing, it is concluded that, under the current provisions of CAR, Part 6.250(a):

a. approval of unlimited service life on the Model S-58 helicopter main rotor blades and main rotor blade cuffs, based upon, respectively, BIM or visual inspections, cannot be granted,

b. the present service life limitation on the Model S-58 helicopter main rotor blade of 1000 hours may be increased to a finite service life, whose magnitude is substantiated by fatigue tests, provided that the following are established and substantiated:

(1) reliable mandatory inspection intervals,

(2) the reliability and accuracy of the BIM gage under all operating conditions, and

(3) an approved process manufacturing specification.

c. the current retirement life on the Model S-58 helicopter main rotor blade cuffs shall be maintained.

REVIEW CASE NO. 8 REQUEST OF BEECH AIRCRAFT CORPORATION FOR REVIEW OF PROPOSED CE-210 PARTICIPATION IN CERTIFICATION OF THE BEECH MODEL H18 AIRCRAFT (Issued 17 July 1963)

1. INTRODUCTION.

The Beech Aircraft Corporation has requested,through the medium of personal representation and in writing, review and reconsideration of the Central Region's proposed extent of participation in the certification program for the Beech Model H18 aircraft.

2. CHRONOLOGICAL HISTORY.

a. By letter to the Central Region dated November 6, 1961, the Beech Aircraft Corporation initiated a certification program under the delegation option procedures of Part 410, Regulations of the Administrator, for a new model, H18, which will be the same as the Model G18S except for: revised engine installation; Hartzell 10152-5½ propeller blades; consolidation of four inboard metal fuel tanks to two conventional bladder cell tanks; larger wheels and tires on main landing gear; increase in gross weight from 9,700 pounds to 9,900 pounds basic, and from 9,800 pounds to 10,000 pounds with JATO.

b. By letter dated November 17, 1961, the Central Region expressed their intent to participate in the Model H18 certification program to the following extent:

(1) Airframe and Equipment Section

(a) Review the Basic Loads Report for this model.
(b) Review structural substantiation of the gross weight increase to 9,900 pounds (10,000 pounds with JATO).

(2) Propulsion Section

(a) Review portions of the Type Inspection Report, Parts I and II, pertaining to the powerplant installation.
(b) Review data or reports demonstrating compliance of the new bladder cells with the applicable portions of the Civil Air Regulations.
(c) Make a general inspection of the powerplant installation.

(3) Flight Test Section

(a) Review Part II of the Type Inspection Report for accuracy and for compliance with the Civil Air Regulations.

(b) Flight Test personnel will also check one or more flight items to determine accuracy of the data.

(4) Manufacturing and Inspection Section

(a) Verify that the applicant has conducted a complete conformity inspection of the product presented for type certification. Review applicant's Form ACA-317, Statement of Conformity.

(b) Conduct reinspections on one or more areas covered by the Type Inspection Report, Part I.

(c) Determine that equipment installed is in agreement with either the aircraft specification or the manufacturer's technical data equipment listing.

3. FACTS IN THE CASE

a. Under the delegation option procedures of Part 410, Regulations of the Administrator, the Federal Aviation Agency is required by Section 410.32(a)(2) to verify compliance with standards, rules, and regulations for unconventional designs and/or design features having a significant effect on safety, and to verify that there are no apparent unairworthy features. Under Section 410.32(b)(1), when the manufacturer makes major changes to a type design for which he holds a type certificate, the FAA will verify compliance as considered necessary.

b. For an aircraft of conventional design with which the manufacturer has experience, the minimum FAA participation will normally be the following:

(1) Airframe and Equipment Section

Spot check basic load report and witness at least one major structural test.

(2) Propulsion Section

Visually inspect the powerplant installation.

(3) Flight Test Section

Spot check the manufacturer's type inspection report (Part II) by conducting a flight inspection.

(4) Manufacturing and Inspection Section

Spot check the manufacturer's type inspection report (Part I) by conducting a ground inspection.

4. CONCLUSIONS.

a. Based upon these facts, it is concluded that the Beech Model H18 is a conventional aircraft having no unique features which would warrant detailed examination and review by the Central Region.

b. In consideration of the foregoing, it is determined that the Central Region letter of November 17, 1961, to Beech Aircraft Corporation established a verification program in excess of presently established procedures, and that only the minimum participation shown above should be deemed necessary by the FAA Central Region for all sections except the Propulsion Section. The Propulsion Section should participate to the extent originally proposed.

REVIEW CASE NO. 9. DOUGLAS PROPOSAL FOR DUAL AIRSPEED LIMITATIONS ON THE MODEL DC-8F; WE-210 MEMORANDUM DATED JANUARY 30, 1962 (Issued 17 July 1963)

1. ORIGIN.

The Douglas letter dated September 18, 1961, to the Western Region requiring the establishment of a dual airspeed limitation of the DC-8F.

This model is a multipurpose aircraft which will be operated as an all-passenger airplane, an all-cargo airplane, or a combination passenger and cargo airplane. Due to the higher density of cargo loading as compared to passenger loading and a desire to provide as much operational flexibility as possible, Douglas will certify higher zero fuel weights for use when a cargo load or a combination cargo-passenger load would cause the airplane weight to exceed the normal zero fuel weight. At the higher zero fuel weights, the airplane becomes gust critical in the high dynamic pressure region, "Q", and the maximum operating limit speed V_{NO} (V_{MO}) must be reduced. This has created a problem in the marking of the airspeed indicator and the setting of the overspeed warning sensor to provide for the two limiting speed ranges.

2. REGULATIONS AFFECTED.

a. CAR 4b.730 Markings and Placards, General

b. CAR 4b.732 Airspeed Limitation Information

c. CAR 4b.741(a)(2) Operating Limitations

3. HISTORY

The Douglas Aircraft Company letter dated September 18, 1961, to the Western Region requested concurrence with its proposal to install a dual airspeed limitation and overwarning sensor on the DC-8F in accordance with the proposed Special Civil Air Regulation published in the Federal Register dated June 8, 1961.

The Douglas proposal calls for the addition of a red radial line to the airspeed indicator and a modification of the sensor by one of the following:

a. Remove the passenger unit and install a cargo unit.

b. Install a dual unit with a selector switch.

The Douglas Aircraft Company proposal further calls for the addition of the following to the airplane flight manual limitations section:

> When operating in an all-passenger configuration, do not exceed the "barber pole". When operating in a partial or all-cargo configuration, do not exceed either the "barber pole" or the red line.

The Western Region Engineering and Manufacturing Branch letter dated December 1, 1961, to Douglas stated they do not concur with the proposed dual airspeed system for the following reasons:

a. Possible error or confusion on the part of the crew since they would be required to disregard the red line when flying the "barber pole".

b. With the present instrument panel and cockpit lighting system on DC-8 airplanes, the red line marking would not be visible under night lighting conditions. However, a dual overspeed warning sensor would be acceptable.

The Douglas Aircraft Company letter to the Western Region dated January 4, 1962, proposed to use an instrument with a two position "settable" red line in conjunction with a dual overspeed sensor with a selector switch.

WE-210 memorandum to FS-100 dated January 30, 1962, requested comments on the Douglas proposal and set forth the following with regard to the Douglas proposal dated January 4, 1962:

a. It does not appear feasible from an operational and safety standpoint to make the crew responsible for determining whether the red line or the "barber pole" should be observed. This could lead to confusion and error since the crew must also manually set the dual overspeed warning system dependent upon aircraft configuration.

b. The "resettable" red line may not meet the intent of policy established for the Convair Model 990 by FS-100 memorandum to the Western Region dated January 27, 1961. This policy required distinctive and unmistakable placards for dual airspeed limitations.

4. SUMMARY.

a. The current airworthiness requirements, CAR 4b, do not prohibit the establishment of a dual airspeed limitation.

b. The currently applicable airworthiness requirements, CAR 4b.732, states that airspeed limitations shall be presented in such a manner that they can be easily read and interpreted by the flight crew.

c. The currently applicable airworthiness requirement, CAR 4b.741, states that the normal operating speed, V_{NO}, shall be presented to the flight crew in accordance with section 4b.732 (b. above).

d. The currently applicable section CAR 4b.730, states that additional information, placards, and instrument markings having a direct and important bearing on safe operation of the airplane shall be required when unusual operating characteristics warrant.

e. The system proposed by Douglas does not present an easily interpreted airspeed indication in that the use of the "barber pole" or red line is dependent upon a particular configuration; wherein, the red line is limiting rather than the "barber pole" as is normally the situation.

f. The Douglas proposal does not provide for changing the maximum speed "barber pole" needle cam to provide a continuous indication of V_{NO} (or V_{MO}) at all altitudes. Such indication has been required for all turbojet aircraft including the standard DC-8 series.

g. The dual setting of the overspeed warning sensor, as proposed by Douglas, is satisfactory, provided adequate procedures and instructions are developed to preclude improper setting. The procedures should at least include an AFM limitation and a check item on the cockpit checklist. Instructions must be provided for airline operations personnel which will require the appropriate setting for each zero fuel weight.

h. When the zero fuel weight exceeds 187,000 pounds, the normal operating limit speed is reduced a maximum of 22 knots in the altitude range of 10,000 to 27,000 feet due to structural considerations.

5. CONCLUSIONS.

In consideration of the above, the Douglas proposal for establishing dual airspeed limitations for the Model DC-8F is found unacceptable since it does not comply with CAR 4b.730 and 4b.732 for the following reasons:

a. There is no provision for maximum speed "barber pole" needle indication of V_{NO} or V_{MO} when the lower airspeed limits are applicable. It would be confusing to the pilots to observe the "barber pole" for one condition and not the other.

b. The dual red line marking of the airspeed indicator could be too easily misset or tampered with after setting, thereby giving the pilots erroneous limitations information.

c. The red lines are not acceptable because they would lose their significance and be confusing to the pilots.

d. Adequate instructions, procedures, and limitations for the setting of the dual overspeed warning sensor are not provided.

REVIEW CASE NO. 10. RULING ON APPLICABILITY OF AIRFLOW PROVISIONS IN CIVIL AIR REGULATIONS 7.382(a), CARGO AND BAGGAGE COMPARTMENTS, TO VERTOL MODEL 107-II ALL CARGO HELICOPTER (Issued 17 July 1963)

1. INTRODUCTION.

Vertol Division of The Boeing Company has proposed to the Eastern Region, EA-212, for the Vertol Model 107-II all cargo helicopter, that the provisions of Cargo Compartment Classification, Class E, CAR 4b.383(e)(3), "Means shall be provided to shut off the ventilating airflow to or within the compartment. Controls for such means shall be accessible to the flight crew in the crew compartment," and CAR 4b.383(e)(5), "Required crew emergency exits shall remain accessible under all cargo loading conditions," be used in lieu of the airflow provisions specified in CAR 7.382(a), "Design of inaccessible compartments and sealing of these compartments shall be such as to contain cargo compartment fires for a period of time sufficient to permit landing and safe evacuation of the occupants," on the premise that the airflow provisions of CAR 7.382(a) are inapplicable to an all cargo version helicopter. The Eastern Region concurs with the request and has asked for Washington Office concurrence and/or comments.

2. CHRONOLOGICAL HISTORY.

a. Eastern Region, EA-212, teletype message of February 16, 1962, to Washington Office, FS-120, outlining problem and requesting concurrence and/or comment with Eastern Region recommendations.

b. Washington Office, FS-120, teletype message of February 27, 1962, indicating answer forthcoming in an Engineering and Manufacturing Division Review Case.

3. FACTS IN THE CASE.

a. CAR 7.382 does not contain provisions similar to those in CAR 4b.383(e)(3) and (5) directly applicable to an all cargo helicopter.

b. At the time of inception of CAR 7, Rotorcraft Airworthiness, Transport Categories, the use of all cargo type helicopters was not envisioned. The related requirements were administered primarily in the type certification of passenger-carrying helicopters.

c. The shortcoming in CAR 4b was recognized with issuance of Amendment 4b-10, issued April 17, 1959, which established a new Class E cargo compartment applicable to fixed-wing transport aircraft used for the carriage of cargo only. The basis for issuance of Amendment 4b-10 is contained in the preface thereto and is considered equally valid for a transport helicopter.

d. In the absence of a requirement in CAR 7 directly applicable to the Vertol request, the provisions of CAR 7.10, Eligibility for Type Certificate, may be invoked by the Administrator.

4. CONCLUSIONS.

In consideration of the foregoing, it is concluded under the provisions of CAR 7.10 that the Vertol Model 107-II all cargo helicopter shall be eligible for type certification providing:

a. Compliance is shown with the provisions of CAR 4b.383(e)(3) in lieu of the airflow provisions contained in CAR 7.382(a).

b. Compliance is shown with the provisions of CAR 4b.383(e)(5) relative to accessibility of crew emergency exits.

c. Compliance is shown with the provisions of CAR 4b.380(c), Protective Breathing Equipment, "If the airplane contains Class A, B, or E cargo compartments, protective breathing equipment shall be installed for the use of appropriate crew members," in lieu of CAR 7.382(c), "If compartments are intended to be accessible in flight, protective breathing equipment shall be available for the use of the appropriate crew member."

d. Compliance is shown with the provisions of CAR 4b.382(d), "Sources of heat within the compartment shall be shielded and insulated to prevent igniting the cargo."

e. Compliance is shown with the provisions of CAR 7.382 except those relative to airflow in CAR 7.382(a) and protective breathing equipment in CAR 7.382(c).

REVIEW CASE NO. 11. CESSNA AIRCRAFT COMPANY REQUEST FOR APPROVAL OF THE EMERGENCY EXIT PROVISIONS ON THEIR SIX-PLACE VERSIONS OF AIRCRAFT MODELS 310E THROUGH 310H, 320, AND 320A (17 July 1963)

1. INTRODUCTION.

The Cessna Aircraft Company has requested the Central Region to approve an increase in the maximum occupancy of their Models 310E through 310H, 320, and 320A from 5 to 6 persons. Cessna contends under CAR 3.10, Eligibility for Type Certificate, that the undersize emergency exit provided on the left side is just as effective as the exit type prescribed in CAR 3.387, Exits. Cessna also proposes the addition of a second emergency exit means on the right side. The Central Region recommends approval of the 6 place versions proposed by Cessna but has requested concurrence from the Washington Office before advising the applicant, particularly as the equivalent level of safety provisions of CAR 3.10 are involved.

2. CHRONOLOGICAL HISTORY.

a. In a letter dated February 10, 1962, to the Central Region, Mr. W. H. Prewitt, Chief Administrative Engineer, Cessna Aircraft Company, acting in the capacity of DMCR 3-3, requested concurrence of his action as follows:

> "Beginning with the 1963 Models 320A and 310H, Cessna intends to offer six-place versions of these aircraft.
>
> "CAR 3.387 requires that an emergency exit opening be provided, the minimum dimensions of which shall be such that a 19 by 26 inch ellipse may be completely inscribed therein.
>
> "Although these openings do not meet the exact requirement specified, they are of adequate shape and area to serve the purpose intended. A copy of Report 1547 is enclosed. This report, originally prepared for purposes of exporting Model 310 aircraft to Canada, documents that compliance with the intent of the regulations has been demonstrated.
>
> "Therefore, I am approving the emergency exit openings for the above models when used as six-place aircraft in that an equivalent level of safety has been provided and demonstrated. It is requested that your office indicate by return letter your concurrence with this approval."

b. The Central Region reply dated February 24, 1962, stated:

> "Our evaluation of the data indicates that the exits are inadequate for approval inasmuch as the height of the exits would be shy by 3.5 and 5.5 inches from that necessary to permit a 19 by 26 inch ellipse to be inscribed therein. The photographs, although indicating a man might escape through the exit, are not considered justification for deviating from CAR 3.387. In the past, demonstrations have been used to verify numbers of persons utilizing the exits, but not to justify reductions of sizes of the amount indicated in this case."

c. The matter was presented to the Washington Office for resolution by Mr. W. H. Prewitt, DMCR 3-3, in a letter dated March 5, 1962, to the Director, Flight Standards Service. His letter stated in part:

> "It is Cessna's contention that the present emergency exit provided an equivalent level of safety and that it has been so demonstrated by tests and service experience. Further, an undue hardship would be placed upon the manufacturer in order to meet the exact dimensional requirements specified by CAR 3.387.
>
> "In summary, I believe the emergency exit provisions should be considered to meet the intent of the regulations for the following reasons:
>
> (a) No significant increase in safety will result by enlarging the opening.
>
> (b) At the time the basic airframe structure was certificated, actual demonstrations were being accepted in lieu of meeting exact dimensional requirements.
>
> (c) The utilization of the Models 310 and 320 as a six-place aircraft is a normal development that should not require extensive change nor place an undue hardship upon the manufacturer.
>
> (d) A generous sized baggage door can also be used as an emergency exit by the passenger in the sixth seat."

d. In a meeting held on March 13, 1962, between Mr. W. H. Prewitt, DMCR 3-3, and representatives of the Airframe Branch, FS-120, and Mr. W. Anderson, Central Region, Mr. Prewitt was advised that approval could not be granted unless, under CAR 3.10, it was shown that the emergency exit means provided is just as effective as the one prescribed in CAR 3.387. Conduct of evacuation tests and use of the baggage door as an emergency exit were suggested.

e. In a letter dated March 19, 1962, to the Central Region, Mr. W. H. Prewitt stated:

> "As previously pointed out in Cessna letters 178-2-30 and 178-3-9, the company intends to offer six-place versions of the Models 310E through H, 320, and 320A. The standard emergency exit required for six-place aircraft is an opening such that a 19 by 26 inch ellipse may be completely inscribed therein. The present Model 310's and 320's have an emergency exit which is smaller than that noted above. However, they have a baggage door which the passengers can use for emergency exit. Both of these exits are of adequate size and shape to permit rapid evacuation of the aircraft in case of emergency. In fact, the time required is greatly reduced below that which would be required if only one standard size opening were provided.
>
> "Enclosed is Report 310G-6212-021, which shows the emergency exits that have been provided for the Models 310E through 310H, 320 and 320A. It verifies that the level of safety is greater than that required by CAR 3.387."

f. In a memorandum dated March 20, 1962, to the Washington Office, the Central Region recommended approval of the Models 310E through H, 320 and 320A aircraft for six passengers on an equivalent safety basis in lieu of literal compliance with the emergency exit size requirement of CAR 3.387.

Recommended approval was contingent on demonstrations conducted using the baggage door as an emergency exit to verify that egress through the window exit; also demonstrations that egress through the window exit on the Cessna 310E through 310H was no more difficult than through the 19 by 26 inch elliptical opening even though the window height was considerably less than 19 inches.

g. Washington requested Central Region on March 29, 1962, to provide comparative evacuation test times, using six persons. Three evacuation tests were asked for - baggage door, current exit, and a 19 by 26 ellipse.

h. The Central Region's priority wire dated March 30, 1962, to Washington stated:

"Re phone this date. Cessna evacuation test conducted four men and two women. Total evacuation time 310G exit 33 seconds, 19 x 26 ellipse 29.6 seconds, baggage door 20.1 seconds. Tests applicable to 310E and up. Reference copy Cessna February 16, 1962, letter. Cessna not asking approval for Models 310 through 310D. Detailed report to follow."

3. FACTS IN THE CASE.

a. The Cessna Aircraft Company Models 310 and 320 series were developed as five-place aircraft and type certificated under CAR 3, dated November 1949, including Amendments 3-1, through 3-10. The main cabin entrance on these aircraft is on the right side with the emergency exit on the left side. Under the provisions of CAR 3.387, Exits, an emergency exit is not required on aircraft approved to carry five persons or less.

b. Cessna Aircraft Company now intends to offer optional six-place versions of Models 310E through 310H, 320, and 320A. The baggage area is replaced by the sixth seat and the baggage door is used as an additional emergency exit.

c. Cessna has conducted evacuation tests to establish under the provisions of CAR 3.10 that the emergency exit means is just as effective as prescribed in CAR 3.387. The exit configurations tested are shown below:

	h_1	h_2	W	Area, Square Inches
Model 310E, F, G, H	15	12	26	373
Model 320, 320A	16	15	26	429
Ellipse (19 x 26)				388
Baggage Door	22½	25	21¼	494

Six occupants were all seated with safety belts fastened.

Sex	Height	Weight	Age
M	5' 8"	231 lbs.	38
M	6' 1"	210 lbs.	39
M	5' 9½"	185 lbs.	33
M	5' 9½"	150 lbs.	28
Fm	5' 3½"	115 lbs.	40
Fm	5' 7"	118 lbs.	45

The following tests were conducted:

Test No. 1 - Time to unfasten seat belt and kick out exit door - 6 seconds

Test No. 2 - Evacuation through Cessna 310G exit - 33 seconds

Test No. 3 - Time to evacuate thru 19 x 26 ellipse - 29.6 seconds

Test No. 4 - Time to evacuate thru baggage door - 20.1 seconds

Test No. 5 - Rerun of Test No. 2 - 23.1 seconds

Test No. 6 - Rerun of Test No. 3 - 21.1 seconds

Analysis of the foregoing indicates the area of the exit on the Cessna 310E through H is 96 percent of the area of a 19 x 26 ellipse, or only 4 percent less than required. For the Models 320 and 320A there is more actual area than a 19 x 26 inch ellipse provides, thus indicating the evacuation time through the Models 320 and 320A would probably be less than for the ellipse. The baggage door also has considerably more actual exit area on both the Cessna 310 and 320 series than a 19 x 26 inch ellipse provides.

Analysis of the evacuation times indicates very little significant time differences between the Cessna 310E through H series and a 19 x 26 inch ellipse. Likewise, the total time is considerably less than might have been expected in a test of this kind. These factors alone would support acceptance of the left hand exit as being equivalent to that prescribed in CAR 3.387.

In addition, there is the extra safety feature present of an additional exit on the right side. Admittedly it does not support the claim of equivalent safety for the left side of the aircraft under all emergency situations, but it is considered a highly desirable safety provision. The evacuation time through it was only 20 seconds, hence, in situations where both the left and right side exits are useable, the total aircraft evacuation time would be greatly reduced.

It should also be noted that under the present requirements of CAR 3.387 for a five-place aircraft no emergency exit is needed. For a passenger capacity of 6 - 15 one emergency exit is needed.

In the case at hand the applicant has only added one person, i.e., he has gone from a five to a six-place version aircraft. Under the literal provisions he must have an emergency exit. He has, but it is not in literal conformance with the dimensions prescribed. The evacuation tests show the effectiveness of the exit he provided is almost comparable to an elliptical exit. Furthermore, the applicant has shown he is providing an additional exit on the opposite side that is superior to the elliptical shaped exit as an evacuation means. This exit is being added in part to account for the transition from a five to a six-place version and must certainly be considered as an additional highly desirable safety provision in the overall appraisal of the problem.

4. CONCLUSIONS.

In consideration of the foregoing it is concluded under the provisions of CAR 3.10 that the Cessna Aircraft Company has satisfactorily shown that the emergency exit means provided on their Models 310E through 310H, 320, and 320A six-place versions are as effective as the means prescribed in CAR 3.387. This finding is supported by actual evacuation tests wherein six persons successfully evacuated through the left exit now installed on the Models 310E through H aircraft, and through a 19 x 26 inch ellipse shaped exit, in total elapsed times of 33 seconds and 29.6 seconds, respectively; also by the fact that the area of the emergency exit in the Models 320 and 320A is greater than that for a 19 x 26 ellipse, and therefore that the evacuation means provided on the Models 320 and 320A would be at least as effective as prescribed in CAR 3.387; also by the fact that an additional exit is provided on the opposite side which has been demonstrated to be more effective than either the presently provided exits or an elliptical shaped exit for a total of 6 persons.

Subsequent six-place installations may therefore be approved providing:

a. The baggage door on the right side is placarded "Emergency Exit - Force to Open."

b. The emergency exits on both sides comply with the applicable provisions of CAR 3.387 except for minimum dimensions prescribed in CAR 3.387(a).

REVIEW CASE NO. 12. VERTOL MODEL 107-II WATER CERTIFICATION AND EMERGENCY EVACUATION (Issued 17 July 1963)

1. INTRODUCTION.

The Vertol Division of The Boeing Company was issued Type Certificate Number 1H16 for their Model 107-II helicopter, a twenty-five passenger twin turbine, transport category helicopter, which has two Type II and two Type IV emergency exits. When landed on the water, with no lift supplied by the rotors, the thresholds of the two forward Type II exits are below the waterline.

Vertol has applied to the Eastern Region for approval of a twelve-passenger/cargo version of the Model 107-II helicopter in which these Type II exits are the only available means of passenger escape, because the bulkhead separating the passenger and cargo compartments is in its most forward location and is forward of the Type IV exits. Vertol contends that the lift supplied by one engine will raise the thresholds of the Type II exits above water level, making those exits available for passengers. Vertol contends further that the failure of both engines to operate would constitute a double emergency and need not be considered.

The Eastern Region believes that an exit on each side of the helicopter should be available after an emergency water landing with no rotor lift provided and asks for our concurrence.

2. CHRONOLOGICAL HISTORY.

a. On January 26, 1962, the Vertol Model 107-II helicopter was type certificated as a Category B rotorcraft under Civil Air Regulations Part 7. Certification was for twenty-five passengers in accordance with CAR 7.357, Emergency Evacuation. Consequently, it has one Type II and one Type IV exit per side. The Type II exits are located forward with the right-hand one serving as the main entrance door. The rotorcraft was considered acceptable for emergency water landings; and although its buoyancy provisions were accepted on the basis that one engine would be operating and supplying rotor lift, the forward portion of the fuselage is a watertight compartment and the helicopter will float indefinitely without rotor lift in spite of the fact that the two forward Type II exits are below the waterline. Under those conditions the two aft Type IV exits are available and adequate for evacuation of the occupants in the twenty-five passenger version.

b. In their memorandum of September 27, 1961, the Eastern Region supplied the following information to the Airframe Branch, FS-120:

> With one engine operating, all exits are above water level.

With both engines inoperative, the thresholds of the forward doors are below water level, but Vertol is providing a Dutch door type arrangement for the left side front exit. An exit, with the approximate dimensions of 19 x 35 inches and its lower frame above the waterline, is available when the lower portion of this door is latched in place. Vertol contended that the no-power condition need not be considered since this assumes a double failure. The Eastern Region, however, stated that they consider the emergency evacuation requirements of CAR 7.357 applicable with the helicopter on water without power and requested a Washington ruling on this.

c. The Washington reply dated October 12, 1961, concurred that no-power water emergency evacuation should be considered, but pointed out that to expect every exit to be above water level was being far more severe than the requirements of Section 4b.362(d), Ditching Emergency Exits, which merely specifies two Type III exits above water level, one on each side, for as many as seventy passengers. This memorandum pointed out that the two Type IV exits in the Vertol would remain above water level without power and in the absence of specific ditching exit requirements in Part 7, this could be accepted as satisfactory for the twenty-five passengers.

d. The Flight Standards Service on November 30, 1961, issued "Interim Criteria for Operation of Multiturbine Helicopters Under CAR 46 - Initial Six Months Period." This document approves the use of water sites as emergency landing areas provided, among other things, that the helicopter has been tested for stability on the water with the rotors turning and also stopped. If stability must be demonstrated on water without power, it follows that this also must be an anticipated condition for emergency water evacuation.

3. <u>FACTS IN THE CASE</u>.

a. The twelve-passenger version of the Vertol 107-II complies with CAR 7.357(c) for emergency landing on land in that a Type II exit is available in each side and a Type III exit per side is required.

b. In a water landing with one engine inoperative and rotor lift supplied by the operating engine, the thresholds of both Type II exits are above water level and are available for emergency evacuation.

c. On the water with both engines inoperative, the thresholds of both Type II exits are below water level. The left side exit has a Dutch door type arrangement such that the lower part is kept closed and a 19 x 35 inch opening is provided above water level.

d. Part 7 of the Civil Air Regulations contains no reference to ditching exits. It is only reasonable to expect, however, that a helicopter certificated with flotation capabilities should have at least one useable exit on each side.

e. The Flight Standards Service Memorandum of November 30, 1961, previously mentioned, requires that stability on water shall be demonstrated both with and without rotor lift. This indicates that the Flight Standards Service believes that the no rotor lift condition during a water landing should be considered.

4. CONCLUSIONS.

In consideration of the foregoing, it is concluded under the provisions of Part 7 of the Civil Air Regulations, Section 7.10, Eligibility for Type Certificates, that the twelve-passenger/cargo version of the Vertol Model 107-II Helicopter shall not be eligible for type certification unless adequate exits are installed on each side of the fuselage above the water level when the helicopter is on the water with all power off.

REVIEW CASE NO. 13. TAIL LIGHT (REAR POSITION LIGHT) INSTALLATION ON BOEING 727 AIRCRAFT (Issued 17 July 1963)

1. INTRODUCTION.

The Boeing Company has requested the Western Region to approve the installation of two rear position white lights mounted on the wing tips of the Boeing 727. Boeing contends this installation meets the intent of CAR 4b.632(c), Rear Position Light, which prescribes that the rear position light shall consist of a white light mounted on the airplane as far aft as practicable. Boeing also contends this location is preferred to a conventional single light location on the vertical stabilizer as the wing tips are more accessible for routine maintenance. The Western Region believes that the proposed wing tip installation complies with the intent of CAR 4b.632(c), and requests Washington Office concurrence.

2. CHRONOLOGICAL HISTORY.

a. The present CAR 4b.632(c) wording adopted February 25, 1957, states, "The rear position light shall consist of a white light mounted on the airplane as far aft as practicable." Based on the numerous comments submitted by interested parties at the time this regulation change was originally proposed, there was universal acceptance that the phrase "as far aft as practicable" applied to the empennage area.

b. The Western Region's memorandum dated January 30, 1962, has indicated that in their opinion the proposed installation complies with the intent of CAR 4b.632(c) for a rear position light. No supporting evidence for this opinion was submitted.

c. In a telephone conversation on February 13, 1962, Boeing advised the Western Region their proposed installation is now in the final design stages. It was indicated the white rear position lights would be housed in the same fixture on the wing tips of the 727 as the forward position lights; also that the wing tip positions are to be separated 110 feet laterally and located 25 feet forward of the most rearward point on the airplane.

3. FACTS IN THE CASE.

a. Based on the comments received at the time CAR 4b.632(c) was adopted there is little doubt that the intended location for a rear position light is the empennage area.

b. The argument that the light cannot be installed in the empennage area because of excessive maintenance problems is fallacious.

Admittedly, the design of an adequate shock mounting is complicated because of vibrating conditions associated with an aft engine pod installation such as on the 727, but this still does not preclude designing an adequate installation, or improving the light to withstand the vibration encountered in this area. It is also common practice for air carriers to conduct fairly frequent visual inspections of the airframe and related systems, at which time the light installation would also be checked.

c. There is no precedent for deviating from the empennage area location for the rear position light. All transport aircraft type certificated under the provisions of CAR 4b.632(c) comply therewith.

d. To our knowledge the problem is not an insurmountable one for foreign aircraft designers as evidenced by installation of the rear position light in the tail cone on comparable aircraft models such as the de Havilland DH-121, Vickers V-10, and the British Aircraft Corporation BAC 111.

e. The present wording of CAR 4b.632(c) is such that it is unreasonable to interpret the two lights instead of one, each 110 feet apart, and each 25 feet forward of the most rearward point on the airplane, meet the single light criteria "as far aft as practicable."

f. The possibility arises during operation of the aircraft that the presence of two white lights 110 feet apart could be misinterpreted in denoting the presence of two airplanes when observed from the rear. This negates any implication of an equivalent level of safety under the terms of CAR 4b.10.

4. CONCLUSION.

In consideration of the foregoing, it is concluded that under the provisions of CAR 4b.632(c), approval of the wing tip installations for rear position lights be denied. Denial is based on the fact that the proposed installation is incompatible with the intent of CAR 4b.632(c), which requires such an installation to be located in the empennage area. There is no supporting evidence submitted by the applicant to justify a deviation from the airworthiness standard.

REVIEW CASE NO. 14. CESSNA AIRCRAFT COMPANY ELECTRIC CLOCK INSTALLATION (Issued 17 July 1963)

1. INTRODUCTION.

The Cessna Aircraft Company, in a letter to the Central Region dated September 14, 1959, requested a deviation from CAR 3.688, "Master Switch Arrangement." Cessna proposed the installation of an electric clock which would be wired directly to the battery, rather than wiring the clock circuit to the master switch. The Central Region informed Cessna that their proposal would meet the intent of CAR 3.688, and that a deviation was not required. The Engineering and Manufacturing Division was later advised of this installation and subsequently questioned the Central Region's decision as being contrary to the intent of CAR 3.688. The region is of the opinion that the Cessna circuit complies with the regulation because the clock circuit bypasses the battery solenoid and, therefore, is not considered to be a part of the "main distribution system." The region also feels that the circuit does not significantly compromise the functions of the master switch. The matter has again arisen as Cessna desires to furnish a similar clock installation in the forthcoming new Model 336 airplane.

2. CHRONOLOGICAL HISTORY.

a. Cessna Aircraft Company letter dated September 14, 1959, to the Central Region, requesting a deviation to permit the installation of an electric clock which, in the opinion of Cessna, would not completely comply with CAR 3.688.

b. Central Region letter to Cessna dated October 7, 1959, informing Cessna that a deviation was not required since the proposed circuit was considered to comply with the intent of the regulation.

c. FS-120 memorandum to the Central Region dated February 3, 1960, advising that the installation was contrary to the provisions of CAR 3.688.

d. FS-120 memorandum to the Central Region dated January 18, 1961, informing the region that, (a) Cessna Service Letter No. 210-21 dated December 30, 1960, states that an electrically wound clock is wired directly to the aircraft battery, and (b) that this type of arrangement would not be in accordance with CAR 3.688.

e. Central Region memorandum to FS-1 dated January 24, 1961, giving the basis for the region's approval of the Cessna circuit. The actions which are referenced as support for the region's approval are, with one exception, dated prior to May 24, 1956. CAM 3.688-1 entitled, "Load Circuit Connections with Respect to Master Switch" and CAM 3.688-2 entitled, "Electric Stall Warning Indicator Circuit" became effective on May 25, 1956.

f. FS-120 memorandum to the Central Region dated March 6, 1961, informing the Region that in view of the present wording of CAM 3.688-1 and CAM 3.688-2, it is apparent that the Cessna electric clock installation is not in accordance with the provisions of CAR 3.688.

g. Central Region letter to Cessna dated May 22, 1961, informing Cessna that the electric clock circuit is not considered to comply with the intent of CAR 3.688. Cessna was asked to consider appropriate redesign.

h. Cessna letter to the Central Region dated June 1, 1961, advising that service experience and overall operation of the electric clock installation were satisfactory and that no installation changes were planned.

i. Central Region memorandum to FS-100 dated September 21, 1961, indicating that no further regional action was planned.

j. FS-120 December 12, 1961, message to the Central Region informing the region that the electric clock installation was considered an open item.

k. Central Region memorandum dated February 1, 1962, to FS-100 in which the region states that in the opinion of their legal personnel, the Cessna electric clock circuit does not appear to comply with the intent of the regulation. The region's comment that the clock might be considered emergency equipment for IFR conditions has some validity. The effectiveness of the clock, however, would not be compromised if a suitable master switch arrangement were employed. One example of an acceptable arrangement is shown in the region's letter to Cessna dated May 22, 1961. In this arrangement, the region suggests the installation of a second switch which would make it possible to disconnect power from the clock circuit. Several different satisfactory master switch arrangements are possible.

l. Central Region February 21, 1962, message requesting acceptability of an electric clock circuit in the Cessna 336, a new model airplane. The proposed circuit would be similar to the electric clock wiring which bypassed the master switch on previous models.

m. FS-120 February 26, 1962, message to the Central Region, informing the region that an Engineering and Manufacturing Division Review Case regarding this matter would be initiated.

3. FACTS IN THE CASE.

a. Civil Air Regulation 3.688 requires that when electrical equipment is installed, a master switch arrangement shall be provided which will disconnect all sources of electrical power from the main distribution system at a point adjacent to the power sources. The policy material contained in CAM 3.688-1 and CAM 3.688-2 states that all load circuits should be connected to electric power sources in such a manner that the master switch can interrupt service, unless such interruption of service would result in the inability to maintain controlled flight or to effect a safe landing. The policy also states that electrical stall warning indicator circuits, when installed, should be connected to the electric power system in such a manner that the master switch can interrupt service. (Prior to the adoption of CAM 3.688-1 and CAM 3.688-2, it was permitted to install stall warning circuits directly to the battery.)

b. Cessna originally requested a deviation to CAR 3.688 to permit the present installation of their electric clock. The region did not consider a deviation necessary and informed Cessna that the installation was satisfactory.

c. No evidence has been submitted that compliance has been substantiated by terms of CAR 3.10, equivalent level of safety.

4. CONCLUSION.

In consideration of the foregoing it is concluded, under the provisions of CAR 3.688, that the Cessna Aircraft Company should be directed to wire the electric clock through a master switch or master switch arrangement in the Model 336 aircraft. It is also concluded, under the provisions of CAR 3.688, that a design change should be incorporated in the Cessna airplanes which are presently in production so as to provide the above mentioned master switch arrangement.

In regard to service installations, it is concluded that these should be examined from the standpoint of whether mandatory corrective action is required, considering both the degree of unairworthiness existing and the provisions of CAR 1.24(a), "Service Experience Change". To implement this, the Central Region should determine the need for corrective action by making a finding by inspection of new production aircraft for the length, gauge, insulation, and adequacy of the mechanical supports used for the unprotected wire between the battery and the fuse, determine the size of the fuse, and obtain this same information for the wire between the fuse and the clock. Repeat this inspection on a sample basis on service aircraft to determine the effects of deterioration, chaffing, etc., if any. Based on these findings together with a review of the manufacturer's service records, the Central Region should determine the need for corrective action to the service aircraft and advise FS-100 of the conclusion reached.

REVIEW CASE NO. 15. BOEING 707-300B SERIES AIRPLANES - PROPOSED OPERATION WITH ANTISKID INOPERATIVE TOGETHER WITH REVERSE THRUST PERFORMANCE CREDIT (Issued 17 July 1963)

1. INTRODUCTION. The question has been repeatedly raised (Pan American Airlines, Convair, Lockheed and others) of granting some degree of reverse thrust performance credit under the provision of SR-422B, Section 4T.115(b) and 4T.122(f), in opposition to the feeling of ALPA that actual operations have shown that landing distance limitations are unrealistic. The Boeing request is a variation involving performance credit only for an inoperative antiskid system.

2. APPLICABLE REGULATIONS (SR-422B).

   a. Section 4T.115(b) (Accelerate-stop distance). This section states, "In addition to, or in lieu of, wheel brakes, the use of other braking means shall be acceptable in determining the accelerate-stop distance, provided that such braking means shall have been proven to be safe and reliable, that the manner of their employment is such that consistent results can be expected in service, and that exceptional skill is not required to control the airplane."

   b. Section 4T.122(f) (Landing distance). This section states, "In addition to, or in lieu of, wheel brakes, the use of other braking means shall be acceptable in determining the landing distance, provided such braking means shall have been proven to be safe and reliable, that the manner of their employment is such that consistent results can be expected in service, and that exceptional skill is not required to control the airplane."

3. CHRONOLOGICAL HISTORY.

   a. Reverse thrust performance credit for landing distances for Lockheed 1329:

   FS-100 memorandum of April 6, 1960, to FS-2000 (Southwest Region).

   This memorandum sets forth the initial policy material for reverse thrust performance credit for landing distances and specifically applies to the Lockheed 1329. This policy material allows for 50 percent performance credit for reverse thrust in comparison to the landing distances without reverse thrust based on the two most critical symmetrical engines being operated in the reverse thrust position with power not exceeding the maximum continuous rating. The policy also allows consideration of three engines in reverse thrust position with power not to exceed the maximum continuous rating for performance credit providing controllability is not a problem.

b. Reverse thrust performance credit for accelerate-stop distance for Lockheed 1329:

FS-100 memorandum of May 5, 1960, to FS-2000 (Southwest Region).

This memorandum sets forth the initial policy material for reverse thrust performance credit for accelerate-stop distances and specifically applies to the Lockheed 1329. This policy material is essentially the same as that policy stated in item a. above for reverse thrust performance credit for landing distance.

c. Reverse thrust performance credit for Convair 880:

Telephone conversation of August 8, 1960, between FS-160 and FS-4160 (Western Region).

FS-4160 was informed that the policy for reverse thrust performance credit issued for the Lockheed 1329 would be applicable for use in handling the Convair 880 request for reverse thrust performance credit.

d. Proposed CAM for reverse thrust performance credit:

Engineering and Manufacturing Circular Memorandum No. 60-22 of December 27, 1960.

This memorandum requests comments on a proposed CAM reverse thrust policy. This proposed policy permitted up to 50 percent distance credit similar to that previously given for Lockheed and Convair.

e. Proposed policy for reverse thrust performance credit:

Conference of April 12, 1961.

A conference was held with ALPA, ATA, and FAA to discuss proposed policy for reverse thrust performance credit. The proposed policy for reverse thrust performance credit basically is the same policy set forth in items a. and b. above. The conference results were summarized as follows:

(1) ALPA is strongly opposed to the publication of any policy for reverse thrust performance credit under the current regulations. ALPA favors reverse thrust performance credit only if the current regulations are revised to require a rational landing distance determination during type certification.

(2) ATA is neutral toward the policy since it does not visualize the recertification of current operating equipment under this policy.

(3) AIA, although not represented at the meeting, favors reverse thrust performance credit but recommends certain provisions to the proposal; namely, the exclusion of the three-degree glideslope and full performance credit for reverse idle thrust rather than 50 percent performance credit based on reverse maximum continuous thrust.

f. Status of reverse thrust performance credit policy:

FS-100 letters of May 10, 1961, to ALPA and to ATA.

These letters state that the FAA will not publish the proposed policy for reverse thrust performance credit but will judge any application for reverse thrust performance credit on its own merits.

g. Status of reverse thrust performance credit policy:

FS-1 letter of August 9, 1961, to ALPA.

This letter emphasizes that SR-422B, Sections 4T.115(b) and 4T.122(f), specifically permit reverse thrust performance credit and reiterates FAA's thinking as expressed in item f. above. It also states that the Safety Regulations Division is making a study which may result in regulatory action on this problem.

h. Proposed SR-422C.

FS-40 Notice of Conference of May 4, 1962.

This notice proposed SR-422C which, among other proposed revisions to SR-422B, included a rationalized basis for determining realistic landing/accelerate-stop distances. Provisions were included to account for different surface conditions and aerodynamic braking means. Appendix E of this notice contains a list of air carrier landing incidents and accidents which include (1) airplanes veered off runway and (2) overshoots.

i. Dispatch for operations with inoperative thrust reversers:

WE-210 memorandum of May 8, 1962, to FS-100.

This memorandum contains Douglas' proposal for dispatch of DC-8-50 Series airplanes with all thrust reversers inoperative. This memorandum included a statement that the Convair Models 22, 22M, and 30 have been approved on a dispatch deviation basis for operation with inoperative thrust reversers by the Regional Air Carrier Operations Branch.

j. Evaluation of proposed SR-422C:

FS-1 memorandum of June 22, 1962, to FS-40, -100, -10, -400, PT-900, and WE-1.

This memorandum established a working group to evaluate the proposed requirements of SR-422C. A flight test program will be conducted on a jet transport airplane to determine the effect of these rules.

k. Proposed SR-422C.

FS-40 withdrawal of Notice of Conference of July 5, 1962.

This notice advised that the conference was being cancelled, that FAA was arranging for a flight test program to determine the extent of testing required by the proposed SR-422C, and that it was planning to hold individual meetings within industry to review and discuss the impact of rule changes on the operation of today's transports. Comments were requested from interested parties.

l. Boeing's proposal for reverse thrust performance credit:

Boeing's letters of July 16 and August 7, 1962, to Western Region.

These letters request reverse thrust performance credit for landing distances with the antiskid system inoperative. Boeing's proposal basically asked for performance credit for the use of reverse thrust based on the operation of the two most critical symmetrical engines in reverse operation at a conservative reverse thrust EPR setting.

m. Western Region's appraisal of Boeing's proposal for reverse thrust performance credit.

WE-216 memorandums of July 19 and August 7, 1962, to FS-160.

The first memorandum transmitted Boeing's proposal of July 16, 1962 to FS-160. The second memorandum transmitted a draft of a proposed reply to Boeing, to Washington for study. This draft concluded that Boeing's proposal is essentially satisfactory.

n. Photographic survey of operational jet transport landings:

Flight Standards Service Release No. 470 (presently being printed).

This Service release is a statistical presentation of operational landing parameters for jet transport airplanes. It shows that there is little correlation between typical airline operation and the operation used to demonstrate landing distance during type certification. The study shows that jet transport airplanes usually cross the threshold approximately 30 feet lower and with a speed of 9 knots faster than the corresponding value used during type certification tests. Landing distances in these operations are approximately 1,000 to 1,500 feet greater than field lengths shown in the airplane flight manual.

o. Reverse thrust summary:

FS-1 meeting on August 22, 1962, with FS-40, -100, and -400.

The merits of the above Douglas and Boeing requests involving reverse thrust were discussed. Points brought out include:

(1) Reverse thrust installations are a voluntary safety feature not required by the CAR, but used for safety in daily operations.

(2) No reverse thrust performance credit to date has been granted except for the Boeing 377 with reversible propellers.

(3) The airworthiness regulations of SR-422B, Sections 4T.115(b) and 4T.122(f), provide specifically for some degree of credit.

(4) The CAR 40.77 operational landing distance 0.6 factor is believed inadequate for turbine transports certificated under the current SR-422 series regulations to compensate for adverse runway conditions in daily operations without the reverse thrust reserve decelerating feature.

(5) Some operators have voluntarily added 1,000 feet or a 10-percent margin to the required field length for their actual operations under adverse runway conditions.

(6) It would not be legal for Operations personnel to apply an arbitrary factor to be used in cases of inoperative antiskid braking systems without the authority of certification performance limitation data.

(7) Action has been initiated towards a test program for developing more realistic landing/accelerate-stop distance limitations.

It has been decided that no runway distance credit should be considered for the effects of reverse thrust until further notice.

4. SUMMARY.

a. Type certification reverse thrust performance credit is allowed by Sections 4T.115(b) and 4T.122(f) of SR-422B for accelerate-stop and landing distances.

b. The adequacy of presently approved landing field lengths (including the 0.6 factor under CAR 40.77) for safe operation without reverse thrust has been questioned since past records indicate that overrun accidents were substantially reduced as a result of the installations of reversing propellers and reverse thrust.

c. Some airline operators are voluntarily adding a margin to the required field lengths (0.6 factor included) for adverse runway surface conditions.

d. As a result of higher threshold speeds and of touchdown point further down the runway, as shown in the photographic survey of operational jet transport landings, the present required field lengths are not representative of actual operations.

5. CONCLUSIONS.

Boeing's request for reverse thrust performance credit for landing distances with the antiskid system inoperative is denied.

REVIEW CASE NO. 16. DESIGN FLAP SPEED AND INTERMEDIATE FLAP SETTINGS (Issued 17 July 1963)

1. INTRODUCTION.

Cessna proposed to type certificate their Model 336 with a flap design speed of 120 m.p.h. when the flaps are extended 30 degrees. An intermediate flap position of 10 degrees at speeds up to 160 m.p.h. is also proposed.

The flap position is controlled by a spring-loaded switch which the pilot must hold in the "on" position while selecting flap position. Flap position is determined by reference to an indicator mounted on the instrument panel. The full attention of the pilot is required for selecting a flap position at speeds above 120 m.p.h. to be certain that the limit deflection of 10 degrees is not exceeded.

The Central Region has raised a question concerning this procedure of determining flap deflection because of the demand on the pilot's attention and the possibility of overextending the flaps at higher speeds with the associated danger of structural failure.

2. HISTORY.

Intermediate flap settings have been used in the past for increased takeoff performance and, in some cases, better climb performance. The settings have been limited to the airspeed for full flap deflection and the intermediate flap setting was usually made on the ground before takeoff. This did not create a problem of pilot attention to the flap indicator while in flight nor were there any dangers of structurally overextending the flaps at higher speeds.

In some of the newer designs, it is desirable to use some flap at the higher airspeeds in order to help slow the airplane more rapidly in the terminal area. Cessna advocates using the landing flap for this purpose. This is done by small angles of flap deflection at speeds above the maximum flap deflection speeds.

In a memorandum dated March 8, 1962, the Central Region requested a ruling on the intent of the regulations concerning the use of intermediate flap settings at speeds higher than the design flap speed.

3. FACTS IN THE CASE.

a. When used in the manner proposed by Cessna, the intermediate flap setting will aid in slowing the airplane to the landing configuration speed in less time than would be required if additional drag devices were not used. Since there is a danger of exceeding the design structural limitations due to overextending the flaps at the higher speed, it is necessary that the pilot closely monitor the flap position indicator and the airspeed indicator while extending the flaps at any speed above the design flap speed. The Central Region considers such a procedure not to be in accordance with the intent of the regulations.

b. The regulations referenced by the Central Region as pertaining to this matter are as follows:

(1) CAR, Section 3.190, Flaps Extended Flight Conditions, defines the minimum design speeds for full flap deflection based on the stalling speed of the airplane. Additional design data are required for critical combinations of airspeed and flap position when an automatic flap load limiting device is employed.

(2) CAR, Section 3,223, Wing Flaps, allows for design requirements for wing flaps, their operating mechanism, and supporting structure when an automatic flap load limiting device is employed.

(3) CAR, Section 3.338, Wing Flap Controls, requires that means shall be provided to indicate flap position to the pilot. If any flap position other than fully retracted or extended is used, such means shall indicate each flap position. This section does not contain any structural requirements.

(4) CAR, Section 3.742, Flaps - Extended Speed, defines the speed for flaps-extended flight and allows for additional combinations of flap setting, airspeed, and engine power, providing the structure has been proven for the corresponding design conditions.

(5) CAR, Section 3,381, Pilot Compartment - General, requires that the pilot be able to operate the controls in the correct manner without unreasonable concentration and fatigue. This section is a general requirement and does not contain any structural requirements.

c. Other airplanes have been certificated with intermediate flap positions and an indicator to show flap position to the pilot. Cessna Models 210, 310, and 320 airplanes have already been DMCR approved with such systems. Certain models of the Beech Model 18 are also approved with intermediate flap settings, but the service and accident records of these airplanes do not show any adverse service experience resulting from the use of intermediate flap positions. None of these airplanes incorporates the use of an automatic flap load limiting device. The Central Region indicates that full compliance with the requirements will require a more positive means for obtaining the intermediate flap position. This could be a system that would allow the pilot to preselect flap settings and would demand much less attention while obtaining the intermediate position, or a system with a load limiting safety feature that would preclude structural overload.

4. CONCLUSIONS. It is concluded that:

a. The proposed wing flap system on the Cessna Model 336 fulfills the requirements of CAR, Sections 3.190, 3.223, and 3.338.

b. Cessna must provide proof of structure for ten degrees of flap deflection at 160 m.p.h. for compliance with CAR, Section 3.742.

c. A multiple opinion evaluation of Cessna's procedure must be made to determine compliance with CAR, Section 3.381(a). Unreasonable pilot concentration, the probability of overshooting flap angle and/or airspeed, and the margins of flap setting and/or airspeed will receive major consideration in this evaluation. If this evaluation determines that the average pilot could easily overshoot the limits, operating limitations must be provided. The value of these operating limitations will be established to assure that the maximum flap and/or airspeed limits will not be exceeded.

REVIEW CASE NO. 17 THE RECENT ACCIDENTS RESULTING FROM SPINS INVOLVING NORMAL CATEGORY PART 3 AIRCRAFT AND THE SOUTHERN REGION'S REQUEST FOR CONCURRENCE TO REQUIRE PIPER TO CORRECT THE SPINNING CHARACTERISTICS OF THE MODEL PA-28-180 TO ELIMINATE ANY UNCONTROLLABLE FEATURES THAT ARE PRESENT AFTER THE ONE TURN FROM POINT OF ORIGINAL HEADING TEST (Issued 17 July 1963)

1. INTRODUCTION.

Several recent accidents resulting from spins involving normal category aircraft, prompt a review of this matter in order to ensure proper guidance for field representatives and a minimum level of safety for the airmen and the aircraft for all environments in which it operates. Current regulation (Section 3.124) requires that aircraft to be certificated must pass a test of a one-turn spin with recovery by normal use of controls. Section 3.10 of the Civil Air Regulations requires that the aircraft shall not possess any feature or characteristic which renders it unsafe. As recent accidents involving Part 3 aircraft have apparently resulted from uncontrollable spins, action to carefully review this matter and determine the proper course of action is believed to be required.

2. BACKGROUND AND HISTORY.

a. Development of present-day regulations:

(1) Bulletin 7A issued October 1, 1934, required six turns in a spin with recovery in one and one-half turns by neutralizing control surfaces. On November 13, 1945, Section 3.135(n) became effective changing the spin requirement to one turn which today is current with the provisions that recovery shall be completed in one turn with assist from the controls for normal recovery. Throughout the changes of these regulations, the requirement in Section 3.124 that uncontrollable spins should not develop from any normal or abnormal use of controls, has remained unchanged.

(2) A question as to the reference point for measuring one turn has been raised by the Southern Region. Except for SO-210, the reference point used by all other regions has been the heading of the airplane at the time that prospin controls are applied. SO-210 feels that the reference heading should be the heading at the time that the airplane starts autorotating. (This is a difficult point to use on a standardized basis by different pilots because of personal opinion as to just when spinning has started.)

(3) Fort Worth and Kansas City personnel have investigated uncontrollable characteristics beyond one turn (from original heading) but have based their recommendation for approval on strictly one turn.

NOTE: Fort Worth personnel, in 1948, approved a Luscombe 11A, at Washington request, on one-turn characteristics only, and even though it was known to possess uncontrollable characteristics if permitted to spin more than one turn.

Mr. Walter Haldeman, Chief, Engineering and Manufacturing Branch, Southern Region, states that, in 1945, a Bellanca air cruiser model was not approved with Washington concurrence because of a similar characteristic which could not be detected in one turn only.

(4) Two accidents have occurred in the last 12 months on certificated aircraft as a result of stall demonstrations followed by uncontrollable spinning. The Mooney Model 20 was loaded beyond the aft C.G. limit and the Piper PA-28-150 was spun beyond one turn.

(5) Two airplanes were lost during official FAA type certification tests from uncontrollable spinning. A Piper PA-28-180 was spun beyond one and three-fourth turns and a Mooney Model 20C was spun at the aft C.G. requested by the manufacturer which was moved forward before a type certificate was issued.

(6) The Mooney Model 20C and the Piper PA-28-180 meet the stall requirements of Section 3.120 when the controls are applied to correct roll and yaw by unreversed use of the rolling and directional control. Maximum control travel is not needed in this maneuver. Full fast positive application of the directional control in the stall buffet area results in a snaproll on the PA-28-180 and a spiraling spin entry on the Mooney Model 20C.

(7) On November 13, 1945, Part 03 was adopted which amended the stall requirement of Section 4a.676. At the same time the spin requirement was changed from a six-turn to a one-turn spin requirement, Section 04a.676 required, for a straight stall only, sufficient directional and lateral control so that when the airplane is stalled, the downward pitching motion following the stall shall occur prior to any uncontrollable roll or yaw.

(8) The new stall requirements under Part 03 required stall demonstrations from straight and turning flight. After the stall, not more than 30 degrees pitch below level, 15 degrees roll or loss of altitude in excess of 100 feet was permitted. Where clear and distinctive stall warning is present (five percent above the stall), any loss of altitude in excess of 100 feet or any pitch in excess of 30 degrees shall be entered in the approved airplane flight manual. Subsequently, changes to the stall requirement includes the addition of climb flight stall. The correction of roll and yaw up to the stall by unreversed use of controls up to the time the airplane pitches and a clear and distinctive stall warning have been required between five and ten miles per hour above the stall in straight and turning flight (an aural stall warning is acceptable). All normal category Part 3 airplanes are required to be placarded against intentional spinning.

b. Spin requirements with respect to pilot certification:

Prior to August 15, 1949, the student pilot was introduced to recovery from spins before solo flight in a spinnable aircraft. The private pilot had to demonstrate recovery from a right and left spin of at least one turn while accompanied by an inspector or a flight instructor. The commercial pilot was required to demonstrate a two-turn spin in each direction with a precision recovery executed of not more than plus or minus ten degrees. The flight instructor was also required to spin the aircraft two or more turns in either direction with a precision recovery. After August 15, 1949, Amendment 20-3 deleted the spin requirements for the private and commercial pilot applicants. Concurrently, Amendment 43-6, deleted the pre-solo spin recovery instructions required for student pilots. These changes in spin requirements were brought about in an effort to increase the interest of aircraft manufacturers to produce a more spin-resistant or spin-proof aircraft. It also sought to interest operators of flight schools to use spin-resistant or spin-proof aircraft. It was also believed that a greater interest in aviation would be shown when such aircraft were used and flight instructions did not require testing in spins. The CAB accident record shows that accidents resulting from spins are quite rare; and the CAB Bureau of Safety Regulations has stopped using the "spin-stall" category classification, and more appropriately classifies these accidents as "stall" accidents.

Considerable emphasis in pilot training is placed on the recognition and recovery from stalls. Today, the flight instructor is required to demonstrate entry and recovery from spins to the right and left from airplanes and glider testings. On May 8, 1962, this was modified when the regions were advised by FS-440 that there was no objection to acceptance of logbook entries certifying dual-spin instruction in lieu of spin demonstrations on the flight instructor flight tests. This policy was established as a means of overcoming the problem of finding suitable test aircraft for spin demonstrations due to the extension of controlled airspace throughout the country.

c. Summary of accidents involving stalls and spins:

(1) A review of stall-spin accidents shows that:

(a) During the 1961 period, 405 accidents were reviewed and 23 of these accidents were from a spin and/or spiral. Witnesses observed the altitude in most all cases to be below 1,000 feet,

(b) In Texas, a mechanical failure appears to have been involved in a spin which on May 15, 1962, took four lives,

(c) In Florida, on June 7, 1962, the PA-28-180 was lost during spin tests; and

(d) In Texas, on September 14, 1961, a Mooney M20C was involved in an uncontrollable spin accident in which the FAA pilot bailed out, and the airplane crashed.

(2) On June 12, 1962, the Southern Region sent a letter to Piper Aircraft Corporation discontinuing project A213 SO-D because of the uncontrollable spin encountered during FAA investigation of Section 3.124. Piper, during further tests, revealed that an uncontrollable spin might occur after one and three-fourths to two turns. The region has taken the position that the aircraft does not meet the spin requirement and is not eligible for a type certificate in accordance with Section 3.124.

(3) On July 3, 1962, a telegram was received by the Engineering and Manufacturing Division from the Southern Region. The telegram stated that Piper contends that spin investigation under Section 3.124 should not extend beyond one turn before application of recovery. Piper has spun the airplane one and three-fourths turns and affected recovery by opposite use of rudder, elevator, and full power application and does not consider this procedure abnormal. Piper contends that other airplanes possess identical spin characteristics. The Southern Region considers the aircraft to have an unsafe feature and claims that the aircraft does not meet the intent of Section 3.10. The Southern Region has advised the Piper Aircraft Corporation that the airplane does not meet the intent of Section 3.10. The Engineering and Manufacturing Division's concurrence is requested.

(4) Piper in a letter to FS-100 dated July 3, 1962, submitted the following statements in regard to the spin characteristics of the PA-28-180:

(a) The PA-28-180 airplane has never failed to recover from a one-turn spin of any nature in one additional turn or less.

(b) The PA-28-180 will recover from one and one-half to one and three-quarter turn spins in either direction with the application of the opposite rudder followed by forward elevator control.

(c) Results of the most recent spin test indicate that the airplane will recover from a fully developed right turn spin after one and three-quarter to two turns with full forward elevator control, opposite rudder, opposite aileron, and full throttle. Additional turns required to recover from this maneuver is a maximum of two and three-eighths turns.

3. SUMMARY.

a. Although true spinning accidents in service are rare, the FAA is now officially aware of two such accidents involving two different models, both of which the FAA knows from its own tests are designs which have uncontrollable spinning characteristics under certain conditions. Both accidents have occurred within a reasonably short time after type certification.

b. In accordance with Section 1.24, the Administrator is required to take corrective action when the service record indicates such action is warranted. He is also required not to type certificate a new model when an "unsafe feature or characteristic" is found. (Reference Section 3.10 and the Federal Aviation Act of 1958, Section VI which requires that the Administrator may prescribe in any such certificate such other terms, conditions, and limitations as are required in the interest of safety.)

c. Sections 3.755 and 3.779 permit the Administrator to require placards or information when unusual design features or characteristics are found to warrant such action to assure safe operation.

d. Accidents, because of latent uncontrollable spinning characteristics, may occur when demonstrating stalls and recoveries, when "spin-inducing" control movements are employed, and when the airplane is loaded fully with a C.G. near or exceeding its approved rearward limit.

e. Presently established pilot-training policy does not require proficiency in spin entry or recovery; it concentrates on complete familiarity with recognition and recovery from the stall attitude. The stall characteristic of the Part 3 aircraft and compliance thereof is therefore fundamental to design and operations.

4. CONCLUSIONS.

a. The Piper Aircraft Corporation provides the following information as part of the PA-28-180 airplane flight manual or placard in accordance with Sections 3.755 and 3.779, Operation Procedures. The information shall provide a description of:

(1) Stall characteristics.

(2) Spin entry resulting from abnormal use of controls.

(3) Conditions in which an uncontrollable spin may be expected.

b. To determine if regulations and compliance have deteriorated and if regulations realistically reflect the policies of the Service. Activate a flight test project to fully explore the stall and spin characteristics of single-engine Part 3 aircraft. Industry will be invited to participate and primary emphasis will be placed upon Piper-, Cessna-, Beech-, and Mooney-built aircraft.

c. Based upon the results of the FS-100 flight research project and as found necessary, resolve differences of opinion and propose revisions to Part 3, specifically Section 3.124 and other appropriate parts of Part 3.

d. That FS-400 will immediately place, under its Safety Education Program, major emphasis in the field of stall recognition and recovery and speed control. This undertaking will include review and analysis of training and certification standards and procedures in stall recognition and recovery.

REVIEW CASE NO. 18 REQUEST BY DOUGLAS AIRCRAFT COMPANY FOR DISPATCH OF MODEL DC-8-50 SERIES AIRPLANES WITH ONE OR MORE THRUST REVERSERS INOPERATIVE (Issued 17 July 1963)

1. INTRODUCTION

Douglas Aircraft Company desires approval of its proposal so that these airplanes will be fully eligible for scheduled air carrier certification and operation with one or more thrust reversers inoperative. The primary objective of the Douglas Aircraft Company request is to enable an airline operator of these airplanes to dispatch or continue a planned passenger flight to its final intended destination when one or more thrust reversers become inoperative. Without this approval, the scheduled flight of the originating airplane would have to be interrupted at the first intermediate stop until the thrust reverser installation was repaired and the airplane returned to an airworthy condition.

2. CHRONOLOGICAL HISTORY

a. Proposed changes to landing distance requirements of SR-422B "Special Civil Air Regulation; Turbine-Powered Transport Category Airplanes of Current Design"

Minutes of September 1961, conference on performance and operating requirements for turbine-powered transport category airplanes, pages 27 and 28

A representative of American Airlines stated the following: "....American Airlines considered it necessary to apply a factor of 1.16 to the certificated landing distance determined for the Boeing 707-120, and for other types of airplanes of later vintage, it was found that a factor of 1.09 was adequate." (It is understood that the term "airplanes of later vintage" applied to airplanes equipped with fan engines as the thrust reversers on these engines are more effective than the thrust reversers on the original engines.)

b. Dispatch of DC-8-50 aircraft with all reversers inoperative

Douglas letter of March 22, 1962, to WE-210

This letter and its attachments introduced Douglas' proposal and presented its reasons and justification for requesting approval with one or more thrust reversers inoperative.

The Douglas proposal includes the establishment of an additional operating speed limit which would be lower than the original limit and the restriction of the maximum operating altitude, so that the airplane decelerating ability and emergency descent time down to an altitude of 14,000 feet would be equivalent or better than those currently approved with all thrust reversers operative. These airplanes are now certificated using the two inboard thrust reversers to comply with the high-speed slowdown requirements of CAR 4b.711 to prevent the exceedence of airspeed limits inadvertently. The DC-8-50 series airplanes do not use the outboard thrust reversers in flight.

The Douglas Aircraft Company is requesting the following type of approval:

(1) One or both inboard or all reversers inoperative dispatch with the following limitations:

(a) For altitudes not to exceed 25,000 feet

(b) For an airspeed not to exceed 290 knots (IAS) (except emergency descent which remains Mach/IAS = 0.8/Barber Pole).

(2) One or both outboard reversers inoperative dispatch with no new limitations.

c. Dispatch of DC-8-50 series aircraft with all thrust reversers inoperative

WE-210 memorandum of May 8, 1962, to FS-100

This memorandum contains a repetition of the Douglas Aircraft Company proposal and presentation, together with copies of the pertinent correspondence between the Douglas Aircraft Company and the Western Regional Office. The Western Region Engineering and Manufacturing Branch concluded that the Douglas Aircraft Company request was unacceptable and requested our concurrence with its position and/or our comments.

The Western Region Engineering and Manufacturing Branch based its recommended disapproval on the basis that the presently approved airport field lengths are considered inadequate for safe operation without reverse thrust even though the certificated airport field lengths for the DC-8-50 series aircraft were established without any performance credit for reverse thrust. The Branch stated that it is certain that the records would show that overshoot accidents of large aircraft were not reduced to a tolerable level, even with piston-engine transports, until after the airplanes were equipped with reversing propellers.

The Convair Models 22, 22M and 30 aircraft have been approved on a dispatch deviation basis (inclusion of inoperative thrust reversers on the minimum equipment list) for operation with one or more inoperative thrust reversers by the Western Region Operations Branch. The Western Region Engineering and Manufacturing Branch voiced no objection at the time, but has since concluded that such approval should not have been granted and wishes to initiate action to rescind the approval.

d. Acknowledgement of Douglas Aircraft Company request for dispatch of Model DC-8-50 series airplanes with one or more thrust reversers inoperative.

FS-100 wire of May 16, 1962, to WE-210

This wire acknowledged the memorandum from the Western Region Engineering and Manufacturing Branch dated May 8, 1962, and informed that Branch that the Douglas Aircraft Company proposal would be processed as an Engineering and Manufacturing Division review case.

3. APPLICABLE REGULATIONS (CAR 4b and SR-422B)

a. Section 4b.711 (Maximum operating limit speed V_{MO}/M_{MO}):

This section states "The maximum operating speed limit is a speed which shall not be deliberately exceeded in any regime of flight (climb, cruise, or descent), ...This operating limitation, denoted by the symbols V_{MO}/M_{MO} (airspeed or Mach number, whichever is critical at a particular altitude), shall be established to be not greater than the design cruising speed V_C and sufficiently below V_D/M_D or V_{DF}/M_{DF} to make it highly improbable that the latter speeds will be inadvertently exceeded in operations".

b. Section 4T.115(b) (Accelerate-stop distance):

This section states "In addition to, or in lieu of, wheel brakes, the use of other braking means shall be acceptable in determining the accelerate-stop distance, provided that such braking means shall have been proven to be safe and reliable, that the manner of their employment is such that consistent results can be expected in service, and that exceptional skill is not required to control the airplane."

c. Section 4T.122(f) (Landing distance):

This section states "In addition to, or in lieu of, wheel brakes, the use of other braking means shall be acceptable in determining the landing distance, provided such braking means shall have been proven to be safe and reliable, that the manner of their employment is such that consistent results can be expected in service, and that exceptional skill is not required to control the airplane".

4. SUMMARY

a. Type certification reverse thrust performance credit is allowed by Sections 4T.115(b) and 4T.122(f) of SR-422B for accelerate-stop and landing distances. The certificated accelerate-stop and landing distances for the DC-8-50 series aircraft do not include performance credit for their thrust reverser installations. Pilots of the DC-8-50 series aircraft are now accustomed to operating these aircraft with all of the deceleration benefits of reverse thrust for accelerate-stop and landing conditions, and the pilots have adjusted their piloting procedures accordingly. Then when the pilots are occasionally called upon to operate the airplanes without the benefit of thrust reverse, the accelerate-stop and landing procedures are no longer normal and natural, and this could result in a lower level of safety operationally.

b. At least one airline operator is voluntarily adding a margin to the required field length for landing (0.6 factor included).

c. All turbo-jet transport airplanes have been certificated to date with thrust reversers except the Sud Aviation Caravelle SE-210 and the North American NA-265 airplanes.

d. Douglas Aircraft Company proposes to establish new, lower operating speeds and a lower maximum operating altitude such that the DC-8-50 series aircraft, with inboard reversers inoperative, will possess emergency descent and slowdown times which are better than those currently approved.

e. The Convair 22, 22M and 30 aircraft have been approved by the Western Region Operations Branch on a dispatch deviation basis (inclusion of inoperative thrust reversers on the minimum equipment list) for operation with one or more thrust reversers inoperative.

5. CONCLUSION

a. The Douglas Aircraft Company's request for operating the DC-8-50 series aircraft with one or more thrust reversers inoperative may be granted on the following basis:

(1) Performance: At least two symmetrically placed reversers must be operative.

(2) Flight Characteristics: Emergency descent and slowdown performance must be equal to or better than that currently approved with all thrust reversers operative. Appropriate operating limitations must be applied.

b. Convair Models 22, 22M and 30. The approval and inclusion of inoperative thrust reversers on the minimum equipment list for these airplanes must be revised in accordance with the conditions specified in conclusion No. 1a. above.

REVIEW CASE NO. 19. LOCKHEED C-141A - MAXIMUM ALLOWABLE SPEED DISPLAY
(Issued 12 April 1963)

1. ORIGIN

a. The Southern Region determined that the proposed Air Force display of maximum allowable speed on the mach tape is not in compliance with CAR 4b.603(a), airspeed indicating system which requires that the airspeed indicator incorporate a maximum allowable airspeed indication which includes compressibility limitations.

b. The Special Projects Office (SPO) determined that the maximum allowable speed need only be displayed on the mach tape and has requested that the Engineering and Manufacturing Division, FS-100, review the Southern Region's interpretation of CAR 4b.603(a).

2. REGULATIONS AFFECTED.

a. CAR 4b.603(a) Airspeed indicating system. "If the airspeed limitations vary with altitude, the airspeed indicator shall incorporate a maximum allowable airspeed indication showing the variations of (V_{MO}/M_{MO}) with altitude including compressibility limitations."

b. CAR 4b.603(j) "Machmeter for airplanes having compressibility limitations not otherwise indicated to the pilot in accordance with section 4b.742."

c. CAR 4b.732 Airspeed limitation information. "The airspeed limitations (see sec. 4b.741(a) shall be presented in such a manner that they can be easily read and interpreted by the flight crew."

3. HISTORY

a. The Southern Region's letter to the Aeronautical Systems Division (ASD) dated May 11, 1962, states in part, "The Civil Air Regulations, Part 4b.603(a) is specific in that where maximum speed indication V_{MO} is variable with altitude (including compressibility limitations), this indication shall be incorporated in the airspeed indicator. Mechanization of the maximum speed indication in the machmeter only is not in compliance with the CAR."

b. Minutes of Preliminary Type Certification Board Meeting held on May 25, 1962, state in part, "In addition, the SPO understands that for certification the safe airspeed index must be displayed against the airspeed tape and the servo loop required for its operation must be monitored."

c. Minutes of the meeting of July 19, 1962. It is stated in paragraph 4 therein that the FAA would be satisfied with an arrangement of maximum safe speed indication on the airspeed tape instead of mach tape. It also stated that the indication of limit markings on the airspeed tape and limit markings on the mach tape might be satisfactory to FAA.

d. In a memorandum from the Southern Region dated November 9, 1962, the Southern Region forwarded a letter from the Aeronautical Systems Division which requested FS-100's interpretation of CAR 4b with respect to certain display and monitoring features. The monitoring aspects of the C-141 tape instruments have been resolved separately and are not considered in this engineering review case. (Reference FS-100 memorandum dated November 20, 1962, to Southern Region, Subject: Performance Standards for CADC/Tape Instruments Systems, Lockheed Model 300 (C-141A) Aircraft; in reply to SO-210 memorandum dated October 12, 1962.)

4. FACTS IN THE CASE

a. In showing compliance with 4b.603(a) Airspeed indicating system, 4b.732 Airspeed limitation information, and 4b.741(a) Airspeed limitations, on past turbine-powered airplanes, an airspeed indicator was modified to incorporate a "barber pole" hand which was activated by both airspeed in knots and mach inputs. This installation was used to display limit speed in a manner easily read and interpreted by the flight crew. All of our acceptable civil experience has been with this type of combined instrument which was calibrated in knots.

b. Section 4b.603(a) of the Civil Air Regulations specifically requires that airspeed limitations shall be shown on the airspeed indicator. This section distinguishes further between the airspeed indicator and machmeter (reference 4b.603(a) and 4b.603(j)); thus, the two instruments are not considered interchangeable.

c. The applicant (SPO) proposes to utilize the machmeter rather than the airspeed indicator to present airspeed limitations. The limit hand of the machmeter would be programmed to present the limit airspeed in terms of mach number. At lower altitudes, this mach number would be continuously changing so as to represent the limit airspeed at all altitudes where dynamic pressure "q" is of primary concern. At high altitudes, this mach number would be constant to reflect the compressibility limits.

d. Any system must present information in a manner easily read and interpreted by the flight crew as specified in 4b.732. For new types of display or methods of presenting speed information, the applicant must provide substantiation that the new aspects presented will, with regard to safety of operation, be equal to the methods used and shown to be operationally feasible and safe on currently operating aircraft of the same type; i.e., transport airplanes. No information is available on the operation of transport type aircraft using the speed limitation marked on a mach tape rather than on the indicator which indicates speed in knots.

e. The data presented by the ASD is not sufficient to determine that the presentation of "q" limits on the mach tape can be easily read and interpreted by the flight crew as required by CAR 4b.732. The information submitted pertains to the operation of aircraft that do not possess a "q" or structural limit speed. The effectiveness of machmeter markings on an airplane that possesses only mach limits does not provide substantiation to determine the adequacy of such markings for aircraft which possess both "q" and mach limits.

5. CONCLUSIONS

a. The maximum airspeed limitation as proposed by the ASD does not comply with CAR 4b.603(a). This regulation specifically requires that the airspeed indicator shall incorporate a maximum allowable airspeed limitation showing the variation of V_{MO}/M_{MO} with altitude including compressibility limitations.

b. The Air Force has not presented sufficient information to substantiate nor have they demonstrated that speed limitation information presented on the mach tape on a transport type airplane results in a display which is easily read and interpreted by the flight crew as required by CAR 4b.732.

c. The presentation of maximum airspeed limitation, as proposed by the ASD, does not comply with CAR 4b.603(a). If the ASD desires further consideration of its proposal, it should submit substantiation and demonstrate that those provisions of CAR 4b.603(a) not complied with are compensated for by factors which provide an equivalent level of safety as specified in CAR 4b.10.

REVIEW CASE NO. 20. HUGHES 269A ALTITUDE FLIGHT TESTS (Issued 12 April 1963)

1. INTRODUCTION. Hughes Tool Company, Aircraft Division, has made a verbal request to the Western Region for a decision on the altitude test requirements to be applied to the Model 269A helicopter (supercharged engine installation) for a determination of compliance with CAR 6.111 and CAR 6.116. The Western Region has requested (telegram 071930) a decision on the matter from FS-100. The Hughes Tool Company is awaiting a decision before proceeding with the design changes.

2. BACKGROUND AND HISTORY.

   a. Development of Present Day Regulations:

      (1) CAR 6 effective May 24, 1946, contained no detailed requirements on helicopter flight testing. The height-velocity envelope and autorotative landing characteristics were not spoken to in this regulation.

      (2) CAR 6 effective January 15, 1951, introduced detailed regulatory material on flight test requirements. Two of these requirements were CAR 6.111 and CAR 6.715. CAR 6.111 required a safe landing following engine failure at any point on the takeoff flightpath. CAR 6.715 (later changed to CAR 6.116) required the development of limiting height and speeds for safe landing following power failure (height-velocity envelope) and made this evelope a flight limitation. These regulations did not specify where the determination of satisfactory characteristics should be made, i.e., sea-level, altitude, or both.

      (3) On October 1, 1959, the height-velocity envelope requirements of CAR 6.715 were transferred to CAR 6.116 and therewith were changed from a flight limitation to performance information. This has been the only substantive change in the autorotative landing requirement since 1951.

(4) Circular Memorandum No. 60-9 dated September 1, 1960, sent to all regions for comments, contained draft regulatory material calling for the determination of the height-velocity envelope at altitude. The Circular Memorandum indicated that this material could be used as guidance material after obtaining concurrence with the Washington Office. After comments from the region, this material was forwarded to FS-40 for proposed rule making on January 18, 1962. When Federal Aviation Agency Order MS 1320.12, dated May 1, 1962, was issued cancelling Circular Memorandums, CM 60-9 was cancelled.

(5) On October 2, 1962, a set of special conditions was issued for the forthcoming certification program on the Army turbine powered LOH helicopters. Because these helicopters contained considerable altitude operational performance, one of the special conditions called for a determination of the autorotative landing characteristics at altitude.

(6) On December 11, 1962, a Notice of Proposed Rule Making (NPRM) was issued through Draft Release 62-52. One of the proposals in this NPRM called for a determination of the autorotative landing characteristics at altitude.

b. Pertinent Current Regulations and Proposed Regulations:

(1) Current Regulations:

CAR 6.10 Eligibility for type certificates. A rotorcraft shall be eligible for type certification under the provisions of this part if it complies with the airworthiness provisions hereinafter established or if the Administrator finds that the provision or provisions not complied with are compensated for by factors which provide an equivalent level of safety: Provided, That the Administrator finds no feature or characteristic of the rotorcraft which renders it unsafe.

CAR 6.111 Takeoff. The takeoff shall be demonstrated at maximum certificated weight, forward center of gravity, and using takeoff power at takeoff rpm and made in a manner such that a landing can be made safely at any point along the flightpath in case of an engine failure and shall not require an exceptional degree of skill on the part of the pilot or exceptionally favorable conditions. Pertinent information concerning the takeoff procedure, including the type of takeoff surface and appropriate climbout airspeeds, shall be specified in the operating procedures section of the Rotorcraft Flight Manual. (See secs. 6.116, 6.740, 6.742, and 6.743.)

CAR 6.116 Limiting height and speeds for safe landing following power failure. If a range of heights exists at any speed, including zero, within which it is not possible to make a safe landing following power failure, the range of heights and its variations with forward speed shall be established together with any other pertinent information, such as type of landing surface. Such an envelope shall be established in full autorotation for an single-engine helicopters and with one engine inoperative for multiengine helicopters provided that engine isolation design features are incorporated to assure continued operation of the remaining engines. (See sec. 6.743 (c).)

(2) Proposed Regulations:

Civil Air Regulations Draft Release No. 62-52, Dated December 11, 1962.

By amending CAR 6.111 to read as follows:

6.111 Takeoff.

(See also 6.116, 6.740, 6.742, and 6.743.)

(a) The takeoff shall be demonstrated at maximum certificated weight, forward center of gravity, and using takeoff power and takeoff r.p.m.

(b) The takeoff shall be made in a manner such that a landing can be made safely at any point along the flight path in case of an engine failure, and shall not require an exceptional degree of skill on the part of the pilot or exceptionally favorable conditions.

(c) Compliance with the provisions of paragraph (b) of this section shall be shown at the maximum certificated weight under sea level conditions, and at weights selected by the applicant for altitudes up to the maximum altitude anticipated for takeoffs and landings.

(d) Pertinent information concerning the takeoff weights and altitudes shall be specified in the performance information section of the Rotorcraft Flight Manual. Information concerning the takeoff procedure, including the type of takeoff surface and appropriate climb-out airspeeds, shall be specified in the operating procedures section of the Rotorcraft Flight Manual.

By amending CAR 6.116 to read as follows:

6.116 Limiting height and speeds for safe landing following power failure.

By amending 6.116 by adding in the first sentence between the words "established" and "together" the words "at the maximum certificated weight and at other weights and corresponding altitudes selected by the applicant."

c. History of Helicopter Autorotative Landing Characteristics:

(1) The helicopters certificated to the CAR 6, 1946 requirements (from 1946 through approximately 1953) had limited altitude performance capabilities. Examples are as follows:

Model	Hovering Ceiling At Gross Weight (IGE)	Year Approved
Sikorsky S-52	3750 ft.	1947
Bell 47D	2800 ft.	1948
Hiller UH-12	3250 ft.	1949
Sikorsky S-51	5000 ft.	1949
McCulloch MC4C	2550 ft.	1953

(2) From approximately 1953 to the present, there was a steady increase in the altitude performance capabilities of helicopters. This was brought about at first by the necessity of meeting the 4000 ft. hovering requirement introduced in 1951; and later by a desire on the part of operators for even better altitude operating performance. In order to achieve this improved performance, larger engines, derated engines and/or supercharged engines were installed. Examples of their altitude performance capabilities are:

Model	Hovering Ceiling At Gross Weight (IGE)	Year Approved
Sikorsky S-58	4000 ft.	1956
Cessna CH-1B	10600 ft.	1957
Brantly B-2	4500 ft.	1959
Hiller UH-12E	8500 ft.	1959
Bell 47GB	20000 ft.	1960

(3) The 1951 requirements called for a determination of autorotative landing characteristics with no specific reference either to sea-level or altitude. Altitude tests, however, were not considered necessary in the early years of this regulation because it was assumed, without knowledge to the contrary, that the autorotative landing characteristics established at sea-level were valid at altitude, or if suspected of not being valid at altitude were not a cause of concern because of the limited altitude performance capabilities of the helicopters.

(4) In more recent years, however, there has been increasing evidence that altitude does have a very deteriorating effect on the autorotative landing characteristics, particularly where these characteristics are marginal at sea-level.

Thus from 1955 until the present, there has been an ever increasing amount of altitude testing in the certification program because of either marginal sea-level autorotative characteristics, combined with limited altitude performance capabilities, or because the helicopter possessed above average altitude performance capabilities. (In addition, altitude tests were required for larger CAR 6 helicopters, and later CAR 7 helicopters, because their anticipated use in scheduled operation dictated a more thorough investigation of the autorotative landing characteristics.)

(a) In 1955, Hiller was required to run autorotative landing characteristics at altitude on the Hiller Hornet (HJ-1) because of the marginal sea-level autorotative landing characteristics. The applicant was unable to verify satisfactory autorotative landing characteristics at altitude, and the helicopter was never certificated. An autorotative landing accident occurred during these tests.

(b) In 1958, because of anticipated altitude operation, autorotative landing altitude tests were required of the Vertol 44 for approval of takeoffs and landings at altitude. The results of these tests showed that altitude had a marked effect on the autorotative characteristics. The exact extent of this effect was not determined at the time.

(c) In 1959, autorotative landing altitude tests were conducted on the Hiller UH-12E by mutual agreement between Hiller and the FAA. These tests results were inconclusive as they ended in an accident.

(d) In 1959, during the Bell 47G3 certification program autorotative landing characteristics were checked at altitude. These tests were not required by the FAA because of the helicopter's excellent sea-level autorotative landing characteristics. However, the manufacturer chose to conduct these tests to assure himself of satisfactory autorotative landing characteristics at altitude. These tests showed that altitude did, in fact, have a significant effect on the autorotative landing characteristics.

(e) During the period, 1960 to 1962, the Sikorsky S-61L, Sikorsky S-62, and the Vertol 107 II, (all CAR 7 transport helicopters) were tested at altitude for the reasons given in the first paragraph of (C) (4).

(f) In 1962, a determination of the autorotative landing characteristics at altitude was required on the Bell 204. These tests showed that altitude had a marked effect on the autorotative landing characteristics.

(g) As mentioned in (A) (5), a determination of the effects of altitude on the autorotative landing characteristics is being required for the forthcoming certification of the three LOH helicopters. Although the sea-level autorotative landing characteristics are as yet unknown, the expected altitude performance capabilities have been considered to be sufficient justification to require these altitude tests under CAR 6.10.

d. Research and Development Testing:

(1) In 1961, the Flight Standards Service requested the Aircraft Development Service to make a determination of the altitude effects on autorotative landing characteristics.

In 1962, these tests were run on a Bell 47G3B helicopter at Bishop and Fresno, California, at altitudes of sea-level, 4000 ft., 7000 ft., and 10,000 ft. Test results while not finalized, in a formal report, as of this date, have definitely proven that altitude does have a marked deteriorating effect upon the autorotative landing characteristics established at sea-level.

e. Background of the Hughes 269A Helicopter:

(1) On April 10, 1959, the Hughes helicopter was certificated with a sea-level engine. After the occurrence of several autorotative landing accidents and further testing the helicopter was grounded in August 1962. Several improvements were made to the helicopter and it was returned to service on August 31, 1962. The autorotative characteristics, however, meet only the minimum requirements with little or no margins.

In January 1963, the height-velocity envelope was again rerun on the 269A after minor design improvements, (i.e., rotor blade modification, landing gear cross tube heat treatment, etc.) These modifications resulted in minor improvements in the autorotative landing characteristics.

(2) Hughes now proposes the installation of a Lycoming engine with an air research exhaust driven supercharger. This will give the helicopter the capability of achieving maximum continuous power up to 15,000 feet and thus excellent altitude performance capabilities.

(3) Based on results of the altitude tests recently conducted by ADS, and to a lesser extent on other past certification tests, considerable deterioration can be expected in the altitude autorotative characteristics of this helicopter.

3. SUMMARY OF THE FACTS.

a. The present regulations (CAR 6.111 and CAR 6.116) call for determination of autorotative landing characteristics without specific reference to either sea-level or altitude.

b. The recent ADS project (343-10V) noted in d(1), has shown that altitude has a significant deteriorating effect on the autorotative characteristics.

c. Altitude tests have been deemed necessary and required on three CAR 6 helicopters (i.e., HJ-1, V-44, and the LOHs), and four CAR 7 helicopters (i.e., S-62, S-61L, V-107, and the Bell 204).

d. The latest regulatory proposal concerning the height-velocity envelope (Draft Release No. 62-52) has clearly recognized the need for altitude testing, and included requirements for such determination.

e. Certificated helicopters (other than the Hughes Model 269A) do not warrant further altitude investigation, inasmuch as they possess one or more of the following:

(1) Limited altitude performance capabilities.

(2) Sufficiently good autorotative landing characteristics at sea-level, to assure reasonably good characteristics at altitude.

(3) A satisfactory service record.

f. The Hughes 269A has recently been grounded for unsatisfactory autorotative landing characteristics. Although it has been returned to service, and has since incorporated additional minor improvements, it is considered to meet the autorotative landing characteristics at sea-level with little margin.

4. CONCLUSIONS.

a. In consideration of the foregoing, there is no evidence that the Hughes 269A helicopter can provide the level of safety at altitude consistent with the requirements of CAR 6.111 and CAR 6.116, without altitude testing.

b. Based on the facts it is determined that, in addition to the tests at sea level to show compliance with the requirements of CAR 6.111 and 6.116, altitude tests are also required unless an altitude operating limitation is placed on the helicopter.

REVIEW CASE NO. 21 CHAMPION AIRCRAFT CORPORATION MODEL 402 - SOURCES OF POWER FOR GYROSCOPIC INDICATORS (Issued 24 May 1963)

1. INTRODUCTION.

Champion Aircraft Corporation, in a telegram addressed to the Administrator dated February 28, 1963, questioned the interpretation of CAR 3.668, Gyroscopic Indicators, as applied to their electric gyro installations in the Model 402 airplane by the Central Region. Subsequent discussion of the matter with the Central Region disclosed that a difference of opinion exists between Champion Aircraft Corporation and the Central Region as to the interpretation of CAR 3.668(a) as applied to two independent sources. The Champion Aircraft position is that the engine-driven generators are the power sources under the intent of CAR 3.668(a). The Central Region takes the position that the electric gyro inverters are the power source for the gyro indicators since AC power is required for their operation. In view of the above, the objective of this review case is to determine the intent of CAR 3.668(a) with regard to power sources for electric gyroscopic indicators and other related regulatory considerations governing the acceptability of the Champion Model 402 installation.

2. CHRONOLOGICAL HISTORY.

a. The Central Region, in a letter dated February 21, 1963, advised Champion Aircraft Corporation that the primary power source for the Directional Gyro (DG) and the Gyro Horizon (HG) indicators is considered to be the power supply providing the AC power to these instruments, and as such must be duplicated for compliance with CAR 3.668.

b. Mr. Robert Brown, President, Champion Aircraft Corporation, telephoned Mr. P. D. Wilburn, Assistant Chief, Flight Test Branch, Branch, regarding the Model 402 gyro instrument controversy on February 21, 1963. Mr. Wilburn explained the Agency organizational structure to Mr. Brown and advised him of the procedure he should follow to resolve the controversy.

c. Champion Aircraft Corporation, in a telegram addressed to the Administrator dated February 28, 1963, questioned the interpretation of CAR 3.668 as applied to the electric gyro installations in the Model 402 airplane and asked for an immediate Washington interpretation of the rule.

d. Champion Aircraft Corporation, in a telegram addressed to the Administrator, dated March 1, 1963, asked for a reply to their telegram of February 28, 1963.

e. The Director, Flight Standards Service, in a March 5, 1963 telegraphic reply to the Champion Aircraft Corporation, advised Champion that advice concerning resolution of the problem would be forwarded to the Central Region by March 7, 1963.

f. On March 7, 1963, Mr. W. H. Weeks, Chief, Engineering and Manufacturing Division, advised the Central Region that the decision would be delayed as a review case to resolve the matter was being prepared.

g. The A.I.R. Corporation (manufacturer of the gyro instrument inverter) under cover letter dated April 4, 1963, submitted limited data pertaining to the gyro instrument inverter to the Washington Office for consideration.

3. FACTS IN THE CASE.

a. When gyroscopic indicators are installed in multiengine airplanes, the Civil Air Regulations require, (3.668(a)), that there be provided at least two independent sources of power; a manual or an automatic means for selecting the power source; and a means for indicating the adequacy of the power being supplied by each source. The following note is also included in the regulation: "NOTE: Power sources are not considered independent if both sources are driven by the same engine."

b. The Champion Model 402 airplane is a two-engine aircraft equipped with a generator on each engine and suitable means of power source selection and indication.

c. Airframe and Equipment Engineering Report No. 50, dated August 21, 1962, entitled "Design Guide for Personal Aircraft Electric Systems," under Paragraph 13.2.3 reads as follows:

> Isolation of Electric Sources: Maximum reliability in a two generator system is only obtained when each generator circuit is completely independent. A single fault (except a bus fault) would then only result in loss of one generating source of power. This feature is particularly significant when there is a requirement that an essential utilization device be supplied from two independent sources of power.

d. Limited data supplied by A.I.R. Corporation (manufacturer of the gyro instrument inverter) under cover letter dated April 4, 1963, indicates that the inverter is a single, dual output unit which supplies 115 VAC for the DG and 26 VAC for the HG. This data also indicates that failure in either system associated with the separate inverter outputs will not affect the remaining gyro indicator. It appears, however, that a failure in the primary of the inverter will result in the loss of both gyro indicators. Likewise, a line fault resulting in a blown inverter fuse will also interrupt power to both gyro indicators.

e. The following is a review of the development of the current CAR 3.668, Gyroscopic Indicators, as applicable to the intent of the regulation concerning a definition of "source of power."

> November 1, 1937. CAR 04.5805. Gyroscopic Instruments. All gyroscopic instruments shall derive their energy from engine-driven pumps or from auxiliary power units. Each source of energy supply and its attendant complete installation shall comply with the instrument manufacturer's recommendations for satisfactory instrument operation. On multiengine aircraft, each instrument shall have two separate sources of energy, either one of which shall be capable of carrying the required load. Engine-driven pumps, when used, shall be on separate engines. The installation shall be such that failure of one source of energy or breakage of one line will not interfere with proper functioning of the instruments by means of the other sources.

The above is the first time any reference to power supply for gyroscopic instruments is made in the regulations. No background material is available regarding the need and/or intent of this regulation. No changes were made in the regulation until November 13, 1945, when Part 03, Airplane Airworthiness, appeared for the first time.

> November 13, 1945. CAR 03.5215. Gyroscopic Indicators (Air-Driven Type). All air-driven gyroscopic instruments installed in ... on multiengine airplanes, the following detail requirements shall be applicable:
>
> 1. Two sources actuated by separate means shall be provided, either one of which shall be of sufficient capability to operate, at the service ceiling of the airplane in normal cruising condition, all of the air-driven gyroscopic instruments with which the airplane is equipped.
>
> 2. A suitable means shall be provided in the attendant installation where the source lines connect into a common line to select either suction air source for

> the proper functioning of the instruments should failure of one source or breakage of one source line occur. When an automatic means to permit simultaneous air flow is provided in the system, a suitable method for maintaining suction shall be provided. In order to indicate which source of energy has failed, a visual means shall be provided to indicate this condition to the flight crew.

The above regulation (CAR 03.5215) clearly permits an applicant to join the two sources of energy into a common system. The sources referred to are suction sources, but an application of the intent of the regulation to electromotive sources would permit an applicant to join two generators into a common system.

CAR Draft Release No. 55, dated May 22, 1955, proposed this regulation in essentially the form in which it is quoted above (03.5215). The draft release and the few comments received concerning this particular regulation as proposed in the draft release do not contain any discussion of the intent of the regulation concerning a definition of "source of power."

> <u>November 1, 1949</u>. CAR 3.668. <u>Gyroscopic Indicators (Air-Driven Type)</u> as amended by CAR Amendment 3-7, Gyroscopic Indicators, effective March 5, 1952.
>
> All gyroscopic instruments installed in.... In addition, the following provisions shall be applicable to multiengine airplanes:
>
> (1) There shall be provided at least two independent sources of power, a manual or an automatic means for selecting the power source, and a means for indicating the adequacy of the power being supplied by each source.
>
> (2) The installation and power supply systems shall be such that failure of one instrument or of the energy supply from one source will not interfere with the proper supply of energy to the remaining instruments or from the other source.

The intent of this regulation, as applicable to a definition of "source of power" is considered to be the same as the intent of CAR 03.5215, dated November 13, 1945, (see above), especially since the preamble to Amendment 3-7 states, "Several minor changes have also been made, the most notable ones pertaining to ... and to the power supply for gyroscopic indicators."

May 15, 1956. CAR 3.668. Gyroscopic Indicators, as amended by Amendment 3-3 effective May 17, 1958, Amendment 3-5 effective October 1, 1959.

All gyroscopic instruments installed in.... In addition, the following provisions shall be applicable to multiengine airplanes:

(1) There shall be provided at least two independent sources of power, a manual or an automatic means for selecting the power source, and a means for indicating the adequacy of the power being supplied by each source.

NOTE: Power sources are not considered independent if both sources are driven by the same engine.

(2) The installation and power supply systems shall be such that failure of one instrument or of the energy supply from one source will not interfere with the proper supply of energy to the remaining instruments or from the other source.

CAR 3.668 is currently worded exactly as shown above for May 15, 1956. Since this current wording is essentially the same as the wording in CAR 3.668 dated November 1, 1949, the intent of the current regulation, as applicable to a definition of "source of power," is the same as the original intent (see November 13, 1945, CAR 03.5215).

4. CONCLUSION.

Nowhere in the development of CAR 3.668 is there a discussion of the definition of "source of power." A review of the development of the regulation leads to the conclusion that:

a. "Sources of power" are intended to mean the source required by the utilization devices (electro gyro indicators). To interpret the requirement otherwise would defeat the purpose of the requirement, that is, provide the availability of two independent sources of power to the utilization devices.

b. If AC power is required for the operation of the electric gyro indicators, two independent sources of AC power at the proper voltage and frequency must be available to each gyro indicator in order to show compliance with the provisions of CAR 3.668.

In view of the above, it is further concluded that the Champion Model 402, as presented, does not comply with the requirements of CAR 3.668.

REVIEW CASE NO. 22 VERTOL 107-II EQUIVALENT SAFETY PROPOSAL
(Issued 10 July 1963)

1. ORIGIN

a. The Eastern Region has made a written request for a review case decision on their finding that there are compensating factors, under the provisions of CAR 7.10, that provide an equivalent level of safety at higher altitudes where the Vertol 107-II does not comply with CAR 7.711.

b. The present regulation, CAR 7.711(a), states that the never-exceed speed (V_{ne}) shall not be less than the best rate of climb speed (BRC speed). As V_{ne} decreases with altitude, it intercepts the BRC speed on the Vertol 107-II at approximately 11,200 feet. Above this altitude, the aircraft does not comply with the regulation as written since a scheduled climb speed is used that parallels the V_{ne} speed. Approval has been granted by the Eastern Region on the basis of equivalent safety.

2. REFERENCE REGULATIONS

a. CAR 7.10 Eligibility for type certificates.

A rotorcraft shall be eligible for type certification under the provisions of this part if it complies with the air-worthiness provisions hereinafter established or if the Administrator finds that the provision or provisions not complied with are compensated for by factors which provide an equivalent level of safety: Provided, that the Administrator finds no feature or characteristic of the rotorcraft which renders it unsafe.

b. CAR 7.10 General.

(1) The performance prescribed in this subpart shall be determined using normal pilot skill and shall not require exceptionally favorable conditions. Compliance shall be shown for sea level standard conditions in still air and for the range of atmospheric variables as selected by the applicant. The performance as affected by engine power, instead of being based on dry air shall be based on 80 percent relative humidity or 0.7" Hg. vapor pressure whichever is less.

(2) Each set of performance data required for a particular flight condition shall be determined with the powerplant accessories absorbing the normal amount of power appropriate to that flight condition.

c. CAR 7.711 Never exceed speed V_{ne}.

(1) The never exceed speed shall be established. It shall not be less than the best rate-of-climb speed with all engines operating at maximum continuous power, nor greater than either of the following:

(a) 0.9V established in accordance with section 7.204, or

(b) 0.9 times the maximum speed demonstrated in accordance with section 7.140.

(2) It shall be permissible to vary the never-exceed speed with altitude and rotor rpm, provided that the ranges of these variables are sufficiently large to allow an operationally practical and safe variation of the never-exceed speeds.

3. BACKGROUND AND HISTORY

a. Development of present day regulations:

(1) CAR 6, Effective May 24, 1946, contained no quantitative requirements for the establishment of V_{ne}. At that time none of the flight requirements were stated in quantitative terms.

(2) CAR 6, Effective January 15, 1951, introduced detailed quantitative regulatory material. One such requirement was that V_{ne} shall not be less than "the maximum level flight speed with all engines operating at maximum continuous r.p.m. and 90 percent maximum continuous power."

(3) CAR 6, Amendment 6-4, Effective May 16, 1953, deleted the requirement stated in a(2) above and substituted in lieu thereof a requirement that the V_{ne} shall not be less than "the best rate of climb speed." This change was requested at that time by industry because they were experiencing difficulties in meeting the requirement of a(2).

(4) CAR 7, Effective August 1, 1956, related the V_{ne} to the best rate of climb speed, as had been done in CAR 6, Amendment 6-4.

b. Eastern Region's request for review case:

(1) October 5, 1962. Eastern Region memorandum to FS-100 requested clarification of the definition for the BRC speed.

(2) November 2, 1962. FS-100 memorandum answered the Eastern Region's October 5, memorandum stating ---"The definition of V_y as the best rate of climb speed in CAR 7.1(e)(8) is considered the speed at which the maximum rate of climb is achieved. V_{NE} may, therefore, not be less than this best rate-of-climb speed as stated in CAR 7.711(a).

We can not concur with any other interpretation of CAR 7.711(a) under the provisions of CAR 7.10 without evidence of compensating factors which provide an equivalent level of safety and present no unsafe feature."---

(3) November 20, 1962. Eastern Region memorandum requested a review case decision from FS-100 on their finding of an equivalent level of safety for noncompliance with CAR 7.711 under the provisions of CAR 7.10.

(4) January 10, 1963. FS-100 memorandum to the Eastern Region requested further detailed justification for equivalent safety and their proposed approval parameters in order that FS-100 may proceed with the review case.

(5) February 21, 1963. Eastern Region answered the FS-100 memorandum (above) relating items they felt could substantiate their claim for the equivalent level of safety proposal.

4. ANALYSIS

a. Present CAR 7.711(a) regulation. This specifies that V_{ne} shall not be less than the BRC speed. This presents no problem at sea level where a large spread generally exists (i.e., 30 to 80 m.p.h.) between V_{ne} and the BRC speed. V_{ne}, however, is required to be less than blade stall roughness speed and also less than the speed for which structural substantiation has been obtained.

As both of these values became progressively lower at altitude, V_{ne} in turn must be progressively reduced at altitude. As the V_{ne} decreases at altitude, it will eventually come in conflict with CAR 7.711 by intercepting the BRC speed lines. This will thus result in an altitude limit unless steps are taken to raise this V_{ne}, (i.e., reduce weight and thereby raise the blade stall roughness V_{ne} and/or conduct additional structural testing at altitude to raise the structural V_{ne}).

On older CAR 6 helicopters, the intersecting point for the V_{ne} and the BRC speed has generally been above the maximum anticipated operating altitude and has not been reported as a problem, likely due to the limited altitude performance capabilities of these helicopters. With the advent of CAR 7 helicopters, however, more attention has been given to altitude considerations and it has come to the attention of the Federal Aviation Agency (FAA) that the V_{ne} might be lower than the BRC speed at some practical operating altitude. This would necessitate an altitude limit on the operation of the helicopter, unless other steps are taken as mentioned earlier. To date, even with the high degree of altitude performance achieved on several helicopters, the V_{ne}, BRC speed requirement has not been a serious problem.

There have been two isolated cases of which we are aware (Sikorsky S-55 and Sikorsky S-62) where a scheduled climb speed has been used in place of the BRC speed in order to avoid conflict with the V_{ne} at altitude. (See Figure 1)

These aircraft had somewhat limited altitude performance and no reports were made of a problem in this area. For this reason, the FAA has not been previously concerned.

CAR 7 helicopters of recent manufacture, such as the multi-turbine airline helicopter, with good enroute altitude potential, have focused far greater attention and concern on the BRC speed - V_{ne} problem area.

(See Figure 1, for examples of helicopters with high altitude performance where conflict may exist between BRC speed and V_{ne}.)

EXAMPLES OF HELICOPTER V_{NE} SPEED RELATION TO BRC SPEED

Helicopter Model	V_{ne} at Max. Altitude	Climb Speed At Max. Altitude (Is this speed BRC speed?)		Maximum Altitude	Does Conflict Exist Between V_{ne} and BRC Speed
Bell 47G-3	45 m.p.h. @ 2650 #	45 m.p.h.	Yes	20,000 ft. (limitation)	No
Cessna CH-1C	50 kts. @ 3100 #	50 kts.	Yes	16,000 ft. (Max. altitude of performance and V_{ne} presentation)	No
Sikorsky S-55	50 kts.	38 kts. (*)	No	10,000 ft. (Max. altitude limit of V_{ne} presentation) (12,000 ft. Max. alt. of performance presentation)	Yes
Sikorsky S-58	63 kts. @ 2500 RPM	63 kts.	Yes	8,000 ft. (limitation on 13,000 # version)	No
Sikorsky S-61	68 kts.	58 kts.	Yes	12,000 ft. (limitation)	No
Sikorsky S-62	45 kts. @ 93% N_F 40 kts. @ 90% N_F	40 kts. (*)	No	10,000 ft. Max. altitude of performance and V_{ne} presentation	Yes
Vertol 107-II	46 kts. (**) @ 100% Rotor RPM	65 kts.	Yes	13,000 ft. (limitation	Yes

* If BRC speed were used a conflict would exist with the V_{ne} speed.

** Noncompliance

FIGURE 1

b. Background material used for the V_{ne} regulation. A search of background material used for the V_{ne} regulation (CAR 6, Amendment 6-4) has not revealed the reasons for this decision other than that the Civil Aeronautics Administration (CAA) recommended keeping the original requirement of CAR 6 effective January 1951 and industry recommended no minimum value on V_{ne}.

The Civil Aeronautics Board (CAB) decision on CAR 6, Amendment 6-4, concluded that BRC speed should be the minimum acceptable V_{ne} speed.

c. Temporary operation below BRC speed. The present regulation permits operation of the helicopter below the BRC speed as a transient maneuver or even as a temporary cruise maneuver. The transient maneuver is necessary and must be done in passing from hover to cruising flight and returning.

d. Continuous operation below BRC speed. Continuous cruising flight below BRC speed is a different matter (as opposed to temporary operation) since it requires more pilot attention and alertness.

When flight is conducted below the BRC speed, the aircraft is operating on the back side of the power required curve which results in an increased pilot workload.

When cruising in this area, a speed reduction results in a temporary climb, followed by descent, and an increase in speed, results in temporary descent followed by climb. This requires more attention in attempting to hold cruise altitude. Therefore, where performance is critical as it is on the Vertol at the proposed operational altitudes small speed variations produce large changes in rates of climb or descent.

By necessitating cruise flight below the BRC speed, as proposed by the Eastern Region, the pilot has no alternative but to operate on the back side of the power required curve. He is also being persuaded to operate as close to V_{ne} speed as possible in order to avoid an even steeper slope portion of this curve. This area of operation can not be accomplished with the same ease as operation above the BRC speed provides.

e. Eastern Region suggestion. The region has suggested that this problem could be solved if the weight were reduced at altitude, adding, however, that this would further complicate an already complicated V_{ne} placard. This objection is not considered valid. A weight reduction limitation at altitude would normally not be placed on a V_{ne} placard but rather in the weight section of the rotorcraft operating limitations. The V_{ne} placard would then be determined in the usual manner by flight testing the helicopter within its limitations, i.e., weight, CG, etc., at each altitude.

f. Problems on other helicopters. The Eastern Region points out that this condition has existed on other helicopter models. In CAR 6 helicopters the maximum anticipated operational altitude has traditionally been treated in a loose manner thus making it difficult to identify the problem.

On CAR 7 helicopters, however, the maximum anticipated operational altitude takes on much greater importance because of the expected use of CAR 7 helicopters in scheduled passenger operation.

It would appear reasonable, therefore, to assume that the altitude to which performance information is presented would serve as the maximum anticipated operational altitude and that compliance with the regulation should exist up to that point.

g. Compensating factors. The Eastern Region contends that the following provide for an equivalent level of safety:

(1) Use of a scheduled climb speed above 11,000 feet which parallels the V_{ne} line provides the necessary margin. This maneuver has been flight evaluated by EA-216 and has been found to be safe, effective, and easy to produce.

Comment

The above is not considered a compensating factor since nothing new has been provided. In addition it forces the pilot to fly on the back side of the power required curve and therefore requires more pilot skill and alertness. (This would be in conflict with CAR 7.110(a) since this regulation states that "the performance prescribed in this subpart shall be determined using normal pilot skill and shall not require exceptionally favorable conditions.")

(2) Use of the above proposal (i.e., scheduled climb speed above 11,000 feet) is compensated for by the fact that the operational envelope is extended approximately 3,000 feet and would permit better terrain clearance capability, better on top weather capability, etc.

This would provide a level of safety equivalent to or higher than that envisioned by the original intent of the regulation.

Comment

The above proposal does not provide for an equivalent level of safety since this approach would permit the helicopter to operate continuously at high altitude. At high altitude where climb performance is already reduced, the use of a scheduled climb speed (below the BRC speed) would reduce this performance still further. The use of a scheduled speed below the BRC speed would place the helicopter in an area of the back side of the power required curve which for continuous operation would require high pilot skill and alertness. (Also in conflict with normal pilot skill required with CAR 7.110(a)).

It is, therefore, felt that the advantages claimed by the proposal are more than offset by the above disadvantages.

(3) Controllability and stability are not limiting at 13,000 feet.

Comment

This may be true but can not be considered a compensating factor in showing equivalent airworthiness for noncompliance with CAR 7.711.

5. SUMMARY OF FACTS

a. CAR 7.711(a) requires that V_{ne} shall not be less than the BRC speed. V_{ne} of necessity, must be reduced at altitude. On helicopters which have high altitude performance capabilities, this reduction can reach the point where it intercepts the BRC speed. Approval of operation above this point can then only be accomplished by noncompliance with CAR 7.711.

b. V_{ne} on the Vertol 107-II crosses the BRC speed at 11,200 feet and thus creates an altitude limitation. At this altitude, the Vertol 107-II has a low rate of climb.

c. When V_{ne} is below the best rate of climb speed, the pilot is forced to operate the rotorcraft on the back side of the power required curve. This in turn requires more pilot attention and engine power to maintain altitude. Because the slope of the power required curve steepens with progressively lower speeds, it also encourages the pilot to fly as close to V_{ne} as possible. These conditions call for high pilot skill and alertness.

d. CAR 7.10 requires compensating factors to provide an equivalent level of safety when there is noncompliance with the provisions of CAR 7.711(a). The compensating factors put forth by the region, i.e., higher allowable operating altitude for terrain clearance and better weather conditions do not offset the lower aircraft performance and higher required pilot attention. The above, therefore, can not be considered to provide compensating factors.

e. This problem could be solved by a weight reduction limitation at altitude which would provide adequate altitude performance, and thus maintain the same level of safety.

6. CONCLUSIONS

The compensating factors presented by the region do not in fact show an equivalent level of safety under CAR 7.10 when there is a noncompliance with CAR 7.711.

REVIEW CASE NO. 23. REMOVAL OF PILOT CHUTE FROM RESERVE PARACHUTES WHEN USED FOR PURPOSE OF SPORT JUMPING (Issued 10 July 1963)

1. INTRODUCTION. The president of Parachutes, Inc., has requested a reconsideration to an enforcement action by the Eastern Region requiring that surplus military parachutes, when used as the reserve (for emergency) in a dual pack parachute assembly, must not have the pilot parachute removed. The pilot parachute assists deployment of the canopy, and its removal seriously affects deployment time. It is claimed that the pilot chute has a tendency to foul with a partially deployed parachute (previously activated); thus, if the main parachute opens improperly and the reserve parachute is pulled, there is a possibility that the pilot chute of the reserve would become entangled with the main parachute rendering both ineffective. The Parachute Club of America also has requested that they be permitted to remove the pilot chute from their approved reserve parachute for sport jumping activities. The president of Parachutes, Inc., and also the president of the Parachute Club of America indicate that this modification is common practice and has been successful over a considerable period of time.

   Two safety problems are involved which require consideration:

   a. A possibility exists that the reserve parachute canopy may still foul in the main parachute suspension lines unless skill is experienced in deployment of reserve pack. Training is required to accomplish this successfully;

   b. The minimum safe altitude at which the parachute should be deployed is considerably increased over the commonly accepted 500 ft. minimum.

2. HISTORY. The Eastern Region is not convinced that removing the pilot chute from the reserve parachute will necessarily increase the level of safety for the jumper. They have recommended that we not permit removal of the pilot chute unless after removal the parachute is checked and found to meet the requirements of Technical Standard Order C23. The Eastern Region formed their opinion after consulting the Army Quartermaster Research and Development Center. In addition, the region points out that other modifications of significance are being made to parachutes which also may be critical. On February 26, 1963, the region issued instructions which, in effect, stated that the pilot chute could not be removed from the reserve parachute. Their decision was based on the fact that Federal Aviation Regulations 105 interprets military parachutes as "approved." However, this approval is based on the fact that the parachute must meet the appropriate military specification. If the pilot chute is removed, the parachute would no longer comply with the military specification and would not be considered as approved under FAR 105. By their memorandum of March 13, 1963, addressed to FS-100, the Eastern Region recommended the following:

a. A regulation amendment be issued to provide that parachutes with the pilot chute removed must be found to conform to the standards of TSO-C23;

b. That certificated riggers be requested to conduct a check for unauthorized alterations which have been made to parachutes and reject or approve them under the provisions of FAR 105.

The sport parachutists concede that removing the pilot chute seriously affects deployment of the parachute and would render it unsafe for use at low altitudes. They point out, however, that in sport parachute jumping they plan on activating their parachute at a relatively high altitude (i.e., over 2,000 ft.). They contend that the added risk attendant in slow deployment is less than the risk of having the pilot chute of the reserve parachute entangle with the main parachute if it were to deploy improperly.

The Parachute Club of America made the following recommendations concerning the removal of the pilot chute from the reserve parachute:

(1) FAA permit removal of the pilot chute from reserve parachutes and designate such parachutes "For Sport Parachuting Use Only";

(2) Require that the jumper activate his main canopy at no less than 2,200 ft. above the ground;

(3) Jumpers be instructed in the proper procedure in deploying the reserve parachute with the pilot chute removed.

a. <u>Regulatory Factors in the Case</u>.

(1) FAR 105 (Parachute Jumping), Section 105.43, requires a single harness dual parachute pack having at least one approved auxiliary parachute. Certain military parachutes are expressly designated as approved within the meaning of FAR 105.

(2) Civil Air Regulations, Maintenance, Repair and Alteration of Airframes, Powerplants, Propellers and Appliances, Section 18.1, excludes parachutes as an appliance; thus parachutes are exempt from the provisions of CAR 18 and the normal processes for approving alterations as applicable to other appliances are not available. Specific regulatory direction on how alterations may be approved and by whom (except for TSO articles under Part 514.5 which covers TSO parachutes) does not seem to be provided in pertinent regulations.

(3) FAR 65, Certification, Airmen Other Than Flight Crewmembers, Section 65.129(d), provides that no certificated parachute rigger may alter a parachute in a manner that is not specifically authorized by the Administrator or the manufacturer.

(4) Under Regulations of the Administrator, Part 514, TSO-C23, Parachutes, has been established to set forth those standards found necessary by the Administrator to assure that the particular article when used on civil aircraft will operate satisfactorily or accomplish satisfactorily its intended purpose under specified conditions. No specific procedures are set forth as to how the Administrator will authorize alterations to approved military type parachutes pursuant to FAR 65.129(d).

(5) CAR 1.55, Replacement and Modification Parts, provide guidance, in effect, that persons making modifications to approved products shall demonstrate continued compliance with the requirements applicable to the original approved product.

(6) FAA Advisory Circular No. 149.9-1, effective October 1962, covers the procedure to be followed for release of certain military surplus parachutes to the public by the Department of Defense. This circular does not cover the matter of alterations to the parachutes.

b. Additional Information:

Representatives of the Maintenance Division, FS-300, and Operations Division, FS-400, have advised us that it need be recognized that sport parachuting is an activity unique insofar as the utilization of the parachutes is concerned. For untrained and inexperienced persons who would jump only in emergency, sometimes at low altitudes, it is essential that the parachutes be equipped with pilot chutes for rapid deployment of the canopy. However, sport parachutists who belong to parachute clubs are trained and experienced and plan their jumps. It is considered that these factors more than compensate for the degradation of minimum safe deployment altitude and delay in deployment of the parachute.

Additionally, removal of the pilot chute reduces the possibility of accidental release of the parachute in the cabin of an airplane prior to the time the parachutist is due to jump. There have been a number of accidents in which the rip cord was accidentally activated and the pilot chute pulled the parachute canopy into the slip stream of the airplane dragging the jumper into the aircraft structure.

3. CONCLUSIONS.

a. The regulations as presently written do not adequately cover the modified equipment used by sport parachutists.

b. A TSO should be issued to cover the dual pack parachute equipment utilized by the sport parachutist.

c. CAR 18 and FAR 105 should be revised to prescribe appropriate procedures for handling alterations to parachutes on the same basis as other appliances.

d. As an interim measure, permission should be granted to remove the pilot chute from the reserve parachute of a dual pack used for sport jumping providing the following is adhered to:

(1) Removal must be accomplished by certificated parachute rigger;

(2) Record made on parachute packing card and rigger's log book;

(3) Parachutist must be made fully aware of removal of pilot chute and instructed in proper deployment method of auxiliary parachute;

(4) Main parachute must be deployed at an altitude of not less than 2,500 ft. above terrain. NOTE: Free-fall time from 2,500 ft. to 500 ft. is approximately 11.3 seconds. This should allow sufficient time for the jumper to realize his predicament and safely deploy his parachute.

(5) Auxiliary parachute pack must be labeled "Pilot Chute Removed". "This Parachute Eligible For Sport Jumping Only."

NOTE: An FS-1 telegram was sent to all regions on April 3, 1963, advising of the provisions for interim approval.

REVIEW CASE NO. 24 HYDRAULIC FLUID QUANTITY GAUGES; LOCKHEED MODEL 300 (C-141A) AIRCRAFT (Issued 6 September 1963)

1. INTRODUCTION.

During the initial Model 300 (C-141A) type design negotiations between the Georgia Division of the Lockheed Aircraft Company, the Aeronautical Systems Division of the Air Force Systems Command (ASD), and the Federal Aviation Agency, it was agreed by all parties that sight gauges mounted on each of three hydraulic fluid reservoirs would be an acceptable means for providing hydraulic fluid quantity information to the flight crew. The reservoirs are located in the cargo compartment and are accessible to the crew during flight. No fluid quantity information was to be presented in the cockpit area. The FAA C-141A Project Group later reversed its position and informed the Aeronautical Systems Division that quantity gauges should be accessible to a flight crew member at his station. ASD and Lockheed do not concur with this subsequent FAA determination. The Southern Region requested that this matter be made the subject of a review case.

2. CHRONOLOGICAL HISTORY.

a. Lockheed Aircraft Corporation letter to SW-210, dated December 28, 1961, concerning proposed design of the C-141A hydraulic system. Hydraulic reservoir fluid level indication was to be provided by a sight gauge mounted on each of the three reservoirs.

b. SW-210 letter to Lockheed, dated January 19, 1962, stating that the proposal regarding hydraulic reservoir fluid level indication was satisfactory in principle, and that approval would depend on the acceptability of the final design.

c. FAA C-141A Project Group (Southern Region) letter to Lockheed, dated September 18, 1962, advising that it had recently come to the attention of the C-141 Project Group that there was no hydraulic fluid quantity indication in the cockpit area, and stating that the FAA believes it is important that hydraulic fluid quantity information be constantly available to the flight engineer.

d. Lockheed letter to the FAA, dated October 12, 1962, advising that tooling and fabrication of the originally proposed reservoir design were well along, and that Lockheed does not believe remote reading quantity indicators in the cockpit area are required by CAR 4b. Lockheed presents several reasons for acceptance of the sight gauge design; added complexity and reduced overall reliability are cited as undesirable features of remote reading indicators.

e. FAA C-141A Project Group letter to Lockheed, dated November 19, 1962, stating that a study of problems associated with hydraulic systems showed the importance of hydraulic fluid quantity indication. The letter advises Lockheed that the FAA will require hydraulic fluid quantity gauges in a location accessible to a flight crew member at his station. CAR 4b.10, "Eligibility for Type Certificates" and recent service experience on other 4b jet transports are cited as the justification for the above requirement. No specific justification is cited. CAR 4b.10 requires ". . . Provided, That the Administrator finds no feature or characteristic of the airplane which renders it unsafe for the transport category."

f. Aeronautical Systems Division letter to the Southern Region dated March 19, 1963, stating that ASD considers sight gauges on the reservoirs to be satisfactory, and that they do not concur with the FAA requirement for a remote transmitting system. Reservoir accessiblity in flight, dual hydraulic systems for essential services, and the added complexity of a remote transmitting system are noted as substantiation for the ASD view.

g. SO-200 memorandum to FS-100 dated April 4, 1963, thoroughly explaining the subject controversy and transmitting copies of pertinent correspondence. SO-200 stated they were preparing to list the remote quantity indicator as a requirement for certification, and mentioned that FS-100 may wish to comment.

h. Conference on May 2, 1963, between Mr. C. Powers of ASD and FAA representatives regarding the subject matter. The participants discussed an FAA survey of hydraulic system components installed on U.S. transport category airplanes certificated during approximately the past 15 years. The survey showed that all but one airplane (Martin 404) have hydraulic fluid quantity gauges visible from a flight crew station. A schematic of the C-141A hydraulic system was examined. Mr. Powers stated there is sufficient redundancy in the system to permit the loss of any one of the three completely independent hydraulic systems without endangering the airplane and occupants. He commented that ASD believes remote quantity gauges for the three reservoirs are not necessary because of the redundancy designed into the hydraulic system.

i. FS-100 memorandum to SO-210 dated May 9, 1963, transmitting the above-mentioned FAA survey of hydraulic system components (par.2h) and a record of the conference between Mr. Powers and FAA representatives.

j. SO-210 memorandum to FS-100 dated May 24, 1963, requesting that the question of hydraulic fluid quantity gauges for the Lockheed C-141A aircraft be made the subject of a review case.

3. SUMMARY OF THE FACTS

a. Section 4b.654 "Hydraulic systems; design" of the Civil Air Regulations requires a means for indicating the pressure in each main hydraulic power system. CAR 4b does not specifically require a means for indicating hydraulic fluid quantity.

b. A survey of hydraulic system components installed on twelve U.S. transport category airplanes shows that all but one (Martin 404) have hydraulic fluid quantity gauges visible from a flight crew station.

c. Three independent hydraulic systems will be installed on the C-141A. The ailerons, elevator, rudder, brakes, wing spoilers, and wing flaps are each powered by two of the three independent hydraulic systems. The landing gear retraction system is powered by one hydraulic system (with "free fall" capability). Manual control of the ailerons and elevator is available. Hydraulic pressure gauges and low pressure warning lights will be installed on the flight engineer's panel. Hydraulic fluid quantity sight gauges will be installed on each reservoir. The sight gauges and reservoirs, located in the cargo compartment, are accessible in flight.

d. A survey of airline accidents which have occurred since January 1, 1955, in which a loss of hydraulic fluid occurred prior to the accident shows that undetected loss of hydraulic fluid was included in the probable cause of two accidents. Both accidents involved collision during ground taxi. It should be noted that the hydraulic systems in both airplanes (a DC-3 and a DC-7) include a means for indicating hydraulic fluid quantity to a crewmember at his station. The probable cause of one of the accidents was determined to be:

(1) Hydraulic system failure resulting in the loss of the brakes.

(2) Failure of the crew to detect a low level of hydraulic fluid.

(3) Lack of crew training in emergency procedures to be initiated in the event of a brake failure due to a loss of hydraulic fluid.

The probable cause of the other accident was determined to be:

(1) Hydraulic system flexible line ruptured.

(2) Lack of vigilance during taxiing and failure to observe loss of hydraulic system fluid.

e. About 80 percent of the mechanical reliability reports involving loss of hydraulic fluid are in the "total loss of normal system fluid" category; that is, all fluid in the normal system is lost because the malfunction is not detected before action can be taken to prevent the loss, or because the loss cannot be stopped by emergency crew action. To prevent an undetected loss of all fluid, hydraulic systems are generally designed so that an emergency supply of fluid is not routed to the pumps during normal operation. The emergency fluid supply is routed to normal or emergency pumps only after deliberate action by the crew. This emergency fluid supply feature is usually achieved by the use of a reservoir standpipe or by the use of multiple reservoirs. Since essential services on the C-141A are powered by two independent hydraulic systems, each with its own reservoir, the continuation of power to essential services in the event of a loss of fluid in one of the reservoirs is not dependent upon emergency action by the crew.

4. CONCLUSIONS

a. The Civil Air Regulations do not specifically require a means for indicating hydraulic system fluid quantity in transport category airplanes (paragraph 3.a).

b. In consideration of the redundancy of hydraulic power to essential services (paragraph 3.c.), and in consideration of the hydraulic system design which provides three independent hydraulic systems, each with its own reservoir (paragraphs 3.c. and 3.e), it is concluded that a requirement for hydraulic fluid quantity gauges in the C-141A in a location accessible to a flight crew member at his station is not justified in accordance with the provisions of CAR 4b.10.

REVIEW CASE NO. 25 . BOEING 727 - LONGITUDINAL STATIC STABILITY
(Issued 29 November 1963)

1. ORIGIN

a. Boeing has informally requested through the Western Region that an interpretation to the requirements of CAR 4b.151 be given as they are applicable to the Boeing 727 airplane.

b. The Western Region, in view of a possible conflict within the provisions of CAR 4b.151, has withheld any decision as to compliance or noncompliance of the Boeing 727 pending a review case on the subject.

2. REGULATIONS AFFECTED

CAR 4b.151 Static Longitudinal Stability

(a) A pull shall be required to obtain and maintain speeds below the specified trim speed, and a push shall be required to obtain and maintain speeds above the specified trim speed. This criterion shall apply at any speed which can be obtained without excessive control force and within the limits of elevator control power, except that such speeds need not be greater than the appropriate operating limit speed or need not be less than the minimum speed in steady unstalled flight.

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

(c) The stable slope of the stick force curve versus speed shall be such that any substantial change in speed is clearly perceptible to the pilot through a resulting change in speed.

3. HISTORY

a. Section 4b.151 is intended to ensure that the airplane's control feel is such that the pilot will be warned when the airplane departs from its trim speed. The section further is intended to ensure that if the airplane is displaced from the trim speed, either intentionally or inadvertently, it will return to or near the original trim speed without requiring corrective action on the part of the pilot.

b. To ensure the attainment of this intent, certain quantitative and qualitative standards were proposed with respect to stick force direction versus speed (4b.151(a)), return to trim speed capability (4b.151(b)), and stick force/speed gradient (4b.151(c)).

c. During the flight test evaluation of the Boeing 727 airplane, compliance with the generally accepted intent of 4b.151 was demonstrated (see subparagraph 3(a)) but some question exists as to whether the characteristics complied with the literal wording of section 4b.151.

d. The points in question are whether a stick force is required to maintain a speed departure from trim when that speed is within the allowable free return speed range (4b.151(b)) and whether the stick force/speed gradient requirement is applicable for speeds within the free return speed range.

e. Boeing contends that within the concept of the free return speed range, we should recognize that any given trim setting is applicable to a range of trim speeds and that on this basis stick force gradients and forces required to maintain a speed (i.e., 4b.151(a) and (c)) are not applicable.

4. FACTS IN THE CASE

a. The Boeing 727, when trimmed in the area of maximum Mach and "q", exhibits neutral static stability characteristics at speeds differing by less than 7 percent of the trim speed. That is, a pull force is required to obtain but not to maintain speeds within this range which is within the allowable 10 percent free return speed range of 4b.151(b).

b. The Boeing 727 has demonstrated the capability of maintaining any given speed hands-off within the ± 7 percent trim speed range when flown into turbulent air conditions the equivalent of ± 0.15"g" loads.

c. When deliberately upset from a trim condition at V_{MO}, or anywhere within its neutral static stability range, the airplane will return to within 10 percent of the original speed and to a speed that is always below V_{MO}/M_{MO}.

d. The direction of stick forces throughout the permissible speed range is always in the right sense. That is, a pull is required to obtain speeds less than the trim speed and a push is required to obtain speeds above the trim speed.

e. The neutral static stability characteristic occurs only between the altitude range of 20,000 to approximately 25,000 feet where the most critical conditions of Mach number and "q" exist.

f. Outside the speed range of neutral static stability, the airplane displays stick force/speed gradients well above the acceptable minimum of one pound per six knots.

g. The airplane has excellent maneuvering stability characteristics with the stick force never being less than 50 pounds per "g".

h. The airplane possesses heavily damped short period dynamic stability.

i. The airplane can be flown with normal pilot techniques and no more attention is required than is normally necessary for any airplane. Target altitudes and airspeeds are easily held to close tolerances.

j. The Western Region has conducted a flight evaluation to determine compliance with 4b.151 and 4b.155 and recommend acceptance of the flight characteristics.

k. The effects of control system friction, elevator control characteristics, compressibility effects, etc. are recognized indirectly by the free return speed range provision set forth in 4b.151(b).

5. CONCLUSIONS

a. The Boeing 727 is considered to comply with the intent of 4b.151 in that it requires correct pilot stick force inputs to depart from the trim speed, returns to a speed near the trim speed and within the allowable speed range when control force input is released, and does not require exceptional attention by the pilot to maintain a desired trim speed and altitude.

b. The Western Region is to assure that necessary information relating to the free return speed range is included in the Airplane Flight Manual.

c. The intent of the free return speed range of 4b.151 with respect to the applicability of stick force and stick force gradient, should be clarified. This clarification will be incorporated by the Washington Office into regulatory project #1198.

REVIEW CASE NO. 26 BELL HELICOPTER COMPANY REQUEST FOR A DECISION REGARDING FIREWALL INTEGRITY ON THE MODEL 206 HELICOPTER IN SHOWING COMPLIANCE WITH CIVIL AIR REGULATIONS 6.384 AND 6.483 (Issued 6 December 1963)

1. INTRODUCTION

a. The Bell Model 206 (LOH) helicopter is powered by an Allison 250-C10 gas turbine engine. The engine is mounted on the top of the fuselage structure aft of the main rotor transmission. The horizontal axis of the engine is approximately horizontal and parallel to the longitudinal axis of the fuselage. The helicopter fuselage, including the tail boom, is of semimonocoque construction. The upper fuselage panel in the area beneath the engine serves as a horizontal firewall, which, with vertical transverse firewalls at the forward and aft ends of the engine, isolates the engine from the remainder of the aircraft. Aluminum alloy cowling with screened cutouts is installed along both sides and over the top of the engine compartment.

b. The upper fuselage skin which forms the horizontal firewall panel is fabricated in sections. The center portion, which constitutes a drip pan, is of 0.020-inch corrosion-resistant steel. The remainder is honeycomb material with 0.012-inch titanium top skin (surface exposed to the engine compartment), 0.375 to 0.750-inch aluminum honeycomb, and a Fiberglas or aluminum alloy lower skin (surface remote from the engine compartment). Some of the loads from the tail rotor, horizontal stabilizer, and tail boom are transferred into the main fuselage structure through this horizontal panel.

c. Bell has been advised by the FAA Southwest Region that the firewall must retain its integrity as a firewall for a period of 15 minutes when subjected to a 2000° plus or minus 50°F. flame with loads based on powered flight for the first 5 minutes and loads based on autorotational flight applied for the remaining 10 minutes. Bell has disagreed and has indicated that, under CAR 6, they do not feel that it is necessary for the firewall to retain its integrity as a firewall under any conditions of loads beyond 5 minutes.

2. REGULATIONS AFFECTED

a. CAR 6.1(i)(1): "Fireproof material means a material which will withstand heat at least as well as steel in dimensions appropriate for the purpose for which it is to be used. When applied to material and parts used to confine fires in designated fire zones, fireproof means that the material or part will perform this function, under the most severe conditions of fire and duration likely to occur in such zones."

b. CAR 6.384: "All structure, controls, rotor mechanism, and other parts essential to a controlled landing of the rotorcraft which would be affected by powerplant fires shall either be of fireproof construction or shall be otherwise protected, so that they can perform their essential functions for at least 5 minutes under all foreseeable powerplant fire conditions. (See also Sections 6.480 and 6.483(a).)"

c. CAR 6.480: "General. The powerplant installation shall be protected against fire in accordance with Sections 6.481 through 6.486. Additional fire prevention requirements are prescribed in Subpart D, Design and Construction, and Subpart F, Equipment.

"Note: The powerplant fire protection provisions are intended to insure that the main and auxiliary rotors and controls remain operable, the essential structure remains intact, and that the passengers and crew are otherwise protected for a period of at least 5 minutes after the start of an engine fire to permit a controlled autorotational landing."

d. CAR 6.483: "Firewall. (a) Engines shall be isolated from personnel compartments by means of firewalls, shrouds, or other equivalent means. They shall be similarly isolated from the structure, controls, rotor mechanism, and other parts essential to a controlled landing of the rotorcraft, unless such parts are protected in accordance with the provisions of Section 6.384. * * * *

"(b) Firewalls and shrouds shall be constructed in such a manner that no hazardous quantity of air, fluids, or flame can pass from the engine compartment to other portions of the rotorcraft.

(c) All openings in the firewall or shroud shall be sealed with close fitting fireproof grommets, bushings, or firewall fittings.

(d) Firewalls and shrouds shall be constructed or fireproof material and shall be protected against corrosion."

3. HISTORY

a. Bell Report 206-193-014, outlining a proposed test procedure to substantiate the load carrying capabilities of aluminum alloy honeycomb materials used in the firewall panels, as well as other proposed fire tests, was submitted to the FAA by a letter dated July 12, 1963. In this report, it was proposed that the panels be subjected to a test flame of 2000° plus or minus 50°F. for a period of 5 minutes, then subjected to their critical loading. The portions of the panels simulating the cabin heater air inlet into the fire zone and cabin heater air outlet would be subjected to a 2000° plus or minus 50°F. flame for 15 minutes, without loading.

b. The Southwest Region, by a letter dated August 9, 1963, advised Bell that the proposed procedure was unsatisfactory because the loading conditions were not proper for the operational envelope of the rotorcraft, the simulated components were not adequately described, and fire-resistance tests for the Fiberglas portions were not included.

c. In a letter to the Southwest Region, dated August 26, 1963, Bell disagreed with the comments of the SW-210 letter of August 9. It was Bell's contention that Part 6 required fire protection to enable vital components of the helicopter to perform their intended function for only 5 minutes. Bell agreed that the portions required to be fireproof should resist flame penetration for 15 minutes, but did not concur that these components were required to carry any loads for this period of time. Based upon this disagreement, Bell requested a review by Washington and issuance of a statement of policy "in order that a uniform interpretation of the applicable regulations will be realized."

4. <u>DISCUSSION</u>

a. A basic question that must be answered in this case concerns whether or not the requirements of Part 6 effectively place a 5-minute ceiling on the time for which fire protection must be provided in the affected rotorcraft.

(1) Reference is made to a 5-minute period in Sections 6.384 and 6.480. In 6.384, it is required that, unless of fireproof construction, "all structure, controls, rotor mechanism, and other parts essential to a controlled landing of the rotorcraft. . . perform their essential functions for at least 5 minutes under all foreseeable powerplant fire conditions." In 6.480, the note states that the powerplant fire protection provisions are intended to protect essential components and occupants "for a period of at least 5 minutes after the start of an engine fire to permit a controlled autorotational landing."

(2) In each case where the time period is mentioned, it is also stated that the purpose of the fire protection provisions is to permit a controlled landing. If this objective is not satisfied, the fire protection that is provided serves no particularly useful purpose. If fire protection is not provided for a period of time sufficient to permit a controlled landing, it would have to be assumed that, at least under some possible operating conditions, a fire would result in an uncontrolled descent. For a normal

category rotorcraft, this would necessarily be regarded as a feature or characteristic of the rotorcraft which renders it unsafe.

(3) The specific reference in each case is to a period of "at least 5 minutes." This means not less than 5 minutes and in no sense imposes an upper limit on the time period for which protection must be provided. It is then consistent with the wording of the regulations to conclude that fire protection is to be provided for a period of time sufficient to permit a controlled landing under all foreseeable powerplant fire conditions, but in no case should this protection be provided for a period of less than 5 minutes after the start of an engine fire.

b. The firewall must comply with two separate but related requirements, CAR 6.384 and 6.483. (See 2. Regulations Affected.)

(1) The test procedure for demonstrating fireproof construction applicable to CAR 6.384 and 6.483 is specified in Flight Standards Service Release No. 453. This requires, among others, ability to carry loads and resist flame penetration when subjected to 2000^{o}F. for 15 minutes.

(2) The alternative requirement in CAR 6.384 in lieu of fireproof construction, provides for protection of essential structure and controls. The test procedure for demonstrating acceptable protection requires the ability to carry loads and resist any failure that could cause hazardous loss of control when subjected to the temperature resulting from any foreseeable fire for the duration of time appropriate to the operation. In this case, the loads are those of the appropriate emergency procedure specified in the Rotorcraft Flight Manual, in accordance with CAR 6.742, and the time is that necessary to complete the emergency descent, landing, and evacuation, starting at the maximum altitude for which normal operation is approved. However, in no case would a time less than 5 minutes be considered acceptable.

c. In the past, operational capabilities of helicopters limited them to relatively low altitudes. Under these conditions, designing to meet the 5-minute criterion for fire protection was considered to provide ample time for controlled autorotational landings. Recently, however, a number of helicopters have been certificated for operation up to 20,000 feet. Bell has advised that the Model 206 will also be certificated for operations at 20,000 feet. In view of the engine location in the Model 206, which makes direct observation impractical, the altitude for which it is being certificated, and the circumstance that no fire detection is provided, it appears that 5 minutes following the start of a fire is an insufficient period of time to allow for a controllable landing to be accomplished. It is understood, for example, that the single act of descending in autorotation from 20,000 feet will require more than 5 minutes. When sufficient time is allowed for recognizing the existence of a fire and initiating appropriate emergency action, it is likely that the total time required for a descent will be considerably in excess of 5 minutes.

d. Section 6.483 states that engines shall be isolated from personnel compartments and other vital parts of the rotorcraft by fireproof firewalls or shrouds. These firewalls or shrouds must be constructed in such a manner that no hazardous quantity of air, fluids, or flame can pass from the engine compartment to other portions of the rotorcraft.

(1) The Bell Helicopter Company agrees that fireproof material, in the sense involved herein, is "a material which will withstand a temperature of 2000° F. for 15 minutes without flame penetration."

(2) If the integrity of the firewall is destroyed, whether by direct action of flame or by structural deformation due to the weakening effect of the flame, it will cease to isolate the engine from the personnel compartment and the remainder of the rotorcraft. Under such circumstances, the firewall would no longer show compliance with 6.483(a).

(3) Inasmuch as Section 6.483(c) specifies fireproof material for the firewall, it must be concluded that it was the intent of this section to provide for containing a fire within the engine compartment for 15 minutes regardless of the concurrent reference to 5 minutes elsewhere. If 5-minute containment had been intended, fire-resistant material for the firewall would be adequate.

(4) There is no primary need for the firewall to perform its isolation function other than when the rotorcraft is in flight. It is necessary, therefore, to evaluate its adequacy when the appropriate flight loads associated with the particular rotorcraft are applied. Otherwise, the ability of the firewall to perform its required function under actual service conditions is not demonstrated. It is appreciated that the loading pattern may vary, depending upon various performance and structural patterns associated with any given rotorcraft.

(5) It is probable that the period of 5 minutes specified as a minimun has generally been adequate in the past. It is also probable that to provide protection for more than 15 minutes is not practicable at this time. It can be shown, however, (par. 4.c) that flight loads, either during powered or autorotational flight, will be applied to the rotorcraft structure for a period considerably in excess of 5 minutes with many new rotorcraft. Evaluation of the operational and loading patterns to determine the period of time during which isolation must be maintained and the loads actually applied to the firewall during this time rests with the office having responsibility for certification of the rotorcraft.

5. CONCLUSIONS

Having given due consideration to the available facts in this case, it is concluded that the test procedure proposed by the Bell Helicopter Company for their Model 206 helicopter will not result in a satisfactory demonstration of compliance with the fire protection requirements of Sections 6.384, 6.480, and 6.483. This conclusion is based upon the following detailed points:

a. The time period of 5 minutes specified in these sections represents a lower limit on the protection to be provided (4.a. and 4.b.).

b. The operational limits being established for the Model 206 helicopter would permit flight at altitudes from which an autorotational descent to the surface within a 5-minute period following the outbreak of a fire may not always be possible (4.c.).

c. Since a firewall must be of fireproof construction, it must resist penetration of flame for 15 minutes under the prescribed temperature conditions.

d. Under the provisions of CAR 6.384 all structure, controls, rotor mechanism and other parts essential to a controlled landing which would be affected by powerplant fires must be capable of withstanding appropriate flight loads, taking into account the time necessary to complete an emergency descent and autorotational landing from the maximum altitude for which certification is desired, but in no case is this total time to be less than 5 minutes.

e. A determination of the need to apply flight loads to the firewall for more than 5 minutes during the fire test (considering such factors as the maximum height above the terrain, maximum rate of descent, and reasonable time for recognizing a fire) is the responsibility of personnel engaged in the certification of the helicopter.

f. Insufficient protection to provide enough time for a controlled landing would represent an unsafe feature or characteristic (4.a.).

REVIEW CASE NO. 27 BOEING MODEL 727 - LONGITUDINAL CONTROL DURING FLAP RETRACTION (Issued 18 December 1963)

1. ORIGIN

a. The Western Region has determined that the wing flap retraction system is not in compliance with CAR 4b.131(c) which pertains to the rate of flap retraction. The regulation requires that the airplane be capable of flap retraction from $1.1V_{s1}$ speed at the maximum retracting rate and from any flap position with no loss of altitude and with not more than maximum continuous power applied simultaneously at the initiation of flap retraction.

b. Boeing has determined to their satisfaction that the $1.1V_{s1}$ speed is inappropriate for the Boeing 727 airplane and that the full intent of CAR 4b.131(c) is met by the Boeing 727 flap system. The Western Region stated that Boeing has requested that the Engineering and Manufacturing Division, FS-100, review the Western Region's interpretation of CAR 4b.131(c).

2. REGULATION AFFECTED

CAR 4b.131 Longitudinal Control

(c) It shall be possible without the use of exceptional piloting skill to prevent loss of altitude when wing flap retraction from any position is initiated during steady straight level flight at a speed equal to $1.1V_{s1}$ with simultaneous application of not more than maximum continuous power, with the landing gear extended, and with the airplane weight equal to the maximum sea level landing weight. (See also sec. 4b.323.)

3. HISTORY

a. December 10, 1962 - Aviation Week and Space Technology

The wing has trailing edge triple-slotted flaps running spanwise from the fuselage to the inboard aileron and from the inboard aileron to the outboard aileron. The leading edge of the wing carries Krueger flaps in three segments from the fuselage to a point about even with the outer end of the inboard trailing-edge flaps, and four-segment leading-edge slats from that point out to the wing tip.

The maximum lift coefficient of the wing is 40 percent higher than that of the Boeing 707 and Boeing 720 wings. The actual ratio of lift coefficient flaps down to flaps up is 1.7 for the Boeing 707 and Boeing 720, and 2.4 for the Boeing 727.

b. October 15, 1963 - Certification Status of Boeing Model 727 (by FS-45 and FS-160)

The "Longitudinal Control (Section 4b.131(c))" was listed as an item of noncompliance. The airplane stalls during the last portion of wing flap retraction at which time the leading edge high lift devices also retract.

c. November 5, 1963 - WE-210 letter to Boeing

The Western Region informed Boeing that the Boeing 727 airplane does not comply with CAR 4b.131(c) because it is not possible to apply maximum continuous power and retract flaps from any position when trimmed at $1.1V_s$, without loss of altitude. The Western Region also informed Boeing that there were no design features which provided equivalent safety.

d. November 11, 1963 - Boeing letter to WE-210

(1) The subject letter refers to a Boeing letter of November 8, 1963, describing an operating procedure for flap retraction that Boeing believes would make the airplane comply with the intent of CAR 4b.131(c).

(2) As a result of flight investigations of the Boeing procedure for flap retraction by the Western Region and Washington FAA personnel, Boeing was informed that while the procedure as outlined in the reference letter mentioned in (1) appeared to be entirely satisfactory from an operating standpoint, compliance with the full intent of the regulation was not achieved by manually delaying flap retraction through the use of gates. Boeing stated that it was concluded that $1.2V_s$ was more representative of the minimum starting speed for this test on the Boeing 727 airplane than $1.1V_s$.

(3) Boeing said that by commencing the test at $1.2V_s$ instead of $1.1V_s$ speed, the Boeing 727 can show compliance with CAR 4b.131(c) while continuously retracting the flaps from the full-down position to the flaps-up position. Boeing enclosed charts showing flight test results for two longitudinal controllability tests during flap retraction from $1.2V_s$ speed. The test data in Figure No. 1, show the airplane climbing at 240'/min. at the beginning of the test instead of being stabilized in steady horizontal flight as specified in CAM 4b.131-1(h).

(4) Boeing gave a number of arguments to support their contentions:

(a) The spread between power-off and power-on stall speeds for turbojet airplanes is much less than for piston-engine airplanes, and the takeoff speeds for turbojet airplanes are higher than for piston-engine airplanes in relation to their respective stall speeds.

(b) The approach and touchdown speeds for landing are higher for turbojet airplanes than for piston-engine airplanes.

(c) Boeing states that during flight at $1.1V_s$ speed, the following conditions exist and would ensure avoidance of such a speed:

1 High airplane attitude
2 Stickshaker stall warning
3 Sluggish control characteristics

(d) Boeing intends to retain the flap control gates and it will investigate the feasibility of using stronger gate detents.

e. November 22, 1963 - WE-210 memo to FS-100

(1) The subject memorandum transmitted a copy of Boeing's letter of November 11, 1963, for consideration with WE-210's request for a review case.

(2) WE-210 stated that the following landing test data point up the fact that a starting speed of $1.1V_{so}$ for the flap retraction test is an unrealistically low speed and is never used in operation:

(a)	Minimum touchdown speed	$1.22V_{so}$
(b)	Average touchdown speed	$1.27V_{so}$
(c)	Approach speed (approximate)	$1.3V_{so}$

f. <u>December 2 - 6, 1963 - FAA Blue Ribbon Team Report (Flight Test Portion, FS-160)</u>

(1) The report states that the Boeing 727 does not meet CAR 4b.131 (c) because of the large spread in flaps-down and flaps-up wing lift coefficients. The airplane cannot accelerate from a stabilized level flight speed of $1.1V_S$ with flaps down to a higher speed with flaps up in the minimum flap retraction time without losing altitude or stalling. The purpose of the regulation is to limit the rate of flap retraction so that the airplane will not stall under the most critical probable condition.

(2) The report states, generally, that on sweptwing turbojet airplanes, CAR 4T.112 allows the use of a minimum speed during the stall which is 8 to 10 percent below the 1g stall speed whereas there is very little corresponding speed spread on piston-engine airplanes. The $1.1V_S$ speed specified in CAR 4b. 131(c) based on $1.1V_S$ (minimum stall speed) is about equal to $1.0V_S$ (1g stall speed) on turbojet airplanes. The report states that the speed is far below any usable flight speed on The Boeing 727 airplane. All other certificated turbojet airplane types have met the requirements of CAR 4b.131(c).

(3) The report states that the $1.1V_S$ speed (based on minimum stall speed) is unrealistic for turbojet airplanes for two reasons:

(a) Propeller-driven airplanes are considerably above the 1g stall speed when the $1.1V_S$ speed (minimum stall speed) is used in complying with the flap retraction requirements of CAR 4b.131(c) due to the increased lift obtained from the propeller slipstream effect on the airplane wing.

(b) The power-off stall speed (minimum speed) on sweptback wing airplanes is far below any usable flight speed.

(4) The report states that if the flap retraction time were increased to permit the airplane to meet the requirement based on $1.1V_S$ speed (minimum speed in stall), there would be danger of exceeding the flap placard speeds in day-to-day operation before final flap retraction, and therefore, the $1.1V_S$ speed is unrealistic because at $1.1V_S$ (minimum speed in stall) the pilot is receiving stall warning and would not retract the flaps. Flight tests show that the requirements of CAR 4b. 131(c) can be met by the Boeing 727 airplane if the speed of $1.1V_S$ (minimum stall speed) were increased to $1.2V_S$.

The report states that photographic data for operational turbojet landings (reference Flight Standards Service Release No. 470) show that the lowest touchdown speed was $1.12V_s$ and the mean touchdown speed was $1.30V_s$. This shows that speeds of $1.1V_s$ are not used in flight.

(5) The report concludes that the Boeing 727 flap retraction appears satisfactory and is nearly ideal but does not meet the current CAR 4b.131(c) requirement.

g. <u>December 2, 1963 - WE-210 letter to Boeing</u>

(1) The subject letter refers to a Boeing letter of November 8, 1963, describing an operating procedure for flap retraction that Boeing believes would make the airplane comply with the intent of CAR 4b.131(c).

(2) WE-210 agreed with Boeing that the primary intent of CAR 4b. 131(c) is to assure that a missed approach or balked landing can be performed quickly and safely, and agreed that in the process of showing compliance the flaps must be retracted from the full landing position to the full-up position as rapidly as the controls permit.

(3) WE-210 disagreed with Boeing who proposed that limiting the amount of flap retraction which can be obtained with a single movement of the flap control lever to an intermediate setting by use of flap quadrant gates would meet the intent of CAR 4b.131(c).

(4) WE-210 informed Boeing that their proposal did not meet the intent of CAR 4b.131(c) and could not be approved.

4. FACTS IN THE CASE

a. The Boeing 727 airplane does not meet the wing flap retraction requirements of CAR 4b.131(c). The airplane stalls during the last portion of flap retraction at which time the leading-edge high lift devices also retract.

b. All turbojet transport airplanes previously certificated, have met the requirements of CAR 4b.131(c) starting from the stabilized level flight speed of $1.1V_s$.

c. The Boeing 727 airplane has a larger ratio of wing maximum lift coefficient, flaps down to flaps up than previously certificated turbojet airplanes.

d. The stall speed requirements of CAR 4T.112 allow the minimum speed obtained in the stalling maneuver to be called the stalling speed. This speed can be as much as 8 to 10 percent below the 1g stall speed on sweptwing turbojet airplanes.

e. The Boeing 727 airplane could meet all of the requirements of CAR 4b.131(c) if the flap down starting speed of $1.1V_s$ were changed to $1.2V_s$.

f. CAR 4b.131(c) is intended to limit the rate of flap retraction so that inadvertent, rapid raising of the flaps during a rejected landing would not cause a loss of altitude or other associated dangerous airplane characteristics such as stall.

g. The $1.2V_s$ speed is 7 knots above initial mechanical stall warning.

h. The operational Boeing 707 and Boeing 720 turbojet airplane landing data in Flight Standards Service Release No. 470 show that the lowest landing touchdown speed was 1.14 for these airplanes and the mean touchdown speed for all of the turbojet airplanes was $1.13V_s$.

i. The Boeing 727 landing distance tests for type certification resulted in the following:

(1)	Minimum touchdown speed	$1.22V_{so}$
(2)	Average touchdown speed	$1.27V_{so}$
(3)	Approach speed (approximate) at 50-foot height above threshold	$1.30V_{so}$

NOTE: The average touchdown speed is $.03V_{so}$ less than the approach speed. This allows a $03V_{so}$ speed differential for flaring the airplane.

j. The average corresponding data in Flight Standards Service Release No. 470 "Statistical Presentation of Operational Landing Parameters for Transport Jet Airplanes" are as follows:

(1)	Average touchdown speed	$1.30V_{so}$
(2)	Average approach speed at 20-foot height above threshold	$1.39V_{so}$

NOTE: The average touchdown speed is $09V_{so}$ less than the average approach speed. This speed spread is 200 percent greater than that for the Boeing 727 landing tests, and the average threshold height is 20 feet instead of the 50 feet used in the Boeing 727 tests.

k. During Boeing 727 flight at $1.1V_s$ the following occurs:

(1) The airplane attitude is very high.
(2) The stickshaker warns of impending stall.
(3) The airplane's response to control movements is noticeably sluggish.

l. The Boeing 727 airplane will have gated flap controls.

m. Boeing will investigate the feasibility of increasing the flap control spring force to provide a more deliberate action on the part of the pilot to pass the gate.

n. Figure No. 2 enclosed with Boeing's letter of November 11, 1963, to WE-210 shows flight test data for longitudinal control in compliance with CAR 4b.131(c) except that the starting speed is $1.2Vs_o$ instead of $1.1Vs_o$.

5. CONCLUSIONS

a. The Boeing 727 does not meet the longitudinal control requirements for wing flap retraction as specified in CAR 4b.131(c), the probable reason being that the starting speed of 1.1Vs for retracting the flaps is based on the minimum speed attained in the stall tests specified under CAR 4T.112. The stall speed obtained under CAR 4T.112 may be as much as 8 to 10 percent below the 1g stalling speed. This means that the $1.1V_s$ starting speed specified in CAR 4b.131(c) may be approximately the same as the 1g stalling speed.

b. The Boeing 727 approach and touchdown speeds relative to the stalling speeds for determining landing distances for type certification are considerably lower than those obtained from operational statistical data for transport turbojet airplanes, and the speed spread between approach and touchdown is much less than the average operational value for similar type airplanes.

c. The $1.1V_{s1}$ speed specified for the initial phase of the longitudinal control flight test under CAR 4b.131(c) is considered to be inappropriate and penalizing for application to the Boeing 727 for the following reasons:

(1) The $1.1V_{s1}$ (minimum stall speed) roughly corresponds to the $1.0V_{s1}$ (1g stall speed). The 1g stall speed is the minimum level flight speed that is possible with wing aerodynamic lift alone.

(2) The power-off stall speed of the Boeing 727 is the same, for all practical purposes, as the power-on stall speed. CAR 4b.131(c) was originally devised for straight-wing, propeller-driven airplanes which had much lower power-on stall speeds relative to the power-off stall speeds than the Boeing 727. For this reason, the $1.1V_{s1}$ speed was an operationally feasible go-around speed for propeller-driven airplanes. The $1.1V_{s1}$ speed is an impractically low speed for this purpose on the Boeing 727 airplane.

(a) Item 4j(1) shows that the minimum touchdown speed during the Boeing 727 landing distance type certification tests was $1.22V_{so}$ and the average touchdown speed was $1.27V_{so}$.

(b) Item 4j(2) shows that the average touchdown speed operationally for present day turbojet transport airplanes is $1.30V_{so}$.

(3) All of the touchdown speeds in (2) are well above the minimum touchdown speeds that would be considered remotely applicable to the Boeing 727 in unusual circumstances. The Boeing 727 operational pilot would not use a go-around speed as low as $1.1V_{so}$ because the noseup attitude would be too high, the stickshaker warning would be functioning, and the airplane's response to the control movements would be too sluggish. For these reasons, an initial go-around speed of $1.2V_{s1}$ is considered to be the minimum appropriate speed for any probable extremity.

d. Under the equivalent safety provisions of CAR 4b.10, the Boeing 727 would be considered to meet the intent of CAR 4b.131(c) flap retraction requirements if the requirements were met using an initial flap retraction speed of $1.2V_{s1}$ instead of $1.1V_{s1}$ for the reasons given in item "c", provided the following compensatory features are included in the airplane design, and flap flight procedures are included in the airplane flight manual:

(1) <u>Design Features</u>

(a) Flap control gates strategically located in relation to large and/or sudden changes in wing lift.

(b) Strong gate detents that will prevent the pilot from passing the gates inadvertently.

(2) <u>Airplane Flight Manual</u>

Suitable flap control operating instructions in relation to flap gates, airspeeds, etc., for safe airplane operation under all appropriate operating conditions.

e. A project will be initiated to consider revising the CAR 4b.131(c) requirements for the purpose of updating the requirements so that they will be appropriate for application to sweptwing, turbojet airplanes as well as straightwing, propeller-driven airplanes.

NOTE: The symbols V_s, V_{s1} and V_{so} for stalling speed are used somewhat interchangeably in this paper. The symbols do have the same significance when the wing flaps are in the landing position.

REVIEW CASE NO. 28. BOEING MODEL 727 - HORIZONTAL STABILIZER STOP SETTINGS (Issued 18 December 1963)

1. ORIGIN

a. The Western Region Boeing 727 Type Certification Board has questioned compliance with CAR 4b.312(b) with regard to the stabilizer stop positioning as proposed by The Boeing Company. A request was submitted by the Western Region for a case review to determine policy with regard to means of showing compliance with CARs 4b.140 and 4b.312(b).

b. The Boeing Company contends that safety margins are improved by increasing the stabilizer stop settings beyond the maximum required to allow increased travel for easier control of approach and landing trim. They have requested a review of the Model 727 Type Board decision with regard to maximum allowable stabilizer stop settings.

2. REGULATIONS AFFECTED

CAR 4b.142 Longitudinal Trim

The airplane shall maintain longitudinal trim under the following conditions:

(a) During a climb with maximum continuous power at a speed not in excess of $1.4V_{S1}$ with the landing gear retracted and the wing flaps both retracted and in the takeoff position,

(b) During a glide with power off at a speed not in excess of $1.4V_{S1}$ with the landing gear extended and the wing flaps both retracted and extended, with the forward center of gravity position approved for landing with the maximum landing weight, and with the most forward center gravity position approved for landing regardless of weight.

(c) During level flight at any speed from $1.4V_{S1}$ to (V_{MO}/M_{MO}), with the landing gear and wing flaps retracted, and from $1.4V_{S1}$ to V_{LE} with the landing gear extended.

CAR 4b.312(b)

When an adjustable stabilizer is used, stops shall be provided which will limit its travel, in the event of failure of the adjusting mechanism, to a range equal to the maximum required to trim the airplane in accordance with section 4b.140.

CAR 4b.112(c)(1)

The stall speeds defined in this section shall be the minimum speeds obtained in flight tests conducted in accordance with the procedure of subparagraphs (1) and (2) of this paragraph.

(1) From a speed sufficiently above the stalling speed to assure steady conditions, the elevator control shall be applied at a rate such that the airplane speed reduction does not exceed one mile per hour per second. This maneuver shall be performed with the airplane trimmed at a speed of $1.4V_{S1}$, except that airplanes utilizing adjustable stabilizers may be trimmed at a speed selected by the applicant but not less than $1.2V_{S1}$, nor greater than $1.4V_{S1}$.

3. BACKGROUND AND HISTORY

a. The Western Region memorandum to FS-100 dated November 15, 1963, outlines the events leading up to the request for the case review. This memorandum includes the conclusions reached by the Type Certification Board that all aircraft should have a trim capability to a minimum approach speed of $1.3V_{SO}$ since this is the minimum approach speed for landing distance determination; it is very desirable to have very low control forces during flare for a landing and for this reason a value less than $1.4V_{SO}$ trim speed capability should be allowed; and, the availability of the wheel trim switch on aircraft with controllable stabilizers provides the pilot with a ready means of trim adjustment for missed approaches and aborted landings. The latter conclusion is considered by them to be a compensating feature which was not envisioned when CAR 4b.312(b) was written.

b. The Western Region memorandum outlines a proposed flight test program designed to provide assurance that the intent of CAR 4b.312(b) is met at the extremes of the stabilizer travel; i.e., that no unsafe flight characteristics exist at these extremes. This program includes: a jammed stabilizer setting, a "go-around" and landing at the extreme stabilizer settings, and controllability checks with 3-second runaway stabilizer trim conditions.

c. The Boeing Company, by letter dated November 27, 1963, to the Western Region, presented their arguments in support of stabilizer stop positioning in excess of CAR 4b.312(b) conformity. They stated that the Model 727 has a greater trim capability than the maximum specified in the regulation to provide more desirable trim characteristics for approach and landing flare. They outlined their reasons for feeling that the system meets the intent of CAR 4b.312(b) as follows:

(1) For the extreme airplane nosedown case, any deliberate or inadvertent mistrim situation, where the stabilizer is driven to the mechanical stop, control is easily maintained by using up to 27 degrees of available up-elevator.

(2) For the extreme airplane noseup case, the electrical stop is positioned 0.3 degrees less than required to trim to $1.4V_{SO}$, and thus meets the limit requirement of CAR 4b.1312(b). The capability to trim at lower speeds is provided by the elevator. Additional trim can be obtained to a mechanical stop limit on the stabilizer by manual trim (use of trim wheel). This enables additional trim for the elevator boost-off landing where elevator trim is not available.

4. <u>FACTS IN THE CASE</u>

a. CAR 4b.142(b) establishes a speed for trim capability in a glide at forward c.g. at $1.4V_{S1}$. It should be noted this regulation does not prohibit designs capable of trimming to a slower speed.

b. An approach speed of $1.3V_{S1}$ is considered an operational approach speed for transport category aircraft and is used for performance determination.

c. CAR 4b.112(c)(1) permits trimming aircraft equipped with adjustable stabilizers to a minimum speed value of $1.2V_{S1}$ during stall speed determination. This implies acceptance of stabilizer stop settings to extreme positions beyond those specified in CARs 4b.312(b) and 4b.140.

d. CAR 4b.312(b) and CAR 4b.140, which encompasses 4b.142 are to safeguard against stabilizer stop setting positions to extremes where unsafe flight characteristics may exist. CAM 4b.140-1 interprets CAR 4b.140 as follows:

> It should be possible to trim the airplane completely for any flight condition which it is reasonable to assume will be maintained steadily for any appreciable time.

5. <u>ANALYSIS</u>

a. One of the basic considerations in establishing trim requirements (see CAM 4b.140-1) is to have trim capability at any normal operating speed and configuration. It is permissible to make approaches for landing at $1.3V_{S1}$ in determining runway lengths during official flight tests. Trim capability to this speed should be permitted.

b. Compliance with CAR 4b.312(b) could be substantiated by actual flight tests to demonstrate that, from any extreme stabilizer setting likely to be attained in operation, the aircraft could be shown to have no unsafe flight characteristics. It is further evident with careful review of CAR 4b.142 that speeds less than $1.4V_{S1}$ could be utilized in showing compliance in this instance. It is, therefore, reasonable to allow stabilizer stop settings established at extremes to permit trimming to zero elevator force within speed ranges and configurations utilized in normal operation, provided flight tests confirm compliance with the intended level of safety provided by CARs 4b.312(b) and 4b.140.

c. The flight test evaluation to determine that no unsafe flight characteristics exist should consider two types of failures which are likely to occur during operation of aircraft equipped with controllable stabilizers:

(1) One type failure, the jammed stabilizer, could be assumed to occur after reaching a limit of travel to an airplane nosedown extreme when loaded at aft c.g. while in high speed cruise flight. This type failure (jammed stabilizer) could also be assumed likely to occur after an approach for landing with a forward c.g. loading.

(2) The other type failure, a "runaway" condition must be assumed possible at any c.g. and weight within the normal operating range of the aircraft.

6. CONCLUSIONS

Compliance with the provisions of CARs 4b.312(b) and 4b.140 may be shown for aircraft equipped with controllable horizontal stabilizers when no unsafe flight conditions result during the following flight test demonstration:

a. Land the airplane safely with the stabilizer jammed in high speed cruise trim position with the aircraft loaded at the aft CG limit.

b. Execute a safe "go-around" and landing with the stabilizer jammed on the airplane noseup stop with the aircraft loaded at the forward CG limit.

c. For airplanes having stabilizer drive systems characteristically capable of a runaway, determine that it is possible to control 3-second stabilizer runaway from any trim condition within the approved CG/Weight range.

REVIEW CASE NO. 29. SOURCE OF POWER FOR EJECTOR/INSTRUMENT VACUUM SYSTEM (Issued 23 January 1964)

1. INTRODUCTION

Beech Aircraft Corporation desires approval for the use of a single Bendix 19E 17-2 ejector which is operated by air pressure sources from each of two turbo prop engines on their Model 65-90. The ejector converts engine air pressure to a low pressure air used for vacuum operated instruments. Beech Aircraft indicated that the high degree of reliability of the ejector with an air pressure source from each engine makes it equivalent to a two vacuum pump system on twin engine aircraft.

2. CHRONOLOGICAL HISTORY

a. Beech Aircraft Corporation in their letter dated August 7, 1963, to CE-210 requested the approval of a single ejector for vacuum supply on Model 65-98 (2 prop-jet engines) aircraft. They stated that the reliability of the ejector with a dual air pressure source is equivalent to a two vacuum pump system.

b. The Central Region in a letter dated August 14, 1963, requested FS-100 to provide a decision on whether or not the pressure air from the two engines could constitute the dual power sources as required in Part 3.668 of the Civil Air Regulations.

c. The Central Region, CE-210, forwarded memorandum on September 3, 1963, which indicated they will make whatever other determinations necessary for an equivalent safety evaluation.

d. FS-100 requested (CE-212) by TWX dated August 30 message 30 1830 drawings of installation of ejector system to be used by Beech Aircraft in their Model 65-90.

e. On November 7, 1963, FS-100 received copy of proposed schematic of ejector installation with details of Bendix ejector drawing Number 19E17-1-A.

3. FACTS IN THE CASE

a. Beech in their August 7, 1963, letter to the Central Region indicated their intent to install a single Bendix 19E17-2 ejector for instrument vacuum supply. They contend that this was considered to be equivalent in reliability to the presently used dual vacuum pump system. The question of compliance to Part 3.668 of the Civil Air Regulations arises where instruments have vacuum operated gyros. In this regard, Beech contends that in a pump vacuum system there is a portion of the system where there is no dual reliability. This they contend is due to the high degree of reliability of the check valves and plumbing involved. They feel that the use of the single ejector in a similarly designed system would provide a comparable high degree of reliability. Beech indicatesthat the ejector being of simple construction and high reliability and the fact that it is operated by two separate air pressure supplies would provide the same degree of reliability as the present vacuum pump systems on twin-engine aircraft.

b. The Central Region in their memorandum of September 3, 1963, indicated that they believed that one ejector is not absolutely equivalent to two. However, they stated further that the effective reliability or safety would not be increased significantly if two ejectors were required. This they contend is due to the high degree of reliability of the ejector and the system design.

c. The Central Region, on the other hand, in applying Part 3.668 of the Civil Air Regulations considers the ejector as the power source because the vacuum is provided by the ejector and not by the air source from the engines using as a basis the arguments expressed in Review Case No. 21.

d. The conclusion of Review Case No. 21 paragraph 4a states as follows:

> "Sources of Power" are intended to mean the sources required by the utilization devices (electric gyro indicators). To interpret the requirements otherwise would defeat the purpose of the requirement, that is, provide the availability of two independent sources of power to the utilization devices.

The Central Region contends a similar situation exists, that is, the power source is the ejector and not the engine air bleed.

e. The development of the current Part 3.668 of the Civil Air Regulations defines the "source of power" as follows:

> November 13, 1945, Part 03.5215 of the Civil Air Regulations. Gyroscopic Indicators (Air-Driven Type) All air-driven gyroscopic instruments installed in. . . . On multi-engine airplanes, the following detail requirement shall be applicable:

(a) Two sources actuated by separate means shall be of sufficient capability to operate, at the service ceiling of the airplane in normal cruising condition, all of the air-driven gyroscopic instruments with which the airplane is equipped.

(b) A suitable means shall be provided in the attendent installation where the source lines connect into a common line to select either suction air for the proper functioning of the instruments should failure of one source or breakage of one source line occur. When an automatic means to permit simultaneous air flow is provided in the system, a suitable method for maintaining suction shall be provided. In order to indicate which source of energy has failed, a visual means shall be provided to indicate this condition to the flight crew.

f. The above regulation (Part 03.5215 of the Civil Air Regulations) clearly permits an applicant to join the two sources of energy into a common system. Although the source of energy is not directly usable it does provide the energy by which the ejector provides suction to the system. A similar intent was expressed in the November 1, 1949, revision which became the first Part 3.668 of the Civil Air Regulations requirement. The current Part 3.668 of the Civil Air Regulations is worded exactly as the October 1, 1959, revision which is stated as follows:

All gyroscopic instruments installed in. . . . In addition the following provisions shall be applicable to multiengine airplanes:

(a) There shall be provided at least two independent sources of power, a manual or an automatic means for selecting the power source, and a means for indicating the adequacy of the power being supplied to each source.

NOTE: Power sources are not considered independent if both sources are driven by the same engine.

(b) The installation and power supply systems shall be such that failure of one instrument or of the energy supply from one source will not interfere with the proper supply of energy to the remaining instruments or from the other source.

4. CONCLUSIONS

a. In view of the similarity of the present and the November 1, 1945, requirements the intent of the current regulation as applicable to the "source of power" is the same as the original intent. In that regard the only logical source of power or energy is the engine; the conversion of that energy by a simple device without moving parts is a secondary function.

b. The ejector because of its high reliability (no moving parts, mechanically strong and free from any tendency to become clogged) is not considered a source of power.

c. The Beech Aircraft Corporation installation on Model 65-90 of Bendix 19E17-2 ejector does not constitute the power source and therefore is not subject to requirements of Part 3.668 of the Civil Air Regulations as a dual power source.

REVIEW CASE NO. 30 BOEING-VERTOL'S PROPOSED 4000 FEET EXTRAPOLATION METHOD FOR CAR 7 CATEGORY A H-V TEST DATA (Issued 12 March 1964)

1. ORIGIN

a. The Boeing Company, Vertol Division, has made a written request to the Eastern Region for a Review Case decision concerning Vertol's contention that the present Federal Aviation Agency CAR 7 test procedure regarding altitude extrapolation of limiting height and speeds for safe landing following power failure (H-V-D) is not appropriate to the V107-II when operating under Category "A". (The Eastern Region has requested the Washington Office to prepare a review case).

b. The test procedure applied to date under CAR 7.100 has limited the permissible extent of extrapolation of H-V data to no more than ± 2000 feet H_d from the test altitude.

c. Vertol contends that the soundness of their extrapolation method should permit them to extrapolate ± 4000-feet H_d from the test altitude.

2. REGULATIONS AFFECTED

CAR 7.100 Proof of Compliance

(a) Compliance with the requirements prescribed in this subpart shall be established by flight or other tests conducted upon a rotorcraft of the type for which a certificate of airworthiness is sought or by calculations based on such tests, provided that the results obtained by calculations are equivalent in accuracy to the results of direct testing.

CAR 7.111 Limiting Height and Speeds for Safe Landing Following Power Failure

(a) Category A. If a range of heights exists at any speed, including zero, within which it is not possible to make a safe landing when the critical engine is suddenly made inoperative with takeoff power on the operating engine(s), the range of heights and its variation with forward speed shall be established (see secs. 7.715 and 7.741(f)).

CAR 7.112 Takeoff General

(a) Category A. The takeoff performance shall be determined and scheduled in such a manner that, in the event of one engine becoming inoperative at any instant after the start of takeoff, it shall be possible for the rotorcraft either to return to and stop safely on the takeoff area, or to continue the takeoff climbout, and attain a rotorcraft configuration and airspeed at which compliance with the climb requirement of section 7.115(a)(2) is met.

3. TEST PROCEDURE

FAA test procedure which has been used on all CAR 7 helicopters to date is as follows:

"The height-velocity diagram will be approved at the test density altitude for which tests are conducted. Additional testing will be required beyond this altitude except that altitude extrapolation will be permitted to ± 2000-feet of density altitude, if an acceptable analytical method, to justify extrapolation, is presented to the FAA by an applicant.-----" (see par 4d)

4. HISTORY

a. The limiting heights and speeds for safe landing following power failure of helicopters has long been a concern of the Federal Aviation Agency.

(1) A major part of this concern has been the unknown variables that may or may not exist at the higher operating altitudes following a sudden engine failure during takeoff.

(2) The manufacturer has indicated that additional economic burden is placed on him if helicopter altitude tests are required. He feels this altitude information can be reasonably obtained through extrapolation of sea level test data.

b. Altitude testing was not conducted on the early helicopters which were certificated by the 1951 requirements for any or all of the following reasons:

(1) The helicopter of that vintage did not possess performance capability at higher altitude.

(2) No knowledge existed to indicate that the H-V diagram may deteriorate with altitude.

(3) A contention by the operators and manufacturers that the major part of any helicopter operation would be conducted at sea level.

c. As concern about altitude height-velocity capability increased, due to the installation of supercharged and turbine-powered engines (which gave higher altitude operation performance) several steps were taken:

(1) Tests were run at altitude on several FAA certification programs. (i.e., Bell 47G-3, Hiller UH-12, Bell 204, Vertol 44 S-62, etc.)

(2) Research tests were run at altitude by FAA ADS. (These tests showed an approximate 2 1/2 percent weight reduction per 1000 feet was necessary to retain the same H-V diagram established at sea level).

(3) Special regulations were issued by the FAA for LOH to require altitude testing of H-V diagram.

(4) FAA established test procedure on CAR 7 helicopters to allow 2000 feet extrapolation only. (see par 3)

d. <u>December 17, 1958 - FAA test procedure established on height-velocity diagram extrapolation</u>

The current test procedure (see par 3) was first established by the special condition on the S-62 helicopter. This special condition was forwarded to Sikorsky by letter from EA-210 (formerly FS-1100) dated December 17, 1958. As this procedure needed further clarification a letter was forwarded to Sikorsky from EA-210 (formerly FS-1100) dated December 21, 1959. This letter contained the detailed procedure outlined in par 3. On July 9, 1959, at a Preliminary Type Board Meeting held on the Vertol 107-II, the test procedure outlined in par 3 was given to Vertol and was made a part of the Preliminary Type Certification Board Minutes, copies of which were forwarded to Vertol.

e. This test procedure (see par 4d) has since been applied to all helicopters certificated under CAR 7.

f. The reason FAA established the 2000-foot density altitude extrapolation limitation was because there has been a lack of substantiating test data to support the manufacturer's contention that they could predict and/or extrapolate the shape of the H-V diagram beyond 2000 feet H_d. None of the previous altitude tests were conducted under sufficiently controlled conditions, (i.e., variable pilot skill, variable margins of conservatism, etc.) to verify any extrapolation method.

g. February 1, 1961 - Vertol letter to the Eastern Region

(1) Vertol contended that the soundness of their proposed extrapolation method should permit them to extrapolate to ± 4000-feet H_d from the test conditions.

(2) The Eastern Region believes that the Vertol proposal offers no proof that their engineering analysis would provide results equivalent in accuracy to the results of direct flight testing as required by CAR 7.100.

h. March 7, 1961 - EA-210 letter to Vertol

The Eastern Region replied to Vertol's inquiry dated 2/1/61 for information relative to FAA test procedure on temperature-altitude testing. The region stated the present FAA test procedure provides for approval of the height-velocity diagram conducted at the test density altitude. Additional testing is required if H-V approval is desired beyond this altitude, however, altitude extrapolation will be permitted to ± 2000 feet of density altitude, if an acceptable analytical method to justify extrapolation is presented to the FAA by Vertol. (see par 3a)

i. May 27, 1963 - Vertol letter and test data to EA-210

Vertol proposal (which included a letter and test data) was sent to the Eastern Region requesting increased altitude approval for CAR 7, Category A and Category B operations, based upon their 4000-feet extrapolation proposal.

j. June 12, 1963 - EA-210 letter to Vertol

Eastern Region's reply informed Vertol that the data submitted with their proposal of May 27, 1961, (to substantiate Vertol's request) was not considered sufficient to deviate from the existing FAA policy. (see par 3)

k. July 11, 1963 - Vertol letter and report to EA-210

Vertol report was sent in reply to EA-210 letter dated 6/12/63. Vertol felt that this report (Aero. Investigation III-237, effect of increased ambient temperature on Category "A" takeoff and landings, dated 7/11/63) answered previous EA-210 objections.

l. August 23, 1963 - EA-210 letter to Vertol

The Eastern Region letter informed Vertol; "Despite the fact that the Vertol Report Aero. Investigation III-237, is fairly rigorous and appears to be reasonable in approach, this office does not believe that CAR 7.100 could be satisfied at this time by other than direct flight test data."

The region further stated that if additional FAA-Vertol flight testing confirmed the Vertol extrapolation method and provided acceptable results (i.e., 4000-feet density altitude from test conditions) that the region would be glad to consider revising their present test procedure at that time.

m. October 7, 1963 - FAA-Vertol Meeting - Washington

A FAA-Vertol meeting was held in Washington in which Vertol made a presentation of their 4000-feet altitude extrapolation method. The following conclusions and actions resulted from this meeting:

(1) Vertol will submit additional substantiating data.

(2) The Eastern Region may make formal request for a review case or make a recommendation for a test procedure change.

(3) Vertol was informed by FAA-Washington that until further experience or test data were obtained the FAA would continue to require testing beyond certain limits and that the present FAA test procedure on extrapolation would be maintained. It was noted, however, that consideration might be given to a test procedure similar to that used on SR-422 transports, where certain additional conservatisms would be added to the manufacturer's extrapolation method for extrapolation beyond a given altitude.

n. October 9, 1963 - Vertol letter to EA-210

The Vertol letter references the EA-210 letter to Vertol dated 8/23/63, the Washington meeting of 10/7/63, and asks that the Eastern Region reconsider the Vertol request for extrapolation limits to 4000-feet for the V107-II program.

o. November 22, 1963 - Vertol letter to EA-216

The Vertol letter to the Eastern Region this date requested a review case relative to their 4000-foot altitude extrapolation method proposal. About this time Vertol hand carried copies of their revised report with further substantiation of their extrapolation method.

p. December 13, 1963 - EA-200 letter and other data to FS-100

Eastern Region forwarded a letter to FS-100, relative to facts needed to reopen Vertol review case. This letter recommends a review case and notes that the Eastern Region is in disagreement with the Vertol proposal.

5. FACTS IN THE CASE

a. The present FAA test procedure which permits a ± 2000-foot density altitude extrapolation limit from the test condition has been used for some years.

b. The present regulations (CAR 7.111(a) and CAR 7.112(a)) call for determination of autorotative landing characteristics without specific reference to either sea level or altitude.

c. The recent FAA-ADS project (343-10V) (see par 4c(2)) has shown that altitude has a significant deteriorating effect on the autorotative characteristics.

d. Altitude tests have been deemed necessary and required on three CAR 6 helicopters (i.e., HJ-1, V-44, and the LOHs), and three CAR 7 helicopters (i.e., S-62, V107 and the Bell 204. None of these altitude tests were conducted under sufficiently controlled conditions (i.e., variable pilot skill, variable margins of conservatism, etc.) to verify any extrapolation method. (At that time the manufacturers' objective was primarily to obtain altitude approval).

e. For extrapolating the height-velocity diagram to altitude Vertol uses the power deficiency parameter (PDP) index method. (see par 4k) If the weight reduction at altitude follows this parameter, Vertol contends that the rate of descent will decrease with altitude and the collective pitch position in descent will be lower. These two factors they contend will make the landing easier and safer. Where these conditions exist Vertol concludes that the PDP is, in fact, conservative. There is some question, however, whether these conditions hold when there is no power deterioration at altitude. (A portion of the PDP calls for a weight reduction due to a reduction in power available, the other portion calls for a weight reduction due to an increase in power required to hover. (i.e., aerodynamic deterioration)

f. The portion of the PDP formula which relates to "aerodynamic deterioration" (i.e., no power change) gives approximately only one percent weight reduction per 1000 feet of density altitude. This is not consistent with the FAA research tests run on a single engine rotor Bell which came to about 2 1/2 percent. Admittedly, the difference in rotor systems and/or number of powerplants may justify some difference in aerodynamic deterioration, however, the magnitude of the difference does create some doubt as to the validity of the Vertol method over relatively wide ranges of altitude.

6. CONCLUSIONS. The method proposed by Boeing-Vertol for extrapolation of the height-velocity diagram on the V107-II should not be permitted beyond the 2000-foot point at this time, until the validity of this extrapolation method is verified by altitude testing.

REVIEW CASE NO. 31 USE OF COMMUNICATIONS/NAVIGATION AND AUTOPILOT/ COUPLER EQUIPMENT IN PART 3 AIRCRAFT IFR OPERATIONS (Issued 26 March 1964)

1. INTRODUCTION

During recent evaluations of communications and navigation equipment for Mitchell Industries radio coupler, Model AK-123, and Mooney Model M20C and M20E airplanes, the Southwest Region imposed certain restrictions on the use of equipment during IFR operations. The restriction was based on the fact that when the communication transmitter is keyed, the navigation signal to the airplane is disrupted. The Aircraft Owners and Pilots Association and the Aerospace Industries Association of America objected to the conclusions reached by the Southwest Region, who requested that a review case be written on the subject.

2. CHRONOLOGICAL HISTORY

a. SW-200 memorandum to FS-1 dated September 19, 1963, regarding

(1) STC application by Mitchell Industries, Inc., Mineral Wells, Texas, for approval of Mitchell Autopilot Coupler System AK-123 on August 1, 1963; Type Inspection Authorization prepared August 5, 1963, and Federal Aviation Agency flight tests conducted on PA-24-250 airplane on August 8, 1963. Mitchell representatives were informed that use of the coupler would be prohibited during IFR flight.

(2) Form FAA 1600 submitted by Mooney Aircraft, Inc. for installation of Motorola Model M400 avionic equipment on Mooney Model M20E; Mooney Model M20C presented for IFR approval with ARC 513A avionic package installed. SW-212 letter of September 11, 1963, to Mooney advised that communication and navigation equipment with the characteristic of disrupting the navigation signal during periods of transmitter use is unacceptable for use in the M20C and/or M20E during IFR conditions.

b. Conference on October 2, 1963, with personnel from FS-100, FS-200, FS-300, and FS-400 attending. Needed actions were resolution of the contents of SW-200 memorandum to FS-1 dated September 19, 1963 (reference paragraph 2.a) and publishing of an advisory circular on the noted subject.

c. SW-200 memorandum to FS-1 dated October 17, 1963, noting a modified position on the Mooney Model M20E for use in IFR conditions of the communication transmitter which disrupts the navigation signal. The prohibition noted in paragraph 2.a(2) was changed to a cautionary note in item 16 of the M20E AFM.

d. SW-212 memorandum to FS-100 dated November 1, 1963, regarding comments on a briefing paper prepared by FS-120 and a draft of an advisory circular on the subject (reference paragraph 2.b.).

e. FS-968 memorandum to FS-100 dated December 5, 1963, noting conclusions reached as a result of an evaluation of the subject. Among others, the memorandum stated "one and one-half radio system satisfactory for IFR flight so long as pilot is aware of its characteristic wherein the navigation signal is interrupted while transmitting."

3. SUMMARY OF THE FACTS

a. Some types of combination communication/navigation radio equipment used in Part 3 aircraft are designed to utilize common circuitry and/or other components. As a result, when the communication transmitter is keyed the navigation signal to the aircraft is disrupted. This results in the course needle deflecting from its normal operation location; if an autopilot is coupled to the navigation receiver, it may return the aircraft to a "wings level" attitude or attempt to slowly follow the course needle, depending on specific type of installation.

b. SW-200 final action on the Mitchell Autopilot System was as follows:

(1) Multiple radio installations whereby only one transmitter affects the Navigation Receiver Signal on which coupler is connected, placard to read - USE OF THIS TRANSMITTER PROHIBITED DURING IFR FLIGHT WHEN COUPLER IS IN OMNI OR LOC POSITION.

(2) Single radio installations whereby the only available transmitter affects the only available Navigation Receiver Signal or Multiple radio whereby all transmitters affect the Navigation Signal - COUPLER OPERATION PROHIBITED DURING IFR FLIGHT.

c. SW-200 final action on the Mooney Models M20C and M20E was as follows:

(1) Note in item 16 of the Mooney AFM which reads "Caution should be exercised when conducting approaches and departures under IFR operations when communications equipment installed interrupts the navigation signal during transmissions."

(2) Addition to the limitation section of the AFM supplement for the ARC AF 512-B automatic pilot installation on the Mooney aircraft which read "AF 512-B Flight Controller cannot be used for IFR Flight when the Controller is coupled to navigation equipment which is disrupted by communication transmitter operation."

d. CAR 43.30 (FAR 91.33) specifies the instruments and equipment required for the particular category of operation specified (type of operation). Autopilots and couplers are <u>not</u> required for <u>any</u> type of operation.

e. CAR 3.655 specifies the required basic equipment for type and airworthiness certification of an airplane. Autopilots and couplers are not required.

f. CAR 43.30(c)(2) (FAR 91.33(d)(2)) requires two-way radio communications system and navigational equipment appropriate to the ground facilities to be used for instrument flight rules operation.

g. CAR 3.721 requires that radio equipment and installation be free from hazard in themselves, in method of operation, and in effects on other components of the airplane. CAR 3.652 requires that each item of equipment essential to safe operation of the airplane shall perform adequately the functions for which it is to be used, shall function properly when installed, and shall be adequately labeled as to its identification, function, operational limitations, or any combination of these, whichever is applicable.

h. There is no known adverse service experience due to use of the type equipment described in paragraph 3.2. The type of equipment in which the course needle deflects when the transmitter is keyed has been in use for over ten years. Installations of this type have been approved by both regional and field personnel. There is no known case of where users of this type equipment have officially complained that it is unsatisfactory or unsafe; quite the contrary, some of them have stated that it is definitely not a problem.

i. The Southwest Region has imposed certain restrictions on the use of communication and navigation equipment during IFR operations. The conclusions of this review case do not support this action. Although the Southwest Region has deemed that such restrictions are necessary, a review by the Washington Office has not concurred with the Southwest Region.

4. CONCLUSIONS

a. Use of an automatic pilot and coupler should not be prohibited on the basis that the navigation signal is disrupted when the communication transmitter is keyed. No regulatory basis has been found to support a prohibition of this kind, since this system has not been found unsafe.

b. Use of combination communication/navigation equipment in which the course needle is deflected away from its normal operating position when the communication transmitter is keyed should not be prohibited because of this characteristic. No regulatory basis has been found to support a prohibition of this kind, since such equipment has not been found unsafe.

c. Appropriate aircraft documents should contain sufficient information to inform operators of all characteristics of combination communication/navigation equipment and installations.

REVIEW CASE NO. 32 LOCKHEED MODEL 300 (C-141A) REVERSE THRUST PERFORMANCE CREDIT (Issued 1 April 1964)

1. INTRODUCTION

Lockheed-Georgia Company has requested approval of their proposal for the use of reverse thrust in the determination of type certification accelerate-stop and landing distances for the Lockheed Model 300 (C-141A) airplane. The Southern Region does not concur with their proposal, but concurs with the findings of Review Case No. 15 dated September 19, 1962, which denied reverse thrust performance credit to Boeing. A decision in this matter has been requested.

2. HISTORY

a. September 19, 1962 Review Case No. 15, request for reverse thrust performance credit in the determination of landing distances for the Boeing 707-300B airplane when the antiskid system is inoperative, contains background and history of the reverse thrust performance credit problem. The memorandum transmitting Review Case No. 15 specifically states that no favorable consideration can be given until further notice to any request for reverse thrust performance credit for accelerate-stop or landing distances under the provisions of SR-422B, sections 4T.115(b) and 4T.122(f).

b. July 15, 1963 Notice of Proposed Rulemaking 63-28, Special Operating Limitations for Turbojet Transport Category Airplanes proposed increasing the accelerate-stop distance for all runway conditions and the required operational field length for wet runway operation.

c. July 23, 1963 Southern Region's memorandum to FS-1 requested Review Case procedure be applied to Lockheed's proposal for the use of reverse thrust in determining accelerate-stop and landing distances.

3. DISCUSSION

a. Review Case No. 15 consistently emphasizes that the landing field lengths, which were determined without the use of reverse thrust, may be inadequate when operating under adverse runway conditions.

b. NPRM 63-28 proposed increasing the accelerate-stop distances and landing field lengths for present turbojet airplanes. Reverse thrust performance credit was not included in the determination of these distances. Comments on NPRM 63-28 are being evaluated in an effort to arrive at an equitable requirement for all turbojet aircraft on a retroactive basis. Consideration is being given to revising the accelerate-stop and landing distance requirements to include operational variables occurring in airline service. Final action may result in the promulgation of a retroactive requirement which may or may not allow reverse thrust performance credit.

c. As turbojet operational experience was gained, especially during takeoffs and landings under adverse runway conditions, it became apparent that operations under these conditions were marginal and that reverse thrust is needed as a standby reserve. The FAA believes that the present field sizes for takeoffs and landings are needed for normal operations, taking into account the availability of reverse thrust and that greater field sizes are necessary for operations under adverse conditions. Reverse thrust performance credit for accelerate-stop and landing distances would tend to decrease the present field sizes which we believe are necessary to assure an adequate level of safety. As a result of this FAA policy, there are no turbojet airplanes at the present time which have been type certificated allowing performance credit for the use of reverse thrust. The instruction in the FS-1 memorandum of transmittal to Review Case No. 15, which specified that no reverse thrust performance credit would be allowed until further notice, is still applicable.

4. CONCLUSION

The Lockheed-Georgia Company request for approval of their proposal for reverse thrust performance credit in determining accelerate-stop and landing distances for their Model 300 (C-141A) airplane is denied. If a retroactive requirement allowing reverse thrust performance credit is published in the future, performance credit in accordance with criteria contained therein shall be applicable.

REVIEW CASE NO. 33 PIPER AIRCRAFT CORPORATION PA-28 POWER ADEQUACY INDICATION FOR ELECTRIC TURN AND BANK INSTRUMENT (Issued 29 April 1964)

1. INTRODUCTION

Piper Aircraft Corporation telephoned SO-EMDO-42 on October 2, 1963, to discuss power adequacy indication for an electric turn and bank instrument in the PA-28 series aircraft. They felt, after reviewing Sections 3.668, Gyroscopic Indicators, and 3.687, Electric Power System Instruments, of the Civil Air Regulations that an ammeter or voltmeter connected to the electrical bus satisfied the pertinent requirements of these CARs. Southern Region personnel did not concur. They informed Piper Aircraft that it was Southern Region policy to accept a voltmeter connected between the instrument and the circuit breaker (or fuse) directly adjacent to the instrument on the instrument panel as compliance with Section 3.668 of the Civil Air Regulations. They also stated that other reliable and readily interpreted means are acceptable. In addition, they did not consider Section 3.687 of the CARs pertinent to the subject. The purpose of this review case is to determine (as requested by the Southern Region) the intent of that phrase of Section 3.668 of the Civil Air Regulations which states "Means shall be available for indicating the adequacy of the power being supplied to the instruments."

2. CHRONOLOGICAL HISTORY

a. Telephone conversation October 2 and 3, 1962, between Southern Region and Piper Aircraft on the subject matter, with the positions taken by the respective parties as noted in paragraph 1.

b. SO-EMDO-42 letter to Piper Aircraft dated October 11, 1963, confirming the comments stated during the telphone conversation on October 2 and 3, 1963, and suggesting that Piper submit a proposal which would be given an evaluation.

c. FS-120 briefing paper dated October 23, 1963, stating the essence of the telephone conversations of October 2 and 3, 1963, (paragraph 2.a), discussing the history of CAR 3.668, and noting that electric gyroscopic instruments have been installed and approved in Part 3 aircraft utilizing only bus connected volt and ampere meters and no other additional power indicating devices at the instrument.

d. Aerospace Industries Association of American, Inc., letter to FS-100 dated November 4, 1963, stating a request for alleviation of economic hardships that are being placed on Part 3 aircraft manufacturers by current interpretations of CARs 3.668 and 3.687 by certain Agency Regional Offices.

e. SO-210 memo to FS-100 dated November 14, 1963, commenting on the AIA letter to FS-100 dated November 4, 1963, and restating their version of events which contributed to the case stated by AIA (paragraph 2.d.) In addition, SO-210 welcomed a review of their application of the subject requirements.

3. FACTS IN THE CASE

a. The first time that a reference to power supply for gyroscopic instruments was made in the CARs was in CAR 04.5805 dated November 1, 1937. It states "All gyroscopic instruments shall derive their energy from engine-driven pumps or from auxiliary power units." No statement relating to the means of indicating the adequacy of power being supplied was made, however.

b. CAR 03.5215, Gyroscopic Indicators (Air-Driven Type) was promulgated November 13, 1945, as a section of Part 03, which was introduced at this time. By the title, this requirement was limited to air-driven gyroscopic instruments. No requirement for electrically driven gyros was included in CAR, Part 03 of this date. Again no mention was made of the means of indicating the adequacy of power being supplied.

c. CAR, Part 3, November 1, 1949, contained Section 3.668 titled "Gyroscopic indicators (air-driven type)" and states the following:

> All air-driven gyroscopic instruments installed in airplanes which are certificated for instrument flight operations shall derive their energy from a reliable suction source of sufficient capacity to maintain their required accuracy at all speeds above the best rate-of-climb speed . . . . . On multiengine airplanes . . . . . Two sources shall be provided . . . . . In order to indicate which source of energy has failed, a visual means shall be provided to indicate this condition to the flight crew.

d. Amendment 3-7, effective March 5, 1952, removed "Air-driven type" from the title of section and the text changed to:

> All gyroscopic instruments installed in airplanes intended for operation under instrument flight rules shall derive their energy from a power source of sufficient capacity to maintain their required accuracy at all airplane speeds above the best rate of climb speed. They shall be installed to

preclude malfunctioning due to rain, oil, and other detrimental elements. Means shall be provided for indicating the adequacy of the power being supplied to each of the instruments....."

The following should be noted:

(1) Requirement was then limited to IFR airplanes.

(2) It is apparent that air-driven gyros are still being considered where the instrument accuracy and functioning would be affected by airplane speed, rain, oil, or "other detrimental elements."

(3) Electrically driven gyros are not excluded by the precise language of this rule. This could have required a separate power adequacy indication for "each of the instruments;" however, it was not so interpreted while this version was in effect.

e. Amendment 3-3 to CARs, Part 3, became effective May 17, 1958, and revised (among other paragraphs) paragraph 3.668 by deleting the words "each of" in the third sentence of the introductory paragraph. The preamble to this amendment states, "It has been found that the provision of paragraph 3.668 requiring a means for indicating the adequacy of power being supplied to each gyroscopic instrument unnecessarily complicates the airplane's vacuum system without giving an indication of all possible instrument failures, such as the clogging of integral filters. For this reason, paragraph 3.668 is being amended to require a power failure indicator only for the power source. In addition, a new paragraph 3.687 is being included which requires electric power system instruments." (Underlining added for emphasis). Reference to "vacuum system and integral filters" confirms that the intended application was and is to air driven gyros. Although the preamble to amendment 3-3 does not so state, the reference to IFR was also dropped. The revision to 3.668 was made at the request of the Aircraft Industries Association. Prior to its introduction at the 1957 Annual Airworthiness Review, a test program was conducted by CAA at Washington National Airport, Hangar 6, July 18 and 30, 1957, with three air driven instruments. The findings were summarized in Mr. A. A. Vollmecke's letter to AIA dated September 6, 1957, in that the words "each of" could be deleted without significant effect on safety. No adverse service experience could be found. There was no mention of the omission of reference to IFR operation in the minutes of the annual airworthiness review; however, gyroscopic attitude indicators are required by CAR, Part 43 (FAR 91) for IFR, not for VFR operation.

f. Section 3.687, Electric Power System Instruments reads as follows:

> Means shall be provided to indicate to appropriate crew members those electric power system quantities which are essential for the safe operation of the system.
>
> NOTE: For direct current systems an ammeter which can be switched into each generator feeder would be acceptable. When only one generator is installed, the ammeter may be in the battery feeder.

This is apparently intended to apply to instruments showing "power system quantities;" that is, power being supplied from the generator and/or battery, rather than adequacy of power being supplied to any single instrument.

g. There have been no changes in either Part 3.668 or 3.687 subsequent to those noted in paragraph 3.d which are relative to the phrases of these regulations under discussion.

h. There has never been a CAR 3 requirement that the means for indicating the adequacy of the power being supplied to the instruments be located immediately adjacent to an instrument on the instrument panel; similarly such means have never been required to be connected between an instrument and a circuit breaker or fuse in the line for the electric source to the instrument.

i. There has been an undetermined number of CAR 3 aircraft approved with electric turn and bank instruments with no meters or lights between the circuit breaker (or fuse) and the instruments. Some have been approved as alterations and some as part of the original type design. There is no known adverse service experience due to installations of this type. A review of representative service manuals indicates that electric turn and bank instruments, with nothing between them and the bus, were approved for: Cessna 150, 172, 175, 180, 182, and 185 Series, Beech 33, 35, and 95 Series and Aero Commander 520.

j. Technical Standard Order No. TSO-C3a, Turn and Bank Indicator, effective July 1, 1948, incorporates Society of Automotive Engineers Aeronautical Standard AS-395, issued July 1, 1947. This TSO covers three types of instruments: air driven, D-C operated, and A-C operated. Electrically driven gyroscopic instruments were obviously available at that time, but not frequently installed on small aircraft. This TSO does not require an indication of power failure in the instrument. Similar gyroscopic instruments per TSO's C4c and C5c are required to have power failure indicators to be compatible with CAR 4b.612(e)(1).

For the purpose of this analysis of the facts in the case, it should be noted that CAR 3 requires neither that instruments be TSO approved, nor equipped with a power failure indicator. The removal of the words "each of" (per amendment 3-3, May 17, 1958) eliminates the possibility of interpreting 3.668 as requiring power failure indication for individual instruments.

4. CONCLUSIONS

a. We conclude that, although similar regulations prior to the present Section 3.668 were intended principally to be applicable to vacuum operated instruments only, the present wording of the regulation does not exclude application to electrically powered instruments, therefore, they must be found to comply with this section.

b. We conclude that the means for indicating the adequacy of power being supplied to each of the instruments is not necessary for the safe operation of the general type aircraft with vacuum operated instruments, and it is not necessary with electric instruments in a system such as the Model PA-28.

c. We conclude that, on the Model PA-28, when means are provided per CAR 3.687 to indicate electric quantities essential for safe operation of the (total electrical) system, the requirements of Section 3.668 are met with respect to "the adequacy of the power being supplied to the instruments," when those instruments are electrical.

d. We conclude that Section 3.668 as applied to the Model PA-28 would not require a power failure warning for each individual instrument.

e. We conclude that the Piper Aircraft Company proposal to use a voltmeter or an ammeter on the electrical bus in their Model PA-28 airplane complies with Sections 3.668 and 3.687, provided such meters are visible to the pilot.

REVIEW CASE NO. 34 POWERPLANT INSTRUMENTS (Issued 7 May 1964)

1. INTRODUCTION

The Hughes Tool Company has requested a review case through our Western Regional Office concerning powerplant instrument requirements in accordance with Section 6.604(c), (h), and (j), Civil Air Regulations. This regulation requires fuel pressure, oil pressure, and oil temperature indicators. Hughes proposed to meet these requirements by replacing gage type instruments with warning lights. Hughes is convinced that a system of warning lights is superior to a gage.

2. CHRONOLOGICAL HISTORY

a. Hughes Tool Company advised WE-210 by letter dated August 23, 1963, that it believed that compliance with Section 6.604(h), Civil Air Regulations, would be satisfied by two warning lights. One light would operate at minimum pressure satisfactory for idle and the other at minimum pressure satisfactory for full speed operation.

b. On September 26, 1963, during a visit by Mr. Jack Sain, WE-214, to this office, the request by Hughes Tool Company in a. above was discussed with Messrs. Auburn, Haddad, and Osborne, FS-140. Mr. Sain was advised that this arrangement did not meet the intent of Sections 6.604(h) and 6.734, Civil Air Regulations.

c. Western Regional Office letter of October 16, 1963, to Hughes Tool Company, advised that lights do not meet the intent of the regulation and do not offer any significant mitigating factors which could be considered to provide equivalent safety to the indicator required in the regulation.

d. Hughes Tool Company, by letter dated January 17, 1964, requested the Western Regional Office to accept warning lights in lieu of gages for compliance with Section 6.604(c), (h), and (j), CAR.

e. FS-100 memorandum of January 23, 1964, confirmed the information given to Mr. Sain on September 26, 1963, (see b.).

f. Hughes Tool Company, by letter dated February 17, 1964, again requested consideration of its request for a review case.

g. WE-210 memorandum of February 27, 1964, transmitted Hughes Tool Company request for a review case.

3. FACTS IN THE CASE

a. Hughes has expressed its philosophy that wherever and whenever possible, gage type instruments should be omitted or replaced by warning devices. Hughes further "feels that this philosophy is in the public interest since it allows the pilot to apply maximum concentration to the task of watching his environment and minimizes the possibility of accidents due to unnecessary pilot preoccupation with panel instruments."

b. The following were listed by Hughes as reasons for justifying the use of lights in place of gages:

(1) Several Federal Aviation Agency pilots have expressed a preference for lights over gages.

(2) Section 6.604(f) and (g), Civil Aeronautics Manual, allows lights for gearbox temperatures and pressures. It seems that these temperatures and pressures are at least as important as those for which this review case is being requested.

(3) The language of Section 6.604(f) and (g), to wit: "A warning device to indicate oil temperature," etc., makes it unclear why Section 6.604(c), (h), and (j) has been interpreted by the Washington Office to mean "a gage." A generalized definition of an indicator is "one that shows or points out; an indication or sign; a device or apparatus for indicating something." Warning lights seem to meet this definition.

(4) It is in the public interest to minimize the number of instruments which are diversionary insofar as the pilot's complete attention to the surrounding terrain is involved.

(5) Substitution of lights for gages results in a lower cost, lower weight installation.

(6) Considerable difficulty has been encountered with certain of the electric transducer gage type instruments in the Model 369 helicopter.

c. The arguments of Hughes have been considered. They do not appear to be persuasive for the following reasons:

(1) The reference to pilot preference cannot be regarded as more than a hearsay report of personal opinion at this time.

(2) The current airworthiness requirements, Section 6.604(c), (h), and (j), Civil Air Regulations, each requires an "indicator." In other places, a "warning device" is required. The intent, where an indicator is required, is that it will actually specify the quantity or value present at the time it is read. It will also indicate a change or trend in this quantity or value.

(3) A "warning device" could be interpreted as a warning light or some other contrivance to alert the crew of an unsafe or impending unsafe condition in such a manner that attention is drawn to the situation without conscious scanning of the instrument panel being required. The specific use of the words "indicator" and "warning device" makes this difference of intent clear.

(4) Section 6.734, CAR, requires the marking of powerplant instruments to specify the (1) maximum and minimum safe operating limits, (2) the normal operating ranges, and (3) the takeoff and precautionary ranges. This requirement provides a clear indication of the type of information intended to be provided by required indicators.

(a) Substitution of lights for gages would prevent recognition of trends in oil pressure, oil temperature, and fuel pressure. Trend information is of significant benefit in maintaining operational safety. It permits recognition of trouble in early stages and correction before a hazardous situation has developed. Recognition of this fact is one reason that the word "indicator" is used rather than "warning device" in the affected requirements.

(b) Pressure and temperature ranges for idle, normal, takeoff, sea level, and altitude conditions may be at different values, and it is unlikely that warning lights would be arranged to accommodate appropriate indications for these varying power conditions.

(c) Indicators are specified, therefore, because in view of the factors discussed herein it is considered that safe overall operations require indication of trends as well as quantitative values of the affected parameters under various operating conditions.

(d) The Hughes report that gages are unreliable is contrary to our experience with gages. Reliable gages are available and have been used extensively for many years.

(5) It is not considered that it would be in the public interest to reduce the level of safety by substituting warning devices that fail to provide to the crew information considered to be necessary. If Hughes wishes to bring about a further improvement in safety by voluntarily providing warning lights in addition to the required indicators, we would concur; but we could not agree that a net improvement in safety would result if lights are provided in lieu of indicators.

(6) No showing of an unreasonable economic burden has been made. Economic considerations would not constitute justification for any measurable reduction in safety, and since any burden involved in this case would not appear unreasonable, this argument is not regarded as carrying any persuasion.

4. CONCLUSIONS

An installation not providing a quantitative indication of the parameters appropriate to the indicators required by Sections 6.604(c), (h), and (j) and 6.734, CAR, would not satisfy either these requirements or their intent. Hughes proposal, therefore, would fail to attain the level of safety intended by these regulations.

REVIEW CASE NO. 35 MINIMUM FLIGHT CREW DETERMINATION FOR THE MODEL DC-9 AIRPLANE (Issued 22 May 1964)

1. ORIGIN

The Western Region, in a memorandum dated March 24, 1964, advised that Douglas had applied for a maximum certificated weight of 85,000 pounds for its Model DC-9. The region also advised that Douglas has requested some assurance, in writing, from the Agency that they will be able to deliver DC-9s with airworthiness certificates with a crew of two at weights in excess of 80,000 pounds. The region has requested: (1) assurance that it is proper to provide Douglas with a written statement which will set forth a specific position by the Agency covering the conditions under which a minimum flight crew of two could be authorized for the DC-9, and (2), clarification of the meaning of the wording on the Form FAA-26, Export Certificate of Airworthiness, which states in part, ". . . is considered airworthy in accordance with a comprehensive and detailed airworthiness code . . ."

2. REGULATIONS AFFECTED

CAR 4b.720 - Minimum Flight Crew

The minimum flight crew shall be established by the Administrator as that number of persons which he finds necessary for safety in operations authorized under section 4b.721.

CAR 4b.721 - Types of Operation

The types of operation to which the airplane is limited shall be established by the category in which it has been found eligible for certification and by the equipment installed. (See the operating rules in this subchapter.)

3. FACTS IN THE CASE

a. The minimum flight crew for the airplane's intended operation is determined during the type certification process in accordance with CAR 4b.720 and 4b.721. CAR 4b does not require that the maximum weight, for which approval is sought, be a limiting factor in determining the minimum flight crew.

b. The Director, Flight Standards Service, in a letter dated March 16, 1964, advised Douglas of its conclusion on the minimum flight crew for the DC-9, based on the findings and recommendations of the working group. The Director, in a letter dated March 24, 1964, to Douglas, restated the conclusion as follows: ". . . our conclusions /are/ that the Douglas DC-9 could be safely operated in air carrier service with a properly trained crew of two pilots. This conclusion was based on the assumption that the information and proposals presented by the Douglas Corporation /to the working group/ would be verified during the certification tests of the DC-9 and provided the elements in which it will be operated are as we know them today."

c. The additional crew requirements set forth in the operating rules are applicable only when operations are subject to the parts involved.

d. The statement on the Export Certificate of Airworthiness was issued under the provisions of Part 406, paragraph 14, of the Regulations of the Administrator. The phrase, ". . . airworthy in accordance with a comprehensive and detailed airworthiness code . . .", on Form FAA-26 relates only to the airworthiness parts of the Federal Aviation Regulations which are applicable to the type certification of the aircraft.

4. CONCLUSIONS

a. The Western Region may advise Douglas in writing that the minimum flight crew for its DC-9 is determined during type certification solely in accordance with the provisions of CAR 4b. The additional crew member requirements, as they might be specified in the operating rules, applicable to the particular operation, are the responsibility of the operator.

b. The Export Certificate of Airworthiness for the DC-9 should be based solely on compliance with CAR 4b and the special requirements of the importing country.

REVIEW CASE NO. 36. LEAR JET MODEL 23 OIL TEMPERATURE AND PRESSURE GAGES (Issued 9 June 1964)

1. INTRODUCTION

Lear Jet Corporation desires revisions to the certification basis for the Lear Jet Model 23 airplane. Specifically, the revisions desired are (a) deletion of the oil pressure warning light as required by Special Condition 3.655(b)(4), while still retaining the oil pressure indicator as required by Special Condition 3.655(b)(9); and, (b) use of the alternative requirement of Special Condition 3.655(b)(10) to provide an oil temperature indicator.

2. HISTORY

a. On August 22, 1962, the Washington Propulsion Branch compiled a list of proposed powerplant special conditions for the Lear Jet. Among the powerplant instruments required in addition to those specified in Section 3.655(b) of the Civil Air Regulations was an "oil pressure warning means to warn when oil pressure has gone below the established low limit." This same requirement was retained in the revised list of September 7, 1962.

b. Personnel of the Washington Office met with Lear representatives on September 25, 1962, to discuss the proposed special conditions. Lear objected to the above item, arguing that since Section 3.655(b), CAR, already required an oil pressure indicator, this should be sufficient. Lear also questioned the need for the oil temperature indicator required by the same section, saying that (1) General Electric Company requires only an oil temperature warning light, (2) there are no provisions in the airplane for cooling the oil, and, (3) any abnormally high temperatures would be reflected in the reading of the oil pressure gage.

c. As a result of discussions with Lear at Wichita on October 4, 1962, Lear requested a complete list of required powerplant instruments. This list when prepared included:

(1) Oil pressure warning means to warn when oil pressure has gone below the established low limits,

(2) Oil pressure indicator,

(3) Oil temperature indicator.

d. On October 8, 1962, the Washington Propulsion Branch (FS-140) sent a memorandum to the Central Region Propulsion Section (CE-214) stating the FS-140 position with respect to the above items, namely, (1) special conditions for other turbine-engine installations have customarily included an oil pressure warning means, (2) at least two other attempts to use a warning light instead of an oil temperature gage have been disallowed, (3) the Eastern Region is unaware of anything unique about the G.E. engine which would make the oil temperature indicator unnecessary, and, (4) an oil temperature gage is required under Part 43, CAR, as well as Part 3.

e. On December 13, 1962, FS-140 sent another memorandum to CE-214, stating the objective of the requirement for an oil temperature indicator, which is to make available to the crew a continuous reading of oil temperature so that normal temperatures can be observed, trends noted as they occur, differences observed between indicators for each engine of a multiengine installation, and confirmation can be obtained that the temperature remains within approved limits. The memorandum also stated the conditions under which an oil temperature warning means might be acceptable, namely, by providing a complete fault analysis of the oil system showing that any possible faults, failures, or deficiencies in the engine oil system would be immediately shown by the oil pressure indicator.

f. On January 28, 1963, CE-214 sent a memorandum to FS-140 recommending a change in the oil temperature indicator special condition as follows:

> Oil temperature indicator unless an equivalent indication of abnormal oil temperature is provided to the flight crew

The reason given was that the previous special condition was restrictive to design and denied the applicant a right to have his design considered under Section 3.10, CAR.

g. In a memorandum dated February 25, 1963, the Washington Office advised the Central Region that the words "or equivalent" could be added to "oil temperature indicator" to avoid the possible restriction to design. The equivalence in this case was intended to relate to a temperature indicator. This would have required a means whereby the normal temperature range as well as any abnormal temperatures could have been monitored.

h. On February 25 and 26, 1964, a preflight type certification board meeting was held at Wichita, Kansas, to discuss items requiring action by Lear prior to issuance of a type inspection authorization. Among the propulsion items still to be resolved were those requiring an oil pressure warning means and an oil temperature indicator.

i. In a letter dated March 10, 1964, to the Propulsion Section Chief, FAA, Kansas City, Missouri, Lear Jet Corporation offered data to substantiate (1) the use of an oil temperature warning light in lieu of temperature indicators, and, (2) the deletion of low oil pressure warning.

j. Lear letter of March 10, 1964, was transmitted to the Washington Headquarters on March 18, 1964, with Central Region recommendations. The Central Region was inclined to concur with Lear proposal to delete the oil pressure warning means requirement but not with Lear proposal to use an oil temperature light.

k. Meetings were held in Washington, D. C. on March 26 and 27, 1964, with Lear and General Electric Company representatives. Further action on the subject items was deferred pending substantiation by G.E. that its engine does not require temperature indication.

l. G.E. sent a letter dated April 17, 1964, to the FAA in Washington and to the Central Region, attempting to justify its stand that there is no need for an oil temperature gage.

m. In response to a telephone request on April 22, 1964, the Central Region sent a message to the Washington Office, stating that it considered the evidence in G.E. letter of April 17, 1964, inadequate justification for a single oil temperature light since the engine is approved under Part 13, CAR, with a maximum oil temperature limit.

3. DISCUSSION.

a. While the principal determination to be made in this case is whether an oil temperature light can be permitted instead of an oil temperature gage, it is necessary to consider also whether the requirement for an oil pressure warning means must be retained because oil temperature and oil pressure are interrelated.

b. Lear contends that oil pressure is the most reliable single measure of lube system performance. Lear cites the CJ610-1 Turbojet Engine Maintenance Manual, SEI-136, dated July 1, 1962, which makes reference to oil pressure for lubrication troubleshooting. Another section of the manual gives methods for relating normal operating oil pressure to r.p.m. for an individual engine and gives characteristics which indicate possible lube malfunction. No reference is made to oil temperature.

c. In further support of the relative unimportance of the oil temperature as an indicator of lube system performance, Lear refers to a study conducted on the T-38 Talon, which uses a J-85 (military version of the CJ-610) engine. The study was conducted by the Aeronautical Systems Division of the Air Force, G.E. and Norair. The conclusion was that a temperature gage will not aid the pilot in determining impending bearing failures. When a bearing begins to fail, material clogs the filter, resulting in an oil pressure rise. A test was conducted in which the J-85 engine was operated at military speed for one minute without oil. The bearing scavenge oil temperature rose to 412°F. but the engine components were undamaged. In view of these results, Lear proposes to use only an oil temperature warning light set to operate when the oil scavenge temperature reaches 355°F. ± 15°F. The maximum oil temperature approved for the CJ610-1 engine is 380°F.

d. The Central Region recommended concurrence with Lear proposal to delete the requirement for an oil pressure warning means. The compensating factors would be (1) the fact that the engine can operate without oil pressure for one minute without damage (based on tests conducted on a J-85 engine which is similar to the CJ610-1), and, (2) the fact that no oil pressure warning means was required by special condition for the Beech 65-90T equipped with Pratt and Whitney PT6-6A turboprop engine. In this case, the Washington Office concurs in this decision.

e. The FAA did not concur with the Lear proposal to use a warning light in lieu of a conventional oil temperature indicator and asked for further substantiation. Lear called upon the General Electric Company to provide the substantiation. G.E. complied in a letter dated April 13, 1964.

f. In this letter, G.E. states that engine malfunctions which result in oil temperature variations usually also result in oil pressure variations of a magnitude which show up on the oil pressure gage, thus indicating to the pilot that corrective action should be taken. It is probable that this is correct in most cases, but a significant pressure indication would probably lag the temperature indication. G.E. states that oil pressure is an indication of high or low oil temperature; high oil temperature is indicated by low oil pressure and low oil temperature is indicated by high oil pressure. This effect would be due to the changes in viscosity with temperature; because other factors may also influence oil pressure, an indication of pressure alone is not sufficient for troubleshooting. In the case of a bearing failure, for example, both pressure and temperature would rise. Bearing seal failures, on the other hand, will cause an oil temperature increase primarily and a pressure decrease secondarily. This seal failure could then lead to a subsequent bearing failure. For these reasons, it is doubted that oil pressure provides a basis on which instantaneous decisions can be made. With the warning light proposed by G.E., bearing trouble must progress to a point where the oil temperature is sufficiently high to activate the light. Meanwhile, the oil pressure would drop due to oil temperature and then increase when the filter became clogged. This would result in late recognition of the real trouble. It is true that there is considerable variation in temperature with changes in flight speed, r.p.m., and altitude. With operating experience, however, the crew would be able to recognize an abnormal rise, particularly since there are two engines and readings can be compared.

g. G.E. states that oil temperature measurement also has its place in detailed ground troubleshooting procedures. If current maintenance manuals issued by G.E. and approved by FAA do not call for any maintenance action based on oil temperature measurements, as G.E. contends, this is not an indication that FAA considers oil temperature unimportant when related to basic engine airworthiness. An oil temperature limit has been established and approved by FAA and is an engine operating limitation that must be observed.

h. At one time, the Washington Office recognized that it might be possible to accept a simple warning means to show when oil temperature has reached established limits if Lear or G.E. could show by a complete fault analysis of the engine oil system that any faults, failures, or deficiences in the engine oil system would be immediately shown by the oil pressure indicator. Lear and G.E. have attempted to do this but have not presented persuasive arguments. The Washington Office also indicated that if this change of instrumentation were to be made, it should have the concurrence of the Eastern Region Engineering and Manufacturing Branch. The Eastern Region does not concur in this proposed change.

i. In general, an indicator is a device for showing the state of affairs with respect to some measurable quantity. Customarily, a gage has been required for this purpose. A warning means, on the other hand, is a device for giving conspicuous notice beforehand of approaching danger. A bell or a light may be used for this purpose. A warning device gives extra protection by calling the crew's attention to a possibly hazardous situation rather than by letting the crewmembers make such a determination for themselves by scanning the indicators. Where the special conditions specify an indicator and a warning means in connection with the measurement of a certain engine condition, both should normally be provided. If compensating factors are such that one or the other may be eliminated, this is another matter. Normally, if a choice is to be made, it would be preferable to eliminate the warning means and to retain the indicator. If any device is to be used as an equivalent to a gage, it must provide substantially the same information that a gage would provide.

j. In summation, arguments in favor of a light instead of an oil temperature gage are:

(1) The light provides a warning to the crew in the event the crew has neglected to observe the temperature gage.

(2) The light, being set well below the oil temperature limit, can usually warn in time to prevent damage to the engine.

(3) Temperatures vary widely with changes in airspeed, engine speed, and ambient temperature so that a gage does not

provide a reliable indication of engine troubles. A light is sufficient when used in conjunction with an oil pressure indicator.

k. Arguments in favor of an oil temperature gage instead of a warning light are:

(1) The gage supplies more information; it shows trends.

(2) The gage is a constant indicator of changes in oil temperature; whereas, a light shows only one selected temperature.

(3) Only a glance at the gage is necessary to confirm that it is in operation. The light could only be checked by a switch and there is no assurance that it will be operable when needed even immediately after a check is made.

(4) Trouble can be recognized early because the crew is familiar with the oil temperature pattern for normal operation.

(5) In multiengine installations, gage readings can be compared and differences between gages will be noticeable.

(6) Gages can be used to monitor temperatures over a wide range, not only at some limit.

(7) The gage can be used for detailed ground troubleshooting.

(8) In the event of a broken line where loss of oil is experienced, the trouble would not be detected by a light because there would be no oil to get hot.

(9) Since, with the G.E. engine, only the oil from No. 2 bearing is being monitored, the only time the temperature is likely to get hot enought to quickly activate a light is when the No. 2 bearing fails. Other bearing failures would be more readily detected if a gage were used.

(10) Where a temperature limit has been set for an engine, as in this instance, it has been the established practice to require a gage.

4. <u>CONCLUSION</u>

After considering all the available evidence in this case, it is concluded that:

a. The Washington Office concurs with the Central Region finding that, in this case, the requirement for an oil pressure warning means may be deleted from the special conditions for the Lear Jet.

b. The requirement for an oil temperature indicator should be retained. The alternative of a warning light is not considered equivalent.

REVIEW CASE NO. 37 MAXIMUM ROTORCRAFT-LOAD COMBINATION WEIGHT (PART 133) (Issued 19 June 1964)

1. INTRODUCTION.

a. Bell Helicopter Company desires approval of a proposed rotorcraft-load combination (sling type) on their model 204B helicopter at a maximum total weight of 9,500 pounds which is 1,000 pounds in excess of currently certificated weight under CAR 7, Category B. Bell recently originated a project with the Southwest Region to approve this installation for operation under Part 8 and is showing compliance at the higher weight with the structural flight loads and main component service life determinations of CAR 7.

b. Bell now desires that the sling-equipped Model 204B at 9,500 pounds total weight be eligible for operation under the new FAR Part 133. Bell has noted, however, that Part 133, which becomes effective May 17, 1964, requires that the rotorcraft-load combination must not exceed the maximum weight certificated under CAR 7. Bell notes that acquiring an external load in flight is normal to sling-equipped helicopters and that it is reasonable, for Class B and Class C loads, to exclude the external load weight when complying with the landing and takeoff structural requirements of Part 7. Bell requests that this consideration be made in interpreting the airworthiness requirements of Part 133 to permit operation of their Model 204B at the 9,500 pounds total weight under Part 133 when it becomes effective on May 17, 1964.

c. The Southwest Region concurs with Bell's proposal to neglect landing loads.

2. CHRONOLOGICAL HISTORY

a. January 17, 1964. The new FAR Part 133 - Rotorcraft External-Load Operations was adopted January 17, 1964, published in the Federal Register on January 24, 1964, to become effective May 17, 1964.

b. March 24, 1964. Bell Helicopter Company in their letter of March 24, 1964, to FS-100 requested that FAR 133 be reviewed and that an interpretation of this regulation be issued to enable operation under this regulation, provided adequate structural substantiation is furnished, but that complete certification requirements under the applicable Part 6 or 7 (in this case Part 7) need not be applied.

c. April 1, 1964. FS-100 acknowledged Bell Helicopter Company's March 24, 1964, letter and indicated that a review of FAR 133 would be initiated and every effort made to clarify the regulation prior to the effective date of FAR 133 - May 17, 1964.

d. April 2, 1964. A memo dated April 1, 1964, was received by FS-100 from SW-210 forwarding additional information relative to approval of the Bell Model 204B at the higher gross weight of 9,500 pounds under Part 8 for external load operations. This memo also contained the Region's recommendation to add an exception under FAR 133.43(c) which would provide for operating Class B and C rotorcraft-loads combinations at total weights in excess of the maximum certificated weight under Part 6 or 7.

e. April 8, 1964. A meeting between AIA and the FAA was held in Room 510B at the request of Mr. Simpson, AIA Technical Director, to discuss various common rotorcraft problems. This meeting included a discussion of the new Part 133 and its airworthiness requirements. The AIA indicated that certain clarifications and revisions of Part 133 were needed including the deletion of landing loads for Class B and C rotorcraft-load combinations.

f. April 9, 1964. A conference was held on April 9, 1964, including FS-40, FS-100, and GC-22 personnel to discuss the Bell request for interpretation of Part 133 and to determine the necessity for further regulatory action under Part 133.

g. April 20, 1964. A memo dated April 20 was sent by FS-100 to SW-210 informing that a review case is being prepared which would permit Bell to operate their Model 204B helicopter with Class B and C rotorcraft-load combinations at the substantiated inflight weight of 9,500 pounds.

3. FACTS IN THE CASE

a. Part 133 which is primarily an operating rule also contains the airworthiness requirements for the rotorcraft-load combination under Subpart D by either defining the applicable standard or by reference to CAR 6 and CAR 7. Additionally, Section 133.19 specifies that the rotorcraft must have been previously type certificated under, and must meet the requirements of CAR 6 or 7 but not necessarily with the external-load attaching means installed.

b. Bell Helicopter Company advises that no ground operations or landings of the Model 204B in Part 133 operations as a Class B rotorcraft-load combination will be performed with the external load attached.

c. CAR 7.230(c) specifies that the design weight used in the landing conditions shall not be less than the maximum weight of the rotorcraft. In view of the facts in paragraph 3b, the maximum weight for showing compliance with the landing loads of CAR 7 for the Bell Model 204B with Class B external load need not exceed the maximum rotorcraft-load combination weight less the weight of the jettisonable external load.

4. <u>CONCLUSIONS</u>

a. External load operations with Class B and C rotorcraft-load combinations do not include takeoff and landings with the external load attached and supported by the rotorcraft.

b. With respect to their Model 204B helicopter as a Class B rotorcraft-load combination, the Bell Helicopter Company should be permitted to exclude the jettisonable external load weight under the structural landing loads and emergency landing conditions of CAR 7 in evaluating compliance with the weight and center of gravity paragraphs of 133.43(c).

c. If different maximum weights and/or ranges are established, the Rotorcraft-Load Combination Flight Manual specified under 133.45(b) should contain the maximum rotorcraft-load combination weight cg ranges with and without the jettisonable external load attached.

d. A recommendation for a regulatory project should be initiated to clarify section 133.43(c).

REVIEW CASE NO. 38 LEAR JET MODEL 23 - STICK SHAKER-PUSHER INSTALLATION (Issued 24 July 1964)

1. ORIGIN AND PROBLEM

a. The Central Region on June 19, 1964, requested a Review Case decision on their finding that the Lear Jet Model 23 does not meet special condition CAR 3.120 in that the inherent flight characteristics did not give a clear indication to the pilot that the airplane was stalled prior to entering a flight condition where normal recovery from a stall could not be accomplished. The Lear Jet Corporation objected to this finding and to the Central Region's proposal to request a multiple-expert-opinion-evaluation team.

b. In lieu, the manufacturer proposed a stick shaker/pusher installation which would be so activated that the airplane would comply with the stall warning and characteristics requirements without actually stalling the airplane during type certification testing.

2. REFERENCE REGULATIONS

a. Lear Special Condition CAR 3.120 Stalling Symmetrical Thrust

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

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

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

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

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

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

b. CAR Section 3.10 Eligibility for Type Certification

An airplane shall be eligible for type certification under the provisions of this part if it complies with the airworthiness provisions hereinafter established or if the Administrator finds that the provisions not complied with are compensated for by factors which provide an equivalent level of safety: Provided, that the Administrator finds no feature or characteristic of the airplane which renders it unsafe for the category in which it is certificated.

3. CHRONOLOGICAL HISTORY

a. May 23, 1964 The manufacturer objected to the regional finding during type tests that the airplane did not meet the stall requirements and to resolve this matter was informed by the region that a multiple-expert-opinion team would be established to evaluate the stalling characteristics of the airplane.

b. May 27, 1964 The manufacturer requested cancellation of the proposed team evaluation and presented a revised design proposal which provided an artificial stall warning and stick pusher installation.

c. June 10, 1964 The Lear Jet Corporation proposed by letter to demonstrate a stick pusher installation to show an equivalent level of safety to Special Condition 3.120 under the provisions of CAR 3.10.

d. June 19, 1964 The Central Region forwarded a letter to the Lear Jet Corporation advising that prior to evaluating the Lear proposal, the following information would be required:

(1) Design Data:

(a) Descriptive data including drawings and schematics.
(b) Environmental test proposal including reliability of components and installations.
(c) Fault analysis.

(2) A preliminary Flight Test Report which includes the effects of the following variables on stick pusher operations:

(a) Weight and center of gravity.
(b) Accelerated stalls.
(c) Engine power.
(d) One-engine inoperative condition.
(e) Turbulence and gusts.
(f) Flap, gear, and spoilers.

(3) Results of the testing conducted by Lear should include the following:

(a) Stick shaker and stick pusher actuation speeds.
(b) Lift coefficient at stick pusher action and C_{LMAX} for the condition.
(c) Elevator control force input and rate.
(d) Change in lift coefficient before pusher input is released.

e. June 26, 1964 In response to the Central Region's request of June 19, 1964, a Review Case Team was formed to: (1) evaluate the concept of using an automatic device to show compliance with the stall characteristics requirements in lieu of inherent characteristics, (2) determine whether or not an equivalent level of safety is provided under the provisions of CAR 3.10 by the Lear Jet Corporation's proposed stick shaker/pusher installation and (3) recommend a course of action to be taken as a result of the team's evaluation of the installation and operation.

f. June 30, 1964 Seven recommendations by the Review Case Team were included in the summary of the team's evaluation given to the Lear Jet Corporation by the team's chairman. (The Lear Jet installation, as evaluated by the team, consisted of a single vane angle of attack sensor, a potentiometer, an angle of attack indicator, a computer, an accelerometer that deactivated the pusher when the normal acceleration on the airplane decreases to a value of 0.5g, use of the auto pilot's pitch servo, and a flap configuration input device. The shaker was activated at approximately 1.07 V_S with the pusher activation considered as V_S.) The recommendations were as follows:

(1) A redundant stick pusher system should be required in addition to the presently installed stick shaker, stick pusher system.

(2) A means should be provided to check the functioning of the stick shaker and the stick pusher prior to flight.

(3) Adequate protection should be provided against malfunctions saturating the magnetic clutches.

(4) A malfunction warning device should be provided to show power failures. This device should be such that a failure is promptly detected. Procedures should be developed for safe continuance of flight subsequent to a failure.

(5) The stick shaker and stick pusher systems should be operative for all normal operations. No cutout or automatic cut-off should be utilized other than guarded on-off switches.

(6) The stall warning system (stick shaker) should be actuated at a speed at least seven percent above the stalling speed at a DV/DT of one knot per second.

(7) The stick pusher and stick shaker systems should not be actuated in normal flight regimes as a nuisance.

g. June 30, 1964 Lear Jet comments were submitted by memorandum to the team chairman. Lear Jet concurred with all the recommendations except the one recommending redundancy of the stick pusher system. Lear Jet contended that the airspeed and angle of attack information provide dual protection, the stick shaker provides advance stall warning, the stick pusher is a back-up device to assure no stall, and the system is simple, rugged, reliable, and thus a redundant stick pusher system is not needed.

h. July 1, 1964 The Lear Jet Corporation advised Washington by phone that it was installing dual vane sensors, dual stick shakers, and dual input signals to a single stick pusher. Each sensor would activate both shakers and the pusher to assure the pilot having both a warning and the pusher as a back-up to preclude a stall.

i. July 7-9,1964 The special Washington Type Certification Review Team established for the Lear Jet Model 23 project, convened at the Lear Jet factory on July 7, and attended an Interim Type Certification Board Meeting conducted by the Central Region on July 8-9. The team members examined the stick shaker/pusher system in detail and flew an airplane with the latest version of the installation. Malfunction flight checks were included. The Central Region was requested to have the manufacturer complete and submit to the FAA for approval the drawings for the final installation. Three basic points for improvement were stressed as follows: (a) better warning indication means, (b) readily accessible quick disconnect switches, and (c) evidence that a malfunction in the pitch servo unit is extremely remote.

4. RELATED BACKGROUND

a. Many of the presently certificated U.S. turbine powered airplanes have artificial stall warning installations, most of which are necessary to meet the stall warning requirements in one or more configurations.

b. All presently certificated U.S. turbine powered airplanes have satisfactorily met the stall characteristics requirements by inherent aerodynamic characteristics or by limiting the elevator control travel.

c. Most manufacturers of swept wing and/or T-tail airplanes have had problems associated with the stall and have redesigned certain portions of the airframe to comply with the stall requirements.

d. The British have some stick shaker/pusher experience and are requiring stick pusher installations on the BAC-111, Trident, and VC-10. Representatives of the British Aircraft Corporation and the Air Registration Board have reported favorably the use of the devices as an anti-stall protection.

e. The U. S. military services are known to have used a stick pusher on two types of fighter airplanes but no official report thereon is known to be available at this time.

f. Automatic devices have been employed in many models to meet flight characteristics requirements. Such devices range from simple bungee springs to sophisticated automatic stabilization equipment installations. To date, for civil certification, the devices have been employed to meet requirements involving stability, control, trim, and stall warning.

5. DISCUSSION

a. For safe operation, it is essential that stalling be prevented when stalling characteristics are unknown or are known to be hazardous. This may be accomplished in more than one way. For example, an automatic and reliable device could be used to assure the pilot having adequate warning to take the correct stall preventive action, or, an automatic and reliable device could be used to cause a correct flight control action to prevent a stall. The use of any automatic device is of course subject to a malfunction or failure hazard that is in inverse proportion to the degree of reliability provided. Hazardous unwanted action of such a device must be protected against.

b. The concept of using auxiliary devices to meet flight characteristics requirements has been acceptable to date for stability, control, trim and for stall warning. A device that meets stalling characteristics requirements is consistent with the concept.

c. The Lear Jet stick shaker/pusher installation is precedent-setting in that the applicant proposes to show compliance with the stall requirements by an automatic device in lieu of demonstrating inherent aerodynamic stall warning and normal stalling recovery characteristics. Lear Jet proposes to provide a stick shaker stall warning at a speed approximately seven percent ahead of the stick pusher activation and to activate the stick pusher before reaching "an angle of attack measurably greater than that of maximum lift."

d. If the concept is acceptable to use an automatic device to meet the stall characteristics requirements similar to that previously accepted for stall warning, stability, control, and trim requirements, then it becomes necessary to determine what compensating factors may be considered to provide the equivalent level of safety under the provisions of CAR 3.10 for the Lear Jet when not complying with the specific special conditions. At the Flight Test Regional Chiefs' Conference in June 1964, it was concluded that the Lear proposal to activate the stick pusher as evidence of the stall would require (a) both the stick shaker stall warning and the stick pusher to provide an equivalent level of safety, (b) that stick pusher activation must be considered the reference V_S for performance, even if occuring before Max C_L,(c) a regular fault and reliability analysis, and (d) A review case team should be made before final determination of compliance was made. These conclusions were transmitted by the Chief, Flight Test Branch by phone to the Central Region.

e. The applicable special condition on stalls (3.120 is the same as CAR 4b.160 and 4b.162) requires a clear indication to the pilot that the airplane is stalled (at an angle of attack measurably greater than that of maximum lift) <u>except</u> that an airplane with an unmistakable inherent aerodynamic warning in all required configurations need not be investigated for compliance with the stall characteristics requirements below a speed value which provides an adequate stall margin (normally seven percent). The special conditions further require the stall warning to be clear and distinctively apparent to the pilot, and with sufficient margin to prevent inadvertent stalling both in straight and turning flight. Because of the stick pusher activation before Max C_L, the Lear Jet must be considered under the "exception" provision of the special condition. The applicant proposes to meet the inherent aerodynamic warning requirement on an equivalent level of safety basis through the use of automatic devices.

f. The use of any automatic or powered device for showing compliance with the flight characteristics requirements must be investigated for structural integrity, effects of malfunction or failure, reliability, and evaluated by flight tests to determine that it performs its intended function as a required item for type certification.

6. CONCLUSIONS

a. We conclude that the Lear Jet proposal to incorporate automatic devices such as a stick shaker/pusher in its Model 23 airplane as a means to provide warning indication of proximity to stall and to preclude stalling the aircraft under the conditions of operations set forth in Special Condition CAR 3.120 (c),(d),(e), and (i), is acceptable under the equivalent safety provisions of CAR 3.10 subject to the following:

(1) The equipment systems, and installation are designed and installed to insure that the intended function is performed reliably under all reasonably foreseeable operating conditions, including expected environmental effects.

(2) The equipment, systems and installation are designed to safeguard against hazards to the airplane in the event of their malfunctioning or failure.

(3) Dual, independent stick shaker stall warning systems are provided. Each system is to actuate in such a manner as to give an unmistakable, reliable warning to the pilot(s) with an adequate margin ahead of the stall. (Duplicate portions are to include the angle of attack transducer (vane), flap position transducer, shaker motor, computer, and cutoff means.)

(4) A stick pusher system is provided. The characteristics of this system should be such that the stick force is sufficiently great and is applied in such a manner so as to preclude the pilot from inadvertently overpowering the device. The resulting angle of attack change shall be such as to prevent inadvertent aerodynamic stalls.

(5) The speed at which the stick pusher is actuated before reaching Max C_L is defined as the stalling speed.

(6) Components common to the stick shaker and pusher systems may be used on a duality basis except that dual acceleration limiters and dual pitch servo units need not be installed in the stick pusher system provided that an acceptable level of reliability is

established for these units. The correction of any reasonably probable fault in the remaining elements of the stick pusher systems shall not obviate the stick shaker system.

(7) The operation of the stick pusher system is such that it automatically disengages when it has decreased the angle of attack of the airplane to a point less than that at which the pusher is set for actuation.

(8) The stick pusher system is designed such that it can be quickly and positively disengaged by the pilot(s) to prevent it from interfering with their control of the airplane.

(9) An accelerometer is provided to automatically render the stick pusher system ineffective when the normal acceleration on the airplane decreases to a value of 0.5g.

(10) Power failure indications for each individual shaker/pusher system are provided.

(11) The stick pusher system design is such that flight in turbulence does not produce hazardous deviations from the flight path.

(12) A visual indicating means is provided to monitor in-flight functioning of at least one of the angle of attack transducer vanes.

(13) Calibrated means are provided to check proper functioning of the stick shaker/pusher system(s) prior to flight.

(14) The related operating limitations and procedures, together with any information concerning the airplane found necessary for safety during operation of the required stick shaker/pusher system(s), are to be included in the airplane flight manual, expressed as markings or placards, or made available by such other means as will convey essential information to the operator and/or pilot(s).

b. The type certification data requested by the Central Region from Lear Jet in its letter of June 19, 1964, is to be obtained and made part of the type design data.

REVIEW CASE NO. 39 HEADQUARTERS, OKLAHOMA CITY AIR MATERIEL AREA, UNITED STATES AIR FORCE REQUEST FOR EXEMPTION FROM SECTION 4b.18 OF THE CIVIL AIR REGULATIONS FOR THE MODEL VC-137C AIRPLANE WITH ITT MODEL 3544 DISTANCE MEASURING EQUIPMENT (Issued 24 July 1964)

1. INTRODUCTION

The Headquarters, Oklahoma City Air Materiel Area, USAF, through The Boeing Airplane Company, has attempted to obtain approval of the International Telephone and Telegraph Corporation's Model 3544 distance measuring equipment (DME) installation on the Presidential airplane VC-137C S/N 62-6000 (Boeing Model 707-353B). The USAD indicated that the Western Region has ruled by correspondence through Boeing that this equipment does not comply with Section 4b.18 of the CARs and, therefore, the installation cannot be approved. As a result, a request for exemption from these provisions was requested by the USAF. It was pointed out by the USAF that the equipment performed satisfactorily in flight tests conducted by the Eastern Region (reference EA-216 report dated July 31, 1963). The USAF contends that replacement of the ITT Model 3544 DME with equipment which complies fully with Technical Standard Order C66, as required by Section 4b.18, would cause undue expense and create a configuration problem relative to other aircraft in the Special Aircraft Missions (SAM) fleet without improvement in flight safety.

2. CHRONOLOGICAL HISTORY

a. In a letter of April 23, 1964, directed to the Federal Aviation Agency, Washington, D.C., Headquarters, Oklahoma City Air Materiel Area, USAF, requested an exemption from Section 4b.18 to permit Agency approval of the ITT Model 3544 DME on the VC-137C S/N 62-6000 airplane.

b. In a message of May 19, 1964, followed by a telephone call on May 21, 1964, to the Western Region, FS-100 requested confirmation of findings and decision in the matter of the ITT Model 3544 DME installation in the VC-137C airplane.

c. In reply to the message from FS-100 dated May 19, 1964, the Western Region indicated in a message of May 20, 1964, that Boeing was denied a certification for the ITT Model 3544 DME because it does not have a TSO approval in accordance with Section 4b.18. It was indicated, however, that the installation and function of the equipment was found to be satisfactory. The Western Region recommended that, on the basis of the flight test report of July 31, 1963, by R. Lamprecht of the Eastern Region, an exemption be processed similar to those granted the air carriers (See paragraphs 3c through 3e).

d. In reply to the letter of April 23, 1964, from Headquarters, Oklahoma City Air Materiel Area, USAF, FS-100 advised, by letter of June 1, 1964, that the issues raised in connection with the ITT Model 3544 DME installation in the VC-137C airplane were being explored with the Western Region and that it might be possible to approve this installation without the need for an exemption.

3. FACTS IN THE CASE

a. The Agency adopted TSO-C66 effective August 1, 1960, which sets forth the minimum performance standards for the approval of distance measuring equipment. These standards incorporate those specified in RTCA Paper 167-59/DO-99 dated September 8, 1959, titled "Minimum Performance Standards - Airborne Distance Measuring Equipment (DMET) Operating Within the Radio Frequency Range of 960-1215 Megacycles."

b. The ITT Model 3544 DME fulfills all of the minimum performance standards necessary for approval by the Agency as set forth in TSO-C66, with the exception of paragraph 2.11(b) of the RTCA paper. This paragraph applies specifically to the receiver decoder selectivity of distance measuring equipment and sets forth the following minimum performance standards:

> 2.11 - Receiver Decoder Selectivity. Over the input signal level range from -48 dbm to the equipment's minimum tracking level, the equipment shall: * * *
>
> (b) Result in an average of not more than one successful end of search out of ten searching cycles and that one to continue in track for not more than five seconds when spacing of the received pulses is less than 6 microseconds and more than 17.5 microseconds.

c. In October 1960, the Agency granted four United States air carriers exemptions from compliance with the provisions of Section 40.170(a) to permit the use of the ITT Model 3544 DME without meeting the minimum performance standards on receiver decoder selectivity set forth in TSO-C66. The authority granted by Exemption Nos. 123, 124, and 125 (Regulatory Docket Nos. 529, 540, 541, and 542, respectively) was to remain in effect for two years from the date of issuance unless sooner superseded or rescinded.

d. The aforementioned exemptions were granted on the basis that the current use of distance measuring equipment which did not meet the receiver decoder selectivity standards would neither jeopardize the exploitation of the VOR/DME common system, nor result in a serious degradation of service to aircraft equipped with distance measuring equipment. In the future, however, with increases in the number of ground stations and aircraft equipped with distance measuring equipment, it would become necessary to require all users to adhere to the established standards in order to avoid limiting the VOR/DME common system and creating a hazardous condition. For this reason, it was considered at that time that safety would be actually enhanced by use of the distance measuring equipment which was available and, due to more expeditious handling of traffic, it would be in the public interest to permit the use of such equipment for a limited time.

e. The Agency also considered that since it was impossible to forecast the rate of increase in the use of a VOR/DME common system, it would not be in the best interest of either the public or safety to approve the use of the distance measuring equipment which lacked the decoder selectivity refinement for an extended period of time. The exemptions were granted, therefore, with the understanding that they might be cancelled at any time the VOR/DME common system requires the refinement afforded by full compliance with TSO-C66. The exemptions were also granted with the understanding that the petitioners would modify the equipment at the earliest practicable data so it would meet the established standards for approval.

f. On August 15, 1962, the Agency issued exemptions (Exemptions Nos. 123A, 124A, and 125A) which extended the authority granted under the original exemptions for an additional two-year period. These extentions were granted on the basis that: the same conditions existed which justified the issuance of the original exemptions; the operations conducted thereunder had been completed without any adverse effect on safety; and, the use of the ITT Model 3544 DME would not jeopardize the development of the VOR/DME common system.

g. The Agency has not yet implemented pulse multiplexing of distance measuring equipment ground stations as a system solution of the problem of anticipated overloading of the VOR/DME system. Therefore, the conditions which justified the issuance of the previously discussed exemptions are equally valid at this time and fully applicable to the VC-137C airplane using the ITT Model 3544 DME.

h. The VC-137C airplane being a public aircraft, is not subject to compliance with the operating rules of Parts 40, 41, 42, and with the DME requirement of Section 91.33(e) of Part 91 /New/, which applies to operation of United States registered civil aircraft.

For this reason, an exemption from the operating rules for the VC-137C airplane with the ITT Model 3544 DME would not be appropriate. Civil airplanes of the size of the VC-137C airplane are subject to the transport category airworthiness requirements of Part 4b. Part 4b contains adequate provisions for approval of this distance measuring equipment on airplanes not subject to the operating rules of Parts 40, 41, 42, and 91 /New/. Such approval under Part 4b can be pursued even though the airplane involved is not required to be type certificated.

i. The provisions of Sections 4b.10 and 4b.18 and relevant policies thereunder provide a basis for Agency approval of appliances not bearing a TSO label where applicants for a type certificate may seek Agency approval by showing that factors are provided to compensate for those standards not complied with, thus achieving a level of safety equivalent to that intended by the rule. The Agency has already found, in the pertinent exemptions issued under Part 40, that the ITT Model 3544 DME fulfills all of the minimum performance standards as set forth in TSO-C66, with the exception of paragraph 2.11(b); and that conditions relating to this deficiency were established to insure attainment of the required level of safety. These conditions can be considered under Section 4b.10 as compensating factors for lack of full compliance with the TSO standards established under Section 4b.18. For this reason, an exemption from Section 4b.18 is not necessary for the approval of the ITT Model 3544 installation in the VC-137C airplane.

4. <u>CONCLUSIONS</u>

a. The ITT Model 3544 DME meets all of the minimum performance standards of TSO-C66 with the exception of the requirements for receiver decoder selectivity in paragraph 2.11(b) of the RTCA Paper No. 167-59/DO-99 which is incorporated in and is thus a part of TSO-C66. The Agency has found compensating factors for this deficiency which insures attainment of a level of safety intended by the operating rules of Part 40 to permit the use of this equipment in air carrier operation by exemption grants.

b. An exemption from the operating rules of Parts 40, 41, 42, and Section 91.33(e) of Part 91 /New/, for the Presidential airplane VC-137C, S/N 62-6000 (Boeing Model 707-353B) for use of the ITT Model 3544 DME in air navigation is unnecessary and inappropriate because these rules do not apply to public aircraft as defined by the Federal Aviation Act of 1958.

c. The conditions which justified the issuance of exemptions to the air carriers for the use of the ITT Model 3544 DME are valid at the present time and are fully applicable to the Model VC-137C airplane. For approval in accordance with Section 4b.18, these conditions may be regarded as compensating factors pursuant to Section 4b.10 for lack of compliance of this equipment with TSO-C66 to the extent previously indicated. Therefore, the ITT Model 3544 DME installation in the VC-137C airplane may be approved subject to the type certification procedures of Parts 1 and 4b provided:

(1) Such equipment meets all of the minimum performance standards set forth in TSO-C66 with the exception of the requirements for receiver decoder selectivity contained in paragraph 2.11(b) of RTCA Paper No. 167-59/DO-99 which is incorporated in and is thus a part of TSO-C66;

(2) Such equipment is installed in accordance with the provisions of the airworthiness requirements applicable to the equipment concerned;

(3) The Airplane Flight Manual is amended by including a notation that approval of the equipment may be rescinded by the Agency at any time that the VOR/DME common system requires the refinement afforded by full compliance with TSO-C66; and

(4) The applicant is advised to modify this equipment at the earliest practicable date so it will meet the established standards for approval.

REVIEW CASE NO. 40 CERTIFICATION OF THE C-82A AIRPLANE UNDER THE PROVISIONS OF PART 9a AND SR-426 WITH A JET ENGINE AS STANDBY AUXILIARY POWER (Issued 18 August 1964)

1. INTRODUCTION

a. The C-82 airplane was originally certificated under Part 8, Airplane Airworthiness, Restricted Category. The applicant, Steward-Davis, Inc., altered the basic configuration of the airplane by installing an auxiliary jet engine, mounted on top of the fuselage. With the auxiliary engine used as standby power, approval was granted to increase the maximum permissible gross weight, and to carry cargo over congested areas. This approval was granted after a series of FAA flight tests in which the airplane demonstrated compliance with the critical performance sections of Part 42.

b. Steward-Davis, Inc., now wishes to certificate the airplane under the provisions of Part 9a, Aircraft Airworthiness; Surplus Military Aircraft, and SR-426. He proposes to use the jet engine as a source of standby power for performance credit.

2. REGULATIONS AFFECTED

a. CAR 9a, Aircraft Airworthiness; Surplus Military Aircraft, effective January 10, 1964. This part established the standards for civil type certification of surplus military aircraft of the United States in the normal, utility, acrobatic, and transport categories.

b. CAR 4b, Airplane Airworthiness; Transport Categories, effective August 25, 1955 (Part 4b as amended to December 31, 1953, including Amendments 4b-1 and 4b-2). This part established standards with which compliance must be demonstrated to be eligible for type certification in the transport airplane category.

c. SR-426, Performance Credit for Transport Category Airplanes Equipped with Standby Power; effective October 27, 1958. This part established standards which provide for granting allowable performance credit for transport category airplanes equipped with standby power.

3. HISTORY OF CASE

a. March 30, 1964, Steward-Davis, Inc., filed formal application for type certification of their C-82A airplane under the provisions of Part 9a and SR-426.

b. March 31, 1964, wire from Western Region, WE-400, to FS-100. The Western Region requested the criteria to be used to determine the performance credit sought by the applicant using an auxiliary jet engine as standby power.

c. July 16, 1964, letter from Steward-Davis, Inc., to FS-100. The letter requested the definition of the jet engine as either an auxiliary engine or a third primary engine.

4. FACTS IN THE CASE

a. The applicant wishes to have the airplane certificated as a twin reciprocating engine airplane, with a jet engine being considered as an auxiliary engine. He proposes to use this engine as standby power in the case of a main engine failure. Performance credit is being sought for the one-engine-inoperative takeoff flight path and the one-engine-inoperative climb conditions. In the en route climb condition, the applicant proposes to use the jet engine over an extended time period.

b. The applicant recommends that the initial power setting of the auxiliary engine in the takeoff and landing conditions be 70 percent r.p.m., which corresponds to approximately 28 percent of the engine's rated maximum available takeoff power. This power setting results in a static thrust of 910 pounds, or approximately 12 percent of the static thrust developed at the propeller of one of the reciprocating engines. The applicant feels that the increase in static thrust gained from the power setting of the jet engine prior to its actual use is not sufficient to consider this engine a part of the primary propulsive system.

c. The static thrust developed by the jet engine when operating at maximum available takeoff power is approximately 44 percent of the static thrust developed at the propeller of one of the reciprocating engines.

d. SR-426 defines standby power as the power and/or thrust derived from a rocket engine, applied for a short duration and in cases of emergency only. Due to the short duration of rocket thrust, no provisions are made for the one-engine-inoperative en route climb condition.

e. Provisions for standby power performance credit are not made in either Part 4b or Part 9a.

5. CONCLUSIONS

a. Because the static thrust of the jet engine is less than 50 percent of the static thrust developed by one of the reciprocating engines, when both are operating at maximum available takeoff power, it is concluded that the jet engine should be defined as an auxiliary engine. The C-82A should be certificated as a twin-reciprocating engine airplane equipped with a source of standby power.

b. The provisions of SR-426 are not considered applicable to this case for three reasons. First, SR-426 defines standby power as power derived from a rocket engine; the standby power for the airplane in question is derived from a jet engine. Second, standby power defined by SR-426 is applied for a short duration and in cases of emergency only; the applicant proposes to use standby power over a long period to augment the one-engine-inoperative en route climb condition performance and to augment the cruise capabilities if a main engine should fail during cruise, in addition to the emergency conditions of engine failure during takeoff and landing. Third, no provisions are made in SR-426 for the one-engine-inoperative en route climb condition which the applicant wishes to demonstrate with standby power.

c. Since SR-426 is not applicable to this case, and Parts 4b and 9a make no provision for standby power performance credit, special conditions, established under the provisions of CAR 4b.10, are specified below for the standby power performance credit demonstration.

d. The certification basis for the C-82A airplane should consist of the pertinent sections of Parts 9a, 4b and the special conditions mentioned in paragraph c. Section 9a.1(b)(1)(i) specifies compliance with CAR 4b effective August 25, 1955, (Part 4b as amended to December 31, 1953, including amendments 4b-1 and 4b-2). In the application of the special conditions, the power of the standby engine is considered to be equivalent to additional takeoff power for the takeoff flight path and the takeoff and approach one-engine-inoperative climb conditions, and to additional maximum continuous power for the one-engine-inoperative en route climb condition. The special conditions are as follows:

<u>General</u>

(1) The operation of the auxiliary jet engine should be safe and reliable.

(2) The overall level of performance should be equivalent to that intended by the CAR 4b requirements for conventional airplane designs.

(3) Full temperature and humidity accountability should be applied to the emergency power obtained from the auxiliary jet engine.

(4) Allowances should be made for such time delays in the performance and procedures as may be reasonably expected to occur in service.

(5) All performance and operating procedures necessary for the safe operation should be included in the airplane flight manual. The performance data should be arranged in the airplane flight manual to provide for full compliance with the operating rules.

Special Flight Items

(1) Takeoff

(a) The airworthiness takeoff climb performance, gear extended and retracted, may be determined with the auxiliary engine operating at the available takeoff power.

(b) The takeoff distance, and the takeoff flight path may be determined with the auxiliary unit operating, assuming a critical engine failure as prescribed in the regulations. The power setting of the auxiliary engine may be increased to the maximum available takeoff power upon the failure of one of the main engines.

(c) The accelerate-stop distance should be determined with the auxiliary engine operating at the initial power setting recommended by the applicant for the takeoff condition. Upon a main engine failure at the speed V_1 the power setting of the auxiliary engine may be reduced to idle.

(2) En Route

(a) It is assumed that the airplane will comply with the all engine en route climb condition performance requirements without the auxiliary jet engine operative. Performance credit may be granted for the one-engine-inoperative en route climb condition with the auxiliary engine operating at maximum continuous power. In this condition, with the auxiliary engine and one primary engine operating, the required rate of climb shall not be less than $.02V_{SO}^2$, in accordance with the formula $(.06 - \frac{.08}{N}) V_{SO}^2$, where N is the number of primary engines installed.

(b) Consideration must be given to operational fuel requirements and capacities during the en route phase of flight with the auxiliary engine operating.

(c) It should be possible to start the turbine engine at any altitude the airplane is expected to operate. This is to cover an engine failure after the en route condition has been reached and the auxiliary unit has been turned off. The time required to attain the maximum continuous power rating of the turbine engine, and the altitude lost during this time, should be entered in the airplane flight manual.

(3) Approach Climb

Performance credit may be granted for the approach climb condition with the auxiliary engine operating at the available takeoff power.

(4) One-Engine-Inoperative Go-Around

The auxiliary engine shall be operating prior to the demonstration at a power setting which will allow takeoff power to be attained readily. Procedures involving the use of the auxiliary engine during this maneuver should be entered in the airplane flight manual.

(5) Landing Distance

Since the auxiliary engine is operative during the approach, it should be operative during the determination of the landing distance.

(6) Flight Characteristics

The proposed use of the auxiliary engine is such that it could be operative during any flight regime. Therefore, the basic one-engine-inoperative flight characteristics, such as trim, stability, controllability, and stalling, should be checked with the auxiliary engine and one primary engine operating. In each case, the power setting of the auxiliary engine should correspond to the applicable section of Part 4b.

REVIEW CASE NO. 41 STATIC DIRECTIONAL STABILITY FOR TYPE CERTIFICATION OF LOCKHEED MODEL 382 (C-130E) AIRPLANE (Issued 21 August 1964)

1. ORIGIN AND PROBLEM

The Southern Region, in a memorandum dated June 30, 1964, advised that Lockheed-Georgia Company requested a determination of compliance with CAR 4b.157(c), static directional stability, as applicable to CAR 9a certification for its Model 382 (Military C-130E) airplane. Lockheed admits, and the Southern Region has confirmed, that the Model 382 fin stall condition does not comply with the requirements of CAR 4b.157(c). Lockheed contends that evidence of satisfactory military service experience of the C-130E airplane establishes compliance for this aircraft under CAR 9a. The Southern Region agrees, provided that FAA flight tests confirm the flight test data presented by Lockheed, and requests concurrence of their findings by FS-100.

2. REGULATIONS AFFECTED

a. CAR 9a.2(h) - Type Certification Requirements

In cases where the applicant has shown to the satisfaction of the Administrator, with respect to a particular aircraft being submitted for type certification, that strict compliance with a specific provision of this section would impose a severe burden on the applicant, the Administrator may accept such compliance as he finds will provide substantially the same level of airworthiness as is provided by the specific provisions of the requirements. In such cases, evidence of satisfactory military service experience may be considered in determining whether the level of airworthiness is substantially the same as that which would be provided by strict compliance with the specific provisions of the applicable requirements.

b. CAR 4b.157(c) - Static Directional and Lateral Stability

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

c. CAR 4b.10 - Eligibility for Type Certification

An airplane shall be eligible for type certification under the provisions of this part if ... the Administrator finds that the provision or provisions not complied with are compensated for by factors which provide an equivalent level of safety.

3. FACTS IN THE CASE

a. An extensive flight test investigation of the fin stall characteristics of the C-130E airplane has not been conducted since it is not considered by military and Lockheed-Georgia Company test pilots to differ appreciably from the C-130B airplane. The results of the limited flight test investigation on the C-130E airplane by Lockheed show:

(1) Compliance with portions of CAR 4b.157(c) up to sideslip angles of approximately 18° right and left. This sideslip angle is more than adequate for all normal flight conditions.

(2) Noncompliance with CAR 4b.157(c) in that above 18°, at low airspeeds in the approach and climb configurations, the variation of pedal force with sideslip angle is constant and at approximately 23° the pedal force reduces to zero.

(3) Depending on the configuration, fin stall can be experienced at all airspeeds below 180 KEAS. The stall is preceded by distinct fin and rudder buffet at approximately 18°, increasing in intensity up to 23° of sideslip. At this sideslip angle the buffet diminishes, and the sideslip angle will continue to increase at a moderate rate. Recovery from the condition is made by returning the rudder just slightly beyond neutral, which requires approximately 125 to 150 pounds pedal force, and by pushing forward approximately 75 pounds on the control column. In the approach configuration, approximately a 20-knot increase in airspeed and less than a 500-foot loss in attitude is experienced during the recovery from the extremely high sideslip (25 to 30 degrees) conditions. Recovery can be made from the 18 to 25 degree sideslip angle conditions in the approach configuration with negligible loss in airspeed

and altitude. Recovery can be made in the climb configuration from sideslip angles between 25 to 30 degrees with little or no change in airspeed and altitude.

b. Lockheed has studied two approaches for changing the fin stall characteristics. They are:

(1) To increase the fin area approximately 300 percent by installing a large dorsal. The costs involved in design, development, testing and manufacturing, however, would impose a severe burden on Lockheed and also potential buyers of C-130E series airplanes that may become surplus in the future.

(2) To use a vane type sideslip sensor and employing this signal, properly modified, to drive a hydraulic actuator located on the input side of the rudder booster. This actuator would provide a pedal force, linearly increasing with sideslip angle to approximately 180 pounds. The Southern Region feels that installation of the sideslip sensor would complicate the rudder control system thereby degrading the reliability of a system which is well substantiated by service history. Introduction of the sensor may produce undesirable effects, particularly in cases of malfunction in the augmented system.

c. Service experience shows that the present noncompliance of the Lockheed C-130E airplane with CAR 4b.157(c) has not resulted in any recognizable hazard to flight safety. The C-130 series airplanes have accumulated in excess of 1,200,000 flight hours in environments of all types with the loss of only seven airplanes, none of which were caused by fin stall condition. The conditions under which the military has operated the C-130 airplanes include normal flight operations, pilot training and check-out, and assault type landing and take-off operations. The C-130 airplanes have demonstrated excellent low speed handling characteristics during airdrop operations which involved conditions that provided a maximum exposure to fin stall. In the course of exploration and testing of this condition by both Lockheed-Georgia Company and the military pilots, all conceivable ways of getting into the fin stall condition have been investigated, resulting in the conclusion that fin stall will not be incurred inadvertently. This conclusion is further substantiated by the fact that over 3,060 military pilots have been trained in C-130 series airplanes without an incident attributable to the occurrence of fin stall.

d. The Air Force has conducted flight tests on the C-130E airplane and concludes in their report FTC-TDR-64-2, Limited C-130E Category II Stability and Control Tests that:

> "In general, the stability and control characteristics of the C-130E were satisfactory and similar to the C-130B.
>
> "Compared to the C-130B, the C-130E had reduced static lateral-directional stability. The C-130E required approximately 20 percent less rudder deflection to obtain a [one] degree of sideslip than the C-130B; however, this did not result in any adverse handling characteristics. A rudder overbalance which occurred at low speeds and extreme sideslip angles (20 degrees) was preceded by moderate airframe buffet. This buffet was considered adequate warning to prevent encountering the over-balance condition. Rudder overbalance was readily overcome by neutralizing the rudder, returning the wings to level, lowering the nose to increase airspeed and decreasing power."

e. The fin stall characteristic of the C-130E airplane was demonstrated to personnel of SO-210 and FS-160 on two separate flights covering a range of weights from maximum take-off to maximum landing at an intermediate c.g. on both flights. These flights were not adequately controlled or instrumented to serve as official flight tests, but did serve well as demonstrations.

f. On the basis of available data from Lockheed and the FAA demonstration flights, this airplane is considered by the Southern Region, to meet CAR 4b.157(c) up to sideslip angles appropriate to the airplane type at the loadings demonstrated. This airplane does not meet CAR 4b.157(c) at greater sideslip angles which are attainable with full rudder control. During FAA demonstration flights, fin stall was experienced in a power-on climb condition and only in a right sideslip. Fin stall was not encountered in sideslips to the left, power on or power off. During the FAA demonstration flights with SO-210 personnel, simulated landing approaches were made with full rudder sideslips to right and left with approach and landing flaps, and steep and flat approaches, with sufficient power to stabilize 300 feet/minute rate of descent. In none of these instances were fin stall characteristics encountered. These in-flight demonstrations did not duplicate the most adverse conditions of weight, c.g., etc. Confirmation of the Lockheed flight test data is anticipated during official FAA flight test.

g. Outside of military service experience, Lockheed has referenced no other compensating factors to establish that the C-130E airplane provides an equivalent level of safety under the provisions of CAR 4b.10, nor have any been found by the Southern Region. The C-130E airplane, therefore, cannot be considered to comply with this part. However, the military service experience (see par 3c) may be used to show that substantially the same level of airworthiness exists for type certification under CAR 9a when a severe burden would be imposed to show compliance with CAR 4b.157(c) (see par 3b).

4. CONCLUSIONS

a. We conclude that the design, installation, and maintenance of a rudder force device or, the 300 percent increase in fin area, to establish strict compliance with CAR 4b.157(c) impose a severe burden on the applicant and the expense is unjustified relative to the potential gain in airworthiness.

b. We conclude that satisfactory military service experience has established that the present noncompliance with the directional stability requirements of CAR 4b.157(c) has not resulted in any hazard to flight safety and that a level of airworthiness exists which is substantially the same as that which would be provided by strict compliance.

c. We conclude that the C-130E airplane does not have compensating factors to comply with the type certification equivalent safety requirements of CAR 4b.10 but that satisfactory service experience exists which makes the C-130E airplane eligible for type certification under CAR 9a, provided that the Southern Region confirms by official flight tests that the model does not comply with CAR 4b.157(c).

REVIEW CASE NO. 42 EFFECT OF ENGINE UNBALANCE (Issued 25 August 1964)

1. INTRODUCTION

a. The Southwest Regional Office has requested that Aero Commander show compliance with the provisions of Sections 4b.401(c) and 4b.606, Civil Air Regulations, by substantiating the structural integrity of the Model 1121 Jet Commander when subjected to the effects of unbalance following the assumed failure of at least three rotor blades in an axial segment for the full length of the compressor and that this condition be investigated through the transient speed range from maximum operating engine r.p.m. to maximum windmilling speed as well as the range of windmilling speeds.

b. Aero Commander Division, Rockwell-Standard Corporation, has requested through our Fort Worth Regional Office that the case be reviewed, contending that compliance with Section 4b.401(c), CAR, requires substantiation of structural integrity at windmilling engine r.p.m. only and, further, that the provisions of Section 4b.606 are not applicable to this condition.

2. CHRONOLOGICAL HISTORY

a. Western Regional Aircraft Engineering Division, WE-400, memorandum dated October 31, 1962, to Southern Regional Engineering and Manufacturing Branch, SO-210, concerned the effects of jet engine rotor unbalance at windmilling r.p.m. on vibration and flutter and the amount of rotor unbalance assumed by West Coast manufacturers, including Boeing and Douglas. This unbalance was in the order of three lost blades per stage in an axial segment for the full length of the compressor.

b. WE-414 memorandum dated August 22, 1963, to SO-210, concerned turbofan rotor unbalance accepted in past type certification programs and described the number of blades assumed removed in substantiation of the Pratt & Whitney JT3D-1 engine in the Boeing 707 and Douglas DC-8 aircraft. The unbalance used was in the order of three blades per stage in an axial segment for the full length of the engine. Douglas also investigated effects of unbalance due to other combinations of failed blades.

c. Aero Commander letter of December 20, 1963, to Southwest Regional Engineering and Manufacturing Branch, SW-210, submitted information from General Electric on factory and field experience with respect to the effects of failed buckets or blades and concluded that structural integrity would not be adversely effected by the degree of failure likely to occur.

d. Southwest Regional Airframe and Equipment Section, SW-212, letter of January 24, 1964, to Aero Commander, advising that the range of speed from full r.p.m. to windmilling r.p.m. be used in the analysis for rotor unbalance and suggesting that seven adjacent blades in each stage of the engine be considered as failed.

e. Aero Commander letter of February 27, 1964, to SW-210, reviewed the provisions of Section 4b.401(c), CAR, and pointed out that there was no mention in these provisions or applicable policy material of speed applications in excess of windmilling for which structural integrity should be determined.

f. SW-212 letter of March 10, 1964, to Aero Commander, advised that Section 4b.606(b) as well as Section 4b.401(c) applied to the condition of engine rotation and that engine unbalance at speeds above windmilling r.p.m. should be evaluated since a hazard to the aircraft could occur at high speeds; also, suggested that the analysis be based on seven blades lost in all rows of the compressor and turbine, pointing out that while this may not be realistic from the standpoint of actual operation of the engine it was, nevertheless, considered conservative.

g. Aero Commander letter of March 20, 1964, to SW-210, strongly objected to the application of Section 4b.606(b) to the condition of engine rotation covered by Section 4b.401(c), insisting that Section 4b.606(b) was not applicable to the engine installation and requesting that an exacting review be made by the Regional Office of the provisions of Section 4b.401(c) with regard to the effects of rotor unbalance on structural integrity of the airplane so that this controversial issue may be resolved.

h. SO-210 memorandum to SW-210 dated April 3, 1964, discussed the Lockheed C-141A turbofan blades unbalance analysis and enclosed the Convair CV-990 reports. SO-210 advised that assurance has been given that the Lockheed-Georgia Corporation will substantiate the structure to the criteria presented at the preliminary type certification board meeting based on ground resonance data.

i. Southern Regional Engineering and Manufacturing District Office at Atlanta, Georgia, SO-EMDO-42, memorandum of April 14, 1963, to Southern Regional Staff Engineer, Propulsion, SO-214, presented a comparison of rotor unbalance analysis for the C-141A with that for Convair 990 aircraft and described the method of analysis used by Lockheed-Georgia Corporation in determining the frequency of resonant wing modes and the correlation with flutter model modes.

j. SW-212 letter of April 16, 1964, to Aero Commander, advised that the interpretation of the requirements and reasons therefor, as presented in SW-212 letter of March 10, 1964, (item f above), were valid and suggest that Aero Commander make this matter the subject of a review case if it does not wish to comply.

k. Aero Commander letter of April 24, 1964, to SW-210, objected to the provisions of Section 4b.606 being applied to the engine, stating that the issue at hand concerns Section 4b.401(c), CAR, which was interpreted to mean that only when provisions to completely stop rotation of turbine engines were not provided, the effects of continued engine rotation, either windmilling or controlled, on structural integrity would need to be substantiated; therefore, if Aero Commander had chosen to install a means to brake the engine after shutdown, no investigation of this type as suggested by the Fort Worth Office would be required.

l. Aero Commander telegram of April 24, 1964, to Engineering and Manufacturing Division in Washington, FS-100, advised that a review case had been requested from SW-210.

m. FS-100 telegram to Aero Commander of April 29, 1964, acknowledged the telegram of April 24, 1964, and requested a copy of Aero Commander letter of April 24, 1964, to SW-210.

n. Aero Commander letter of April 30, 1964, acknowledged receipt of item m above and enclosed a copy of the April 24, 1964, letter to SW-210.

o. SW-210 memorandum of May 8, 1964, to FS-100, reviewed the background and summarized both Aero Commander and SW-210 positions on the matter.

p. FS-100 telegram of June 24, 1964, to SO-210, advised that review case was being prepared stating (1) that Section 4b.401(d) does not require substantiation at speeds above windmilling speed, and, (2) that Section 4b.606 is not applicable to the engine but to installation components, equipment, systems and installations.

q. SW-210 telegram of July 8, 1964, to FS-100, indicated a possible misunderstanding of the problem and stressed that uncontrollable conditions of aircraft due to dynamic engine unbalance was being considered. SW-210 considers Section 4b.606, CAR, applicable to the installation; however, if not, Section 4b.10 should be applied.

3. <u>FACTS IN THE CASE</u>

a. Aero Commander has agreed to investigate the effects of a windmilling unbalanced engine on the aircraft structure. It is the company's contention, however, that since Section 4b.401(c), CAR, specifically refers (through Civil Aeronautics Manual 4b.401-3(a)) to continued rotation if a rotor brake is not provided, only windmilling speeds need be investigated. In further support of this contention, Aero Commander affirms that if a means were provided to stop the engine or reduce windmilling r.p.m. below 400 r.p.m., no investigation of the effects of an out-of-balance engine, either at windmilling r.p.m. or any other engine speed, would be necessary to comply with Section 4b.401(c). The company further asserts that Section 4b.606 is applicable to only those systems which are referred to in the manual material of Subpart F, Equipment, and is neither related nor applicable to engine operation.

b. Southwest Region considers that:

(1) Subpart E of Part 4b, CAR, is no more self-sufficient than is the Subpart D, Design and Construction, or Subpart F, Equipment. If a failure occurs, therefore, the conditions of failure exist from the time of failure until the airplane is landed. The results of such failures must be examined and a determination made of the effect on the airplane. As Section 4b.606, CAR, states that "all equipment, systems, and installations shall be designed to safeguard against hazards to the airplane in the event of their malfunctioning or failure," this regulation is considered by the region to apply to any equipment, system, or installation which might become a hazard to the aircraft in the event of failure or malfunctioning.

(2) Section 4b.401(c) does not use the word "continued" rotation as Aero Commander contends. Only Section 4b.401-3, CAM, used that term. Since CAM material seldom is prepared to cover the entire regulations to which it refers, manual material cannot be used to govern a regulation. Section 4b.10, CAR, also requires that no hazards exist.

(3) Section 4b.652, CAR, Engine-Driven Accessories, and Section 4b.659, CAR, Equipment Incorporating High Energy Rotating Parts, were rescinded by Amendment 4b-12 on the basis that Section 4b.606, CAR, is concerned with the reliability of all equipment, systems, and installations (Reference Page 3, Amendment 4b-12). This preamble to Amendment 4b-12 specifically relates Section 4b.606 to engine-driven accessories for which requirements also exist in Subpart E of Part 4b, CAR. If Section 4b.606 applies to parts of the powerplant installation, specific extension of its provisions are not considered necessary for it to be applicable to other parts of the powerplant installation.

(4) The Convair Model 30 was investigated for conditions very similar to those which Aero Commander was requested to investigate. Convair Report DF-30-161 reads ".....transient and

steady state loads. The transient loads are assumed to occur at 100 percent r.p.m. and should be considered limit loads." Both the Lockheed Model 300 (C-141) and Model 1329 were analyzed for the r.p.m. range from 100 percent to windmilling.

4. ANALYSIS

a. Section 4b.401(c) of the Civil Air Regulations, among other things, states that means shall be provided for individually stopping and restarting the rotation of any engine in flight, except that for turbine-engine installations means for stopping the rotation need be provided only if such rotation could jeopardize the safety of the airplane.

b. The Federal Aviation Agency policies which apply to this regulation are set forth in Section 4b.401, CAM. In this section, it is stated that if means to stop completely the rotation of the engine are not provided, it should be shown that continued rotation of the engine either windmilling or otherwise controlled will not cause powerplant structural damage which might adversely affect other engines or the aircraft structure, flammable fluid to be pumped into a fire or ignition source, or a vibration mode which might adversely affect the aerodynamic or structural integrity of the airplane. Engine rotor speeds under 400 r.p.m. are not required to be investigated.

c. It is clear that, in showing compliance with Section 4b.401(c), CAR, and following the guidance material applied to it, no investigation of the effects of engine vibration at any speed are required by this section of the regulations if a means are provided for stopping the rotation of the engine. In such a case, no investigation of vibration effects would be required for the period of engine deceleration. There is no rational basis upon which to conclude that the hazard to the aircraft during this transient period is altered by presence or lack of a means for bringing the engine to a complete stop. It must be concluded, therefore, that this requirement to investigate possible hazards associated with continued rotation on account of not having provided a means for stopping rotation of the engine does not also require a similar investigation of the hazards associated with the transient condition of engine deceleration nor was it intended to do so.

d. Section 4b.606(b) requires that all equipment, systems, and installations shall be designed to safeguard against hazards to the airplane in the event of their malfunctioning or failure.

e. It is noted that Section 4b.606 is in Subpart F, Equipment. This subpart does not apply to engines. This conclusion is supported by the circumstances that Subpart E is devoted to the powerplant, that the engine is not listed as an item of required basic equipment under Subpart F, and that a number of the general requirements of Subpart F repeat, in substance, the requirements that are applied to the powerplant installation under Subpart E.

f. It is appreciated that there are areas wherein there may be some uncertainty as to whether the provisions of Subpart E or Subpart F, or both, apply to a particular component; but it is clear that this uncertainty does not exist for the engine.

g. Considering the foregoing, it is considered that Section 4b.606(b) does not provide a basis for requiring a determination that rotor unbalance of the engine during the transient condition between operating and windmilling r.p.m. will not cause a hazardous condition.

h. The question of the effect of engine rotor unbalance was considered in "A Report by the CAA Turbine-Powered Transport Evaluation Team" dated January 1954. In this report, only windmilling and controlled r.p.m. were discussed as representing areas wherein design consideration would have to be given; there was no indication of intent to consider higher r.p.m.

5. CONCLUSIONS

In consideration of the foregoing, it is concluded that:

(1) Section 4b.401(c), CAR, requires substantiation only at windmilling r.p.m. or controlled r.p.m. of a shutdown turbine if such speeds are in excess of 400 r.p.m. If means are provided to completely stop rotation of a turbine engine, Section 4b.401 does not require such substantiation.

(2) Section 4b.606, CAR, is not applicable to the basic engine which is certificated under Part 13, CAR, and is not a piece of equipment.

(3) The Aero Commander interpretation of Section 4b.401(c), CAR, and Section 4b.401-3, CAM, is essentially in accord with the conclusions stated in (1) and (2).

REVIEW CASE NO. 43 MOONEY AUGMENTED LATERAL STABILITY SYSTEM
(Issued 2 September 1964)

1. ORIGIN AND PROBLEM.

a. Mooney Aircraft, Incorporated, has proposed to install a wing leveling system as required equipment on their 1965 models. This system will be called "Augmented Lateral Stability System (ALS)." The announced purpose of this system is to help prevent accidents resulting from unskilled pilots, i.e., noninstrument qualified pilots, being caught inadvertently in bad weather. The system will maintain a wings level attitude continuously unless directed otherwise by the pilot. Since it is designed for continuous duty, Mooney has requested that compliance with the lateral stability requirements of Civil Air Regulations, Part 3.118, Directional and Lateral Stability be demonstrated with the device "ON". The Southwest Region believes the inherent aerodynamic lateral stability should be positive with the device "OFF."

b. There is a lack of guidance policy for such systems under CAR 3. The Southwest Region therefore requested a review case to resolve the problems.

2. REFERENCE REGULATIONS.

a. CAR 3.118 - Directional and Lateral Stability.

Para. (a)(2). Static lateral stability, as shown by the tendency to raise the low wing in a sideslip, shall be positive for all landing gear and flap positions with symmetrical power up to 75 percent maximum continuous power at all speeds above 1.2 V_{s1} up to the maximum permissible speed for the configuration investigated but shall not be negative at a speed of 1.2 V_{s1}. The angle of sideslip for these tests shall be appropriate for the type of airplane and in no case shall a sideslip be less than that obtained with 10 degrees of bank.

3. CHRONOLOGICAL HISTORY.

a. April 9, 1964 During a meeting with the Southwest Region, Mooney proposed to install an ALS system on their 1965 models.

b. May 19, 1964 The Southwest Region forwarded a memorandum to Washington requesting a review case to establish guidance policy for an ALS system certification.

4. RELATED BACKGROUND.

a. The description of the ALS system is quoted from the minutes referred to in paragraph 3.a. of this review case. "The ALS means Augmented Lateral Stability and is installed, according to Mooney, to help prevent accidents when unskilled pilots inadvertently get into bad weather. It will be able to fly the airplane 'hands-off'; that is, straight and level except for pitch. (Similar to roll axis autopilot but continuous duty.) It cannot be turned off permanently as long as the vacuum system functions. There is a release on the left hand grip of the pilot's control wheel which operates a spring loaded valve. When the push button is actuated, it requires two seconds for the vacuum to 'bleed-off.' In this event the airplane is operated without ALS. The ALS system is pneumatically operated from the vacuum system and is established by means of an inclined axis turn and bank gyro. This ALS system will be standard equipment except for the Model M20D. As a follow-on program, they will include a heading select feature and an omni-coupler. Since the system is operated pneumatically there will be no dependence on the electrical system and consequently they (Mooney) feel this will be a very dependable system. At the present, the airplane is using a Tact Air component although Brittain equipment is being investigated. There will be no conventional command control. It will be controlled within 90° when the heading select feature is put into the system. This command is accomplished by means of adjusting the directional gyro. Later on they will install pitch control. All the actuators will be of the pneumatic type."

b. The ALS system is designed to operate with low control forces. The pilot may override the system at anytime by applying control effort about double the amount associated with the procedure of releasing the ALS system with push button on the pilot's control wheel.

c. There is no guidance policy in CAR 3 pertaining to the acceptance of equipment installations such as the ALS system for showing compliance with the stability requirements.

d. Automatic devices have been employed in many models to meet flight characteristics requirements. Such devices range from simple bungee springs to sophisticated automatic stabilization equipment installations. To date, for civil certification, the devices have been used to meet requirements involving stability, control, trim, stall warnings, and stalls.

5. DISCUSSION.

a. For safe operation, it is essential that lateral stability be provided. This may be accomplished in more than one way. For example, an automatic and reliable device could be used to provide apparent stability by maneuvering the airplane back to a desired attitude following an upset. The use of any automatic device is subject to malfunction or failure hazard that is in inverse proportion to the degree of reliability provided. Hazardous action of such devices must be guarded against.

b. The concept of using auxiliary devices to meet the required flight characteristics has been acceptable to date for stability, control, trim, stall warning and stalling. A device that provides the required lateral stability characteristics is consistent with this concept.

c. If the concept is acceptable to use an automatic device to meet the lateral stability requirements, then it becomes necessary to determine what compensating factors may be considered to provide the same level of safety that is provided by meeting the stability requirement without using an automatic device. It would be necessary too, that in addition to providing an adequate level of apparent lateral stability, a fault and reliability analysis be made to show that the device is highly reliable.

d. The use of any automatic or powered device for showing compliance with the flight characteristics requirements must be investigated for structural integrity, effects of malfunctions or failure, reliability, and evaluated by flight tests to determine that the installation performs its intended function as a required item for type certification.

e. Required equipment installations are essential to the safe operation of the airplane. When required equipment fails or malfunctions, the airplane no longer complies with one or more of the regulatory requirements. A fault analysis or other means is required to assure that the probability of failure is remote. The use of redundant installations must be considered if a malfunction or failure renders the airplane uncontrollable or otherwise produces an unsafe condition.

f. Mooney has proposed that the lateral stability requirements of CAR 3.118 be demonstrated with the ALS system "ON" only. On this basis, the system must be considered required equipment and would preclude determining the inherent lateral stability characteristics with the system inoperative. The ALS system must therefore have a very high degree of reliability.

g. The ALS system is proposed for lateral mode only and any aileron deflection to "pick-up a wing" will result in an adverse yawing moment. At slow speeds and high angles of attack, relatively greater aileron deflections are required to provide a constant rolling moment. In this case the adverse yawing moment is increased and could produce a spin inducing condition. It must be demonstrated by flight test that the airplane can safely be stalled and normal stall recoveries made with the ALS system continuously engaged.

h. If the fault analysis shows that there is a chance for malfunction, tests of any likely malfunction should be tested in flight. Malfunctions must not result in control forces of such magnitude as to interfere with the pilot's immediate override of the ALS system. The airplane must have flight characteristics with an ALS system malfunction or failure such that a controlled safe descent and landing can be made without exceptional piloting skill. Turbulent air conditions should be considered when conducting these tests.

6. CONCLUSIONS.

a. We conclude that the Mooney proposal to install an ALS system and show compliance with the lateral stability requirements of CAR 3.118 with the ALS system "ON" only is acceptable subject to the following:

(1) The ALS system becomes an item of required equipment for type certification.

(2) The equipment and system are designed and installed to insure that they perform their intended function reliably under reasonably foreseeable operating conditions, including expected environmental effects.

(3) The equipment, system, and installation are designed to safeguard against hazards to the airplane in the event of malfunction or failure.

(4) A power failure indicator for the ALS system is provided.

(5) A means is provided to check the ALS system for proper functioning prior to flight.

(6) The flight characteristics of the airplane are such that a controlled safe descent and landing can be made without exceptional pilot skill in the event of ALS system malfunction or failure. IFR and turbulent air conditions will be considered.

(7) The flight characteristics are such that the airplane can be safely maneuvered when the release button on the pilot's wheel is depressed and the ALS system is rendered inoperative.

(8) The related operating procedures, together with any information concerning the airplane found necessary for safety during operation of the required ALS system, are to be included in the airplane flight manual, expressed as markings or placards, or made available by such other means as will convey essential information to the operator and pilot.

REVIEW CASE NO. 44 DOUGLAS AIRCRAFT COMPANY PROPOSAL FOR FOLDING ARMRESTS TO CLEAR TYPE III EXIT AREA (Issued 9 September 1964)

1. INTRODUCTION

The Douglas Aircraft Company has requested, by letter to the Chief, Engineering and Manufacturing Division dated May 14, 1964, that a review be made of their proposal to prevent the passenger seat arms at the overwing exits of the Model DC-9 from obstructing the exit. The Western Region had considered the Douglas proposal and by letter dated May 4, 1964, informed Douglas that the proposal was not considered to fulfill the requirements of CAR 4b.362(g) and CAR 4b.362(e)(1). Douglas has endeavored to make the seating arrangements of the DC-9 as flexible as possible, and has attempted to design the seats so that it would not be necessary to install particular seats at the overwing exits. In order to meet the stepdown distance of the Type III exits, the exit sill is so low that the outboard seat arms project about six inches into the exit area. Douglas, therefore, proposes to design all outboard seat arms so that they can be folded upwards and thereby retract out of the way. Douglas feels that they have provided equivalent safety and requests a ruling.

2. CHRONOLOGICAL HISTORY

a. The Douglas Aircraft Company in a letter to the Western Region dated March 3, 1964, proposed to use passenger seats on their Model DC-9 in which the outboard armrests would be hinged to enable them to be retracted. This was proposed in lieu of mounting armrests on the emergency overwing exit hatches, as this would require special seats without outboard armrests at the overwing exits.

b. The Western Region in their reply to Douglas dated May 4, 1964, stated that they considered the Douglas proposal did not fulfill the requirements of CAR 4b.362(g) and CAR 4b.362(e)(1).

c. The Douglas Aircraft Company, in a letter dated May 14, 1964, asked the Chief, Engineering and Manufacturing Division for a ruling.

d. The Western Region was asked by wire dated May 25, 1964, for their comments on the Douglas proposal. A followup request was dispatched June 24, 1964.

e. By memorandum dated July 9, 1964, the Western Region explained their reasons for considering the Douglas proposal unacceptable.

f. By memorandum dated July 20, 1964, FS-120 asked the Western Region whether or not the DC-9 may have any compensating features tending to decrease the time required for evacuation. Such features as door mounted slides or increased seat spacing at overwing exits might be considered as compensatory. Western Region stated that the subject had again been reviewed with Douglas and the latter offered no further compensating factors.

3. FACTS IN THE CASE

a. Douglas has spent considerable effort to make the DC-9 as versatile as possible with respect to the various possible passenger seating arrangements with the minimum expenditure of labor on the part of the operator. One operator has specified that it should be possible to make a complete change in passenger seating arrangement during a turn around of 30 minutes time. One of the features contributing to this versatility is the avoidance of a requirement for special seats at the overwing window exits.

b. When a passenger seat is opposite the Type III exit in the DC-9, the seat armrest projects approximately six inches into the exit area. Douglas, therefore, proposes to design all seat armrests except the one nearest the aisle, such that they will fold up and stow quite easily. It is hinged and automatically stows when folded. Douglas proposes that the instructions for opening the Type III exits would read, "Lift armrest - pull handle." The exit can be opened, however, without regard to the sequence of these actions. Furthermore, the outboard side of the outboard armrest of each seat will be marked with the word "Lift" in large letters so that it could be seen by rescue personnel working from the outside.

c. Douglas takes the position that their proposal is no worse than our present policy of allowing seat backs to project into the exit area provided they can be pushed forward to clear the area without the use of a release or catch.

d. CAR 4b.362(g) states in part that access shall be provided from the main aisle to all Type III and Type IV exits and such access shall not be obstructed by seats, berths, or other protrusions to an extent which would reduce the effectiveness of the exit, except that minor obstructions shall be permissible if the Administrator finds that compensating factors are present to maintain the effectiveness of the exit. This is further clarified in the Civil Aeronautics Manual 4b.362-6. Paragraph (c) explains the policy of allowing seat backs to project into the exit area provided they can be pushed forward to clear the area. It further states that "a clear opening should permit the required minimum exit shape to be projected inward past the seat bottom and back cushion. Minor protrusion of the seat upholstery is acceptable if it does not interfere with exit removal and if it could be compressed without special effort by the person(s) using the exit." CAM 4b.362-6(d) states that "armrests, curtains, or other protuberances should not restrict the required minimum opening unless they are removed simultaneously with opening of the exit."

e. Douglas offers as compensating factors the following:

(1) The exit may be opened from inside or outside without regard to the sequence or folding or the position of the armrest.

(2) The exit opening instructions inside the airplane will read, "Lift armrest - pull handle," thus requiring two movements on the part of the person attempting to open the exit.

(3) The exit opening instructions on the outside will be augmented by having the word "Lift" on the outboard side of the armrest adjacent to the exit opening.

(4) The armrests are easily folded without the use of latches, etc.

(5) The armrests are automatically stowed, when folded, because they are hinged, thus they cannot beome missiles or stumbling hazards in an emergency such as might be the case with improperly installed armrests of the plug-in type.

(6) The hinged armrests preclude difficulties which could arise if the armrests were attached to the exit hatch where there may be more than one position for it to be bolted, depending on the particular seating configuration being used.

4. CONCLUSIONS

a. Today more and more effort is being made to simplify the process of opening and using exits. High density loadings make it imperative to ensure that exit areas remain unobstructed. Present requirements and existing published policy ensure that seat armrests cannot obstruct this area because they are required to be removed simultaneously and automatically with the opening of the exit. The Douglas proposal does not do this and, therefore, does not offer equivalent safety.

b. Douglas does not offer any compensating factor which would tend to reduce the evacuation time sufficiently to offset the extra action necessary to clear the exit opening. It is, therefore, concluded that the Douglas proposal of a folding armrest on passenger seats to clear access to the Type III exits is not satisfactory.

REVIEW CASE NO. 45 SPERRY SP-40 AUTOPILOT INSTALLATION ON THE GRUMMAN G-159 AIRPLANE (STC SA2-931)
(Issued 18 September 1964)

1. INTRODUCTION

a. During flight tests conducted at the National Aviation Facilities Experimental Center, Atlantic City, New Jersey, on the Grumman G-159 airplane to evaluate equipment changes incorporated into an existing STC (the Sperry SP-40 autopilot), the autopilot was found to respond to false course signals of the glide slope while flying at constant altitude utilizing localizer frequency for navigation with the autopilot operating on RADIO mode.

b. The Southwest Region took the position that the autopilot response to these ambiguous glide slope signals was in non-compliance with Civil Air Regulations 4b.612(d)(5) and further contended that possibly the autopilot could not meet Technical Standard Order C9b requirements. Accordingly, the Western Region and Sperry were notified of this contention. The Western Region disagreed with the Southwest Region and since the Western Region is the controlling region for the Sperry SP-40 autopilot, regarding compliance with the TSO, no corrective action was taken against Sperry. The Southwest Region altered its position and allowed approval of the installation by prohibiting the use of RADIO mode while utilizing the localizer frequency for navigational purposes, by appropriate limitations in the Flight Manual Supplement. The Southwest Region then called this problem to the attention of the Washington Office for review.

2. CHRONOLOGICAL HISTORY

a. November 30, 1960; the Southwest Region issued Type Inspection Authorization A543-2 to Associated Radio Service Company, Dallas, Texas, covering the Sperry SP-40 autopilot installation on the Grumman G-159 airplane. Subsequent ground and flight inspections revealed no unsatisfactory conditions and approval was granted by issuing STC SA2-931.

b. August 8, 1962; Sperry made application with the Southwest Region to amend STC SA2-931 to incorporate equipment changes to the SP-40 autopilot. The Southwest Region agreed that these changes could be incorporated into the STC, subject to satisfactory flight tests of the installation incorporating these equipment changes.

c. October 8, 1962; Results of tests conducted at National Aviation Facilities Experimental Center on the subject installation revealed autopilot response to ambiguous glide slope signals while utilizing the localizer frequency for navigational purposes. The response resulted from the lack of means in the autopilot control head design to decouple the glide slope frequency while utilizing the localizer. The Southwest Region felt that this condition was hazardous from a passenger safety standpoint, although no critical flight loads or adverse attitudes were encountered. Based on this, the Southwest Region notified the Western Region and Sperry that they considered the installation in noncompliance with CAR 4b.612(d)(5) and possibly that the autopilot could not conform to the TSO requirements. As a result, approval of the STC revision was withheld.

d. August 13, 1963; The Western Region advised the Southwest Region by memorandum that they did not agree with their position and that the Western Region considered CAR 4b.612(d)(5) and the TSO satisfied. The Western Region indicated that the TSO has no requirement that an autopilot be capable of discriminating between localizer and glide slope frequencies and further that auxiliary controls are defined in the TSO, and that the SP-40 autopilot contains the proper disengage functions in its design. The Western Region also pointed out that although the approach function, which the Southwest Region desired to have incorporated into the autopilot might be desirable, it is not specifically required by regulation. Upon receipt of this information, the Southwest Region relaxed its position and allowed approval of the installation with supplemental flight manual limitations prohibiting the use of RADIO mode during certain flight operations.

e. September 20, 1963; The Southwest Region brought the problem to the attention of the Washington office. The Region outlined the problem action taken, and the disagreement which resulted during the approval. The region requested Washington review and concurrence of their action.

f. October 8, and October 28, 1963; The Maintenance Branch, FS-300, concurred with the Southwest Region's action. (Route Slips)

g. October 9, 1963; Memorandum from the Flight Test Branch, FS-160, to the Airframe Branch, FS-120 concurs with the installation approval.

h. October 11, 1963; Memorandum from the Operations Division, FS-400, to the Engineering and Manufacturing Division, FS-100, recommending that all autopilot control heads be required to have an Approach selection

i. December 3, 1963; Memorandum from the Western Region to FS-100, reiterating their disagreement with the Southwest Region's contention that the autopilot is in noncompliance with CAR 4b.612(d)(5) or does not conform to TSO-C9b.

j. December 23, 1963; Memorandum from FS-100 to the Engineering and Manufacturing Field Extension, FS-968 (FS-190), transmitting background material on the subject for evaluation and preparation of a review case. (Material delayed enroute, received FS-190, 12:30, March 13, 1964.)

k. February 5, 1964; Memorandum from FS-100 and FS-190 requesting current status of the review case.

l. February 7, 1964; Memorandum from FS-190 to FS-100 advising that the subject material has not been received by this office, and requested copies of material so that request for preparation of review case can be complied with.

m. March 5, 1964; Memorandum from FS-100 to FS-190, transmitting appropriate material for preparation of review case.

n. March 23, 1964; The Southwest Region was called by FS-190 regarding the case, to determine the magnitude of the loads and attitudes encountered during the tests. The loads were well within the airplane design envelope and no adverse attitudes were encountered.

o. March 24, 1964; Conference with Electronic Engineers acquainted with SP-40 autopilot operation. The problem in this case derives from the ground facility radiating ambiguous glide slope signals. The response of the SP-40 autopilot to these false signals when operating on RADIO mode is a normal function of the autopilot.

p. March 26, 1964; Covering memorandums from FS-190 to FS-100, transmitting the review case for their further action.

q. April 14, 1964; Memorandum from FS-100 to FS-190, returning the review case with comments to be incorporated, where appropriate, into finalized review.

r. May 1, 1964; Revised review case forwarded to Washington for their action and publication.

s. May 22, 1964; Memorandum from FS-100 to FS-190 returning the review case to incorporate regional limitations placed on installation and regulatory basis for such limitation. The comments recommend expanding on the facts in the case and revising the conclusions accordingly.

3. FACTS IN THE CASE

a. The Southwest Region withheld approval of equipment changes to the SP-40 autopilot due to the autopilots response to ambiguous glide slope signals. After corresponding with the region controlling the TSO, the Southwest Region relaxed its position and allowed approval with flight manual limitations.

b. The limitations in the Airplane Flight Manual placed on the installation by the Southwest Region are as follows:

(1) Limitations: "When flying holding patterns based on localizer frequencies or navigating by use of localizer frequencies, do not use RADIO mode."

(2) Operation of Autopilot - Automatic Approach (add this statement; preface to normal procedures): "The following instructions cover the operation of the autopilot during a coupled approach, and applies any time the aircraft is at or below the normal glide path signal."
(This statement added.)
"When navigating by use of the localizer frequencies or flying holding patterns based on localizer frequencies, it is possible to be physically located with respect to the glide path such that ambiguous glide path signals may be received. If the autopilot is coupled to RADIO mode, these ambiguous signals will cause undersirable response in the autopilot of considerable magnitude. For this reason, use either 'hdg. Sel.' mode or basic Turn Knob operation when navigating or holding by use of localizer frequencies."

c. The SP-40 autopilot does not incorporate in its design a separate glide slope engage function and lacks means in the autopilot control head design to decouple the glide slope frequency while utilizing the localizer. As a result, if ambiguous glide slope signals are received, it is a normal design function for the autopilot to respond to these signals.

d. The regulations or the TSO do not stipulate what markings should be contained on the autopliot control head and further, do not stipulate that the autopilot should be capable of discriminating between localizer and glide slope frequencies when operated in the coupled mode. Although such a mode of operation may be desirable, it is not required by present standards.

e. CAR 4b.612(d)(5) states, "When the automatic pilot integrates signals from auxiliary controls or furnishes signals for operation of other equipment, positive interlocks and sequencing of engagement shall be provided to preclude improper operation. Protection against adverse interaction of integrated components resulting from a malfunction shall be provided." Auxiliary controls are defined in the Aeronautical Standard 402a, paragraph 4.5.4, of TSO-C9b. Adequate interlock provisions are incorporated into the SP-40 autopilot design.

f. The autopilot is an optional equipment item, and not required for airplane operation. If it is found that certain operational functions of the optional item jeopardizes the safety of flight, operating limitations and other information found necessary for safety shall be included in the AFM. (Ref. CAR, Section 4b.700(b)). When neither a specific hazard nor a characteristic jeopardizing safety of flight exist, AFM limitations are not mandatory; however, additional items of information may be required by regional certificating personnel when such items are found to have a direct and important bearing on safe operation due to unusual design, operating or handling characteristics. (Ref. CAR, Section 4b.740(c).) The limitations imposed by the Southwest Region provide information having a direct bearing on safe operation. These limitations impose no operational penalty and reduce the probability of autopilot misuse resulting from the lack of separate glide slope and localizer coupling.

g. The Western Region in their memo (Ref. para 3.i.) concurred with the flight manual limitations imposed by the Southwest Region while continuing to disagree on the matter of compliance.

h. The response of any autopilot to these ambiguous signals, when operated in the manner described, is a normal function of the autopilot and to preclude such response other operational modes are usually provided. The response experienced by the SP-40 autopilot during these tests (Ref. para 2.C) would be experienced by any autopilot installation being operated in "Approach" mode while attempting to hold utilizing the localizer frequency. In the case of the SP-40 autopilot the "RADIO" mode when operating on the localizer frequency is the "Approach" selection.

i. The SP-40 autopilot includes in its design other modes of operation that would preclude airplane response such as that experienced during the subject installation approving tests. It is quite obvious that had operational instructions been furnished for this autopilot installation that the test conducted using "RADIO" selection would not have been accomplished.

4. CONCLUSIONS

a. In consideration of the foregoing, it is concluded that the Sperry SP-40 installation in the Grumman G-159 airplane is in compliance with CAR 4b.612(d)(5) and is also in conformance with TSO-C9b. (Ref. para 3.e.)

b. It is further concluded that the AFM revision required by the Southwest Region contributes to the safety of flight operations and should remain in effect as accepted by the applicant and concurred with by the Western Region, although lack of hazard or jeopardy to safety indicates this AFM revision should be considered information rather than limitation. (Ref. para 3.f and 3.g.)

c. It is finally concluded that no change to regulatory or TSO requirements are needed as a result of this case. Means to discriminate between localizer and glide slope signals, however desirable, have not been shown to be required for safe autopilot coupler operation.

REVIEW CASE NO. 46 PRUE SUPER STANDARD GLIDER VISIBILITY REQUIREMENTS (Issued 9 October 1964)

1. ORIGIN AND PROBLEM.

a. The Western Region has received an application for the approval of the Prue Super Standard Sailplane under the airworthiness requirements of Civil Air Regulations, Part 5.

b. The Federal Aviation Agency Basic Glider Criteria Handbook, revised 1962, Page 113, Item a, Ventilation and visibility, states that "In cabin gliders, the windows should be so arranged that they may be readily cleaned or easily opened in flight to provide forward vision to the pilot." The region considers this handbook, to appropriately modify Part 3 requirements to meet the airworthiness requirements for gliders.

c. The design of the Prue Super Standard Sailplane does not incorporate an openable window nor provisions to clean the windshield in flight. Mr. Prue, the applicant, has requested a waiver from the provisions referenced in this paragraph, item b, because, for all practical purposes, a single place sailplane is flown in good weather.

d. There exists a need for an official clarification on the relationship between the Basic Glider Criteria Handbook and the applicable airworthiness regulations.

2. REFERENCE REGULATIONS.

a. 5.10 Eligibility for type certificates.

A glider shall be eligible for type certification under the provisions of this part if it complies with the airworthiness provisions of Part 3 /or Part 6/ of this subchapter modified to the extent the Administrator finds /are applicable to the type design and are/ appropriate for gliders: Provided, That the Administrator finds no feature or characteristic of the glider which renders it unsafe.

b. CAR 3.382 Vision

The pilot compartment shall be arranged to afford the pilot a sufficiently extensive, clear, and undistorted view for the safe operation of the airplane. During the flight in a moderate rain condition, the pilot shall have an adequate view of the flight path in normal flight and landing, and have sufficient protection from the elements so that his vision is not unduly impaired. This may be accomplished by providing an openable window or by a means for maintaining a portion of the windshield in a clear condition without continuous attention by the pilot. The pilot compartment shall be free of glare and reflections which would interfere with the pilot's vision. For airplanes intended for night operation, the demonstration of these qualities shall include night flight tests.

c. FAA Basic Glider Criteria Handbook, Page 113, Item a., Ventilation and visibility.

The pilot's compartment should be so constructed as to afford suitable ventilation and adequate vision to the pilot under normal flying conditions. In cabin gliders the windows should be so arranged that they may be readily cleaned or easily opened in flight to provide forward vision for the pilot.

3. CHRONOLOGICAL HISTORY.

a. June 18, 1964 - WE-412 letter to Mr. Prue stating that the Prue Super Standard Sailplanes must have windows that are readily cleaned or easily opened in flight.

b. July 23, 1964 - Letter from Mr. Prue to WE-210 requesting a waiver from the provisions listed in this paragraph, item a..

c. August 19, 1964 - WE-400 memorandum to FS-100 requesting a review case to resolve the visibility requirements for the Prue Standard Glider.

d. August 31, 1964 - FS-160 memorandum to EA-216 requesting details associated with administration of pilot visibility requirements for gliders.

e. September 21, 1964 - Memorandum reply from EA-216 stating that the Eastern Region has not required a cabin window that could be opened in flight. Such windows are installed in gliders exported to Great Britain where glider flight is permitted under instrument flight rules including icing conditions.

4. RELATED BACKGROUND.

a. Civil Air Regulations, Part 5, Glider Airworthiness refers to Part 3 or Part 6 for the airworthiness provisions. In the case of the Prue Super Standard Glider, the applicable portions of Part 3 would be used to determine the eligibility for certification. In particular, CAR 3.382 Vision, prescribes the visibility requirement for this glider.

b. A review of glider designs type certificated does not reveal an instance of imposing a requirement for a cabin window or windshield that could be easily opened or cleared in flight.

5. DISCUSSION.

a. For safe operation, it is essential that the glider pilot be afforded adequate vision during all approved operations. This may be accomplished in any manner or by any means that is effective. For example, the airflow characteristics over the windshield and canopy may be such that vision is not impaired during flight in moderate rain thereby eliminating the need for a window that can be opened or cleaned in flight.

b. If a device such as a windshield wiper is required to maintain adequate vision during approved operations, it becomes required equipment. In this case, the installation of such a device must perform its intended function and be sufficiently reliable to assure that the probability of failure is remote.

c. Adequate pilot visibility must be maintained during all operations approved for the glider. If some operations, for example, flight in air with visibile moisture, unduly impair the pilot's visibility it is prudent to impose suitable operating limitations.

d. The FAA Basic Glider Criteria Handbook is provided to furnish glider design and operating information and to show acceptable means of compliance with some of the airworthiness requirements. The contents of this handbook should not be construed to be requirements but may be considered when determining compliance with the applicable requirements.

6. CONCLUSIONS.

a. The Prue Super Standard Glider must comply with the visibility requirements of CAR 3.382, Vision, for all approved operations.

b. Appropriate operating limitations should be imposed if the visibility is unduly restricted during flight in air with visible moisture or for any other reasons.

c. The purpose of the Basic Glider Criteria Handbook is to provide individual glider designers, the glider industry, and glider operating organizations with guidance material that augments the glider airworthiness certification standards specified in CAR 5. Acceptable methods of showing compliance with the standards are presented as compliance suggestions. Considerable material regarding common practices of construction and fabrication has been included primarily for the information of novice builders and designers, and should not be considered as the only satisfactory practices.

REVIEW CASE NO. 47 APPLICABILITY OF CAR SECTION 3.381(b) TO THE DE HAVILLAND DOVE DH-104 (Issued 19 October 1964)

1. INTRODUCTION

a. The Southern Region requested a review case to determine applicability of CAR 3.381(b) to a de Havilland DH-104 STC modification. The particular modification involved the installation of two (2) IO-720 reciprocating Lycoming engines and Hartzell propellers in place of the currently approved two (2) de Havilland Gypsy Queen 70-4 recoprocating engines and propellers.

b. The Southern Region, SO-210, concludes that the de Havilland DH-104 STC modification must meet the provisions of CAR 3.381(b).

2. CHRONOLOGICAL HISTORY. SO-210 memorandum to FS-100 dated October 2, 1964. This memorandum transmitted the region's review and recommendations concerning this STC application.

3. FACTS IN THE CASE

a. The de Havilland DH-104 was approved on April 4, 1951, on the basis of United Kingdom Certificate of Airworthiness for Export under the bilateral agreement between Great Britain and the United States.

b. The basis for U.S. certification of the de Havilland DH-104 was summarized in a memorandum from the Director, Office of Aviation Safety, W-270, to the Regional Administrator, Region 1, dated December 8, 1950. This memorandum stated that this aircraft was eligible for U.S. certification under CAR Part 3 provided it met the relevant British requirements and the additional conditions stipulated by the CAA, at that time. These additional conditions were set forth in a letter from CAA Administrator, T. P. Wright, to the Ministry of Civil Aviation, R. H. Walmsley, dated November 12, 1947, and in a subsequent letter from CAA Administrator, F. B. Lee, to the Secretary and Chief Executive Air Registration Board, R. E. Hardingham, dated May 26, 1949.

c. The British regulations used in the certification of the de Havilland DH-104 were those which had been in effect since 1944. These requirements did not contain a provision concerning the location of the propeller plane of rotation with respect to the pilot or primary flight controls as required by current CAR Section 3.381(b).

d. The Civil Air Regulations which were in effect when the de Havilland DH-104 was approved did contain a requirement concerning the location of the propeller plane of rotation, reference CAR 03.380. This particular provision, on the basis of the general comparison of the U.S. and British requirements which were performed at that time, was not considered a significant safety item and was not prescribed as an additional condition for U.S. certification (see 3b).

e. The Atlanta District Office, SO-EMDO-42, concluded that CAR 3.381(b) would not be applicable and that this particular STC modification need comply only with the original certification requirements. (See 2 a., page 4, item 2.)

f. The Southern Region, SO-210, concludes that the de Havilland DH-104 must comply with CAR 3.381(b), Pilot Compartment, General. This regulation states that "The primary flight control units listed on figure 3-14, excluding cables and control rods, shall be so located with respect to the propellers that no portion of the pilot or controls lies in the region between the plane of rotation of any inboard propeller and the surface generated by a line passing through the center of the propeller hub and making an angle of 5° forward or aft of the plane of rotation of the propeller." The region contends that compliance with this regulation is necessary because there are no compensating factors which will provide an equivalent level of safety for provisions not complied with as required by current CAR 3.10. The region points out that there are no means to establish an equivalent level of safety since the de Havilland DH-104 has never been approved with Lycoming engines and Hartzell propellers.

g. SO-210 provides no evidence that the requirements under which the de Havilland DH-104 was certificated, which regard to location of the propeller plane of rotation, have not been adequate. In this connection, the region points out that the applicant advised that the propeller plane on the modified version of the DH-104 would be the same as on the original aircraft. In regard to service experience Mr. J. M. Riley, in a letter dated September 5, 1961, to the Director, Flight Standards Service, points out that as of that date there were approximately 80 DH-104 aircraft certificated in the U.S. and that more than 600 have been in operation over a period of up to 15 years with no problems of any kind occuring due to the propeller location.

h. SO-210 also contends that under the present wording of CAR 3.11(d)(2) and its accompanying note, compliance with CAR 3.381(b) would be required. Their basis for this finding stems from the reference to component changes and the example in the Note concerning fire or operational hazards involving new powerplant installations. The region provides no supporting evidence that the installation of Lycoming engines will not provide a level of safety equal to that incorporated at the time of issuance of the type certificate during which the Gypsy Queen engines were installed. Both the Lycoming IO-720 and the Gypsy Queen 70-4 engines were reciprocating-type engines.

i. The primary intent of CAR 3.11(d)(2) is to provide a basis upon which the Administrator can require compliance with additional requirements if the original certificating requirements "do not provide complete standards with respect to such change." In this particular case, i.e. propeller plane, standards were available at the time of approval in the U.S. Civil Air Regulations but compliance was not required of the DH-104 imported from Great Britain.

j. Region 2 was advised by memorandum from the Chief, Aircraft Engineering Division, W-235, on November 4, 1953, (see 2 a., attachment G) that design changes need only meet the British airworthiness requirements and all of the U.S. special conditions applicable on the date of the original U.S. certification. This memorandum was prepared in response to a specific request for a policy ruling concerning the requirements governing STC approvals for <u>this same aircraft</u>.

k. CAM 18.30(b) and accompanying Note states that airworthiness requirements applicable to an alteration are normally those with which the manufacturer originally demonstrated compliance for the issuance of a type certificate. Since the de Havilland DH-104 was approved under the British requirements and the additional conditions stipulated by the CAA (see 3b) an alteration for this aircraft need only comply with these criteria.

l. This review case does not consider the question as to whether or not this particular modification complies with the STC requirements of CAR Sections 1.25 through 1.28.

4. <u>CONCLUSIONS</u>. In consideration of the foregoing, it is concluded that de Havilland DH-104 STC modifications, subject to the provisions of CAR 1.25 through 1.28, need not comply with CAR 3.381(b) where this regulation is a factor since compliance with this provision was not required for original certification.

REVIEW CASE NO. 48 LOCKHEED MODEL 382(C-130E); DESIGN LANDING DESCENT VELOCITY, CAR 4b.230(b)(1)(i) (Issued 19 November 1964)

1. INTRODUCTION

a. The Southern Region is currently administering the application for type certification of the Lockheed-Georgia Company Model 382 (Military C-130E) airplane under CAR 9a - "Aircraft Airworthiness; Surplus Military Aircraft," which specifies the requirements of CAR 4b, effective on October 1, 1959, including Amendments 4b-1 to 4b-11. The applicable CAR 4b requires structural design for ground loads in landing conditions based upon a minimum design descent velocity of 10 feet per second(fps).

b. The Model 382 airplane, as the Model C-130E, was designed to military specification MIL-A-8866, which specified a minimum design descent velocity of 9 fps and, hence, literal compliance with the higher value of 10 fps of CAR 4b does not exist.

c. The Lockheed-Georgia Company contends, for the Model 382, that an equivalent level of safety, and level of airworthiness equivalent is all essential factors has been more than adequately demonstrated by analysis, drop testing, static testing, and actual landing tests. Lockheed further contends that the redesign and associated testing, to show and prove literal compliance of the Model 382 with the CAR 4b design descent velocity of 10 fps, is prohibitively expensive and imposes a severe burden on Lockheed.

d. Lockheed further believes that a redesign, to comply with the 10 fps design descent velocity, would be speculative and inconclusive since only one of the six components, which would require redesign, has been involved in actual failures associated with extra-high design descent landings in service or during testing.

e. The Southern Region concurs with Lockheed's position that the C-130E value of 9 fps provides substantially the same level of airworthiness it would have, if designed to the CAR 4b requirements and that literal compliance with the CAR 4b design value of 10 fps, would result in no appreciable increase in safety.

2. CHRONOLOGICAL HISTORY

a. March 23, 1964. Date of FS-100 memorandum to Regional Branch offices regarding uniform application of CAR 9a and transmitting a copy of highlights on FS-100 Meeting on Uniform Application of CAR 9a, held in Washington, D. C., on March 11 and 12, 1964.

b. April 21 and 22, 1964. Dates of the Preliminary Type Certification Board Meeting, Model 382 at Marietta, Georgia (FAA Project No. CT 355SO-D).

c. August 17, 1964. Date of letter from Lockheed-Georgia Company to Southern Region requesting review of need for application of CAR 4b, design descent velocity, to the Model 382 type certification under CAR 9a.

d. September 25, 1964. Date of memorandum from SO-210 to FS-100 requesting concurrence with Regional favorable position on Lockheed's Review Case request.

3. FACTS IN THE CASE.

a. The basis of type certification for the Lockheed Model 382, CAR 9a, requires under CAR 9a.2(b)(2)(i) that compliance must be made with Part 4b as amended on December 31, 1953, including Amendments 4b-1 through 4b-11.

b. The Preamble to CAR 9a indicates that the Agency realizes that compliance with all of the specific regulations might not be practical, in the case of surplus military aircraft, since they have been constructed to a somewhat different design philosophy than civil aircraft. This Agencies position was reflected by the provisions of CAR 9a.2(h), quoted here:

> In cases where the applicant has shown to the satisfaction of the Administrator, with respect to a particular aircraft being submitted for type certification, the strict compliance with a specific provision of the applicable requirements prescribed in paragraphs (a) through (f) of this section would impose a severe burden on the applicant, the Administrator may accept such compliance as he finds will provide substantially the same level of airworthiness as is provided by the specific provision of the requirements. In such cases, evidence of satisfactory military service experience may be considered in determining whether the level of airworthiness is substantially the same as that which would be provided by strict compliance with the specific provisions of the applicable requirements.

c. The Lockheed Model 382, as the C-130E military transport under military specification MIL-A-8866, was designed for landing conditions with a design limit descent velocity of 9 fps. The applicable CAR 4b requirements specify a minimum value of 10 fps, at maximum landing weight.

d. Lockheed states that literal compliance with the higher 10 fps value would require major changes to six major gear components:

> Main landing gear piston, cylinder, forward and aft section of the torque strut, the torque strut attach bolt, and the track support beam. The Souther Region advises, additionally, that the entire outer wing lower surface is critical in compression.

e. Lockheed has stated that the redesign and associated testing, to show and prove literal compliance with the 10 fps value, is prohibitively expensive and imposes a severe burden. The Southern Region has noted that the wing skin panels are integrally stiffened and redesign would impose a substantial burden in cost and weight.

f. The magnitude of non-compliance of the landing gear limit load strength for the 10 fps value, as reflected in the CAR 4b landing conditions from Lockheed data, is summarized as follows:

> <u>Two Point Level Landing</u> - 18 and 10 percent negative margin based on analysis of the spin-up and springback loads. Drop tests with simulated spin-up and springback show adequate strength for the 10 fps value.
>
> <u>Tail Down Landing</u> - 20 per cent negative margin based upon analysis. About zero margin based on drop tests simulating wheel spin-up. About 8 percent negative margin for spin-up and springback based upon airplane flight tests.

g. Lockheed advises that over 300,000 landings have been made on C-130 series airplanes including operation into rought, unprepared, or semiprepared airstrips during the Lebanon Crises, the Congo Airlift, Peruvian Airlift, Marine Assault, and operation into Laos and Vietnam. Lockheed further notes that an unprecedented record of safety, reliability, and operational availability in military service has been established, despite the character of the reference mission which precluded strict adherence to landing conditions, descent rate limitations, and standard maintenance procedures.

h. The Southern Region notes that failures experienced, during the experience quoted in g. have note been of an extremely hazardous nature, have not been excessive in number, have brought redesign and improvement where practical, and have seldom involved any of the components requiring redesign for literal compliance (3.d.).

i. Lockheed notes that three major types of service failures have occurred and have been corrected:

(1) Main Landing Gear Piston/Axle cracks on early C-130A airplane, due to faulty grinding procedures. Corrected in production by change in manufacturing process.

(2) Main Landing Gear Track Failures - Early C-130E airplanes, corrected by track beef-up and increased length of main landing gear shoes.

(3) Port Chop Fitting Failures - C-130B and E corrected by changing material to steel and improved method of attachment to track back-up structure.

Lockheed advises that all of the above changes and improvements have been incorporated in the Model 382 airplane.

j. Lockheed reports that the Model C-130E airplane has been landed, without damage, during flight test operations at measured descent velocities in excess of 10 fps, at weights less than the design landing weight of 130.000 pounds, as follows:

11 fps @ 85,000
10.2 fps @ 118,000
17.2 fps @ 124,000

k. The Model 382 landing gear has successfully, without any structural failure, withstood a drop test at 135,000 pounds weight and descent velocity of 12 fps, which is slightly in excess of the CAR 4b energy absorption requirements.

l. The Model C-130E landing gear, also incorporated on the Model 382, was analyzed and tested for fatigue strength. Fatigue strength analysis and testing is in excess of the applicable CAR 4b requirements.

4. CONCLUSIONS

a. The degree of failure to comply in strict accordance with the applicable CAR 4b landing condition loads, is not severe.

b. Fatigue analysis and testing of the landing gear performed for the Model C-130E, and not required by the applicable CAR 4b requirements, tend to compensate for the lack of strict compliance with CAR 4b.

c. The Model 382, as the C-130, has had extensive service experience with no evidence of any serious deficiency in fatigue or structural strength.

d. The landing gear modifications and reinforcements incorporated, as a result of service experience under landing operations considered much more severe than envisioned by CAR 4b, also tends to compensate for the lack of strict compliance.

e. The conclusions a. b. c. and d. permit concluding that the design velocity applied for the Model 382 airplane provides a level of airworthiness substantially the same as that which would be provided by strict compliance with the provisions of CAR 4b.230 (b)(1)(i).

f. As provided by CAR 9a.2(h), type certification of the Lockheed Model 382 need not require strict compliance with the provisions of CAR 4b.230(b)(1)(i).

REVIEW CASE NO. 49 DOUGLAS AIRCRAFT COMPANY AMENDED PROPOSAL FOR FOLDING ARMRESTS TO CLEAR TYPE III EXIT AREA (Issued 21 December 1964)

1. INTRODUCTION

Review Case No. 44 concerned a Douglas Aircraft Company proposal to provide access to the DC-9 Type III exits by folding the adjacent seat armrests. Review Case No. 44 findings concurred with the Western Region that the Douglas proposal did not provide equivalent safety to the requirements of CAR 4b.362(g) and CAR 4b.362(e)(1) nor did they provide any compensating feature which would reduce evacuation time and so tend to offset the failure to meet the requirement. Subsequent to publication of Review Case No. 44, Douglas in a letter to the Western Region dated November 2, 1964, requested consideration of a revision to their original proposal.

2. CHRONOLOGICAL HISTORY

a. The Douglas Aircraft Company in a letter dated November 2, 1964, requested consideration of an amendment to their original proposal of March 3, 1964.

b. The Western Region transmitted the Douglas proposal to the Washington Office with their memorandum to the Engineering and Manufacturing Division dated November 25, 1964.

c. The Western Region further stated that they do not consider the Douglas revised proposal satisfactory.

3. FACTS IN THE CASE

a. Briefly the original Douglas proposal was to provide the DC-9 airplane with seats equipped with folding armrests. All armrests except those on the aisleway between seats could be foled upward out of the way. With the armrest down on the seats adjacent to the Type III emergency exits, the armrest projects approximately six inches into the exit area. Our conclusions in Review Case 44 were that present requirements and existing published policy ensure that seat armrests cannot obstruct the exit area because they are required to be removed simultaneously and automatically with the opening of the exit. Since the Douglas proposal did not do this it was considered not to offer equivalent safety.

b. In Douglas' revised proposal the only change over the original proposal is the addition of a placard stating that "Armrests must be raised on takeoff and landing." This placard would be installed on the inboard face of the exit door, thereby requiring the adjacent seat armrest to be raised during takeoff and landing.

c. Douglas considers that such a placard is in line with the precedent wherein the last sentence of CAR 4b.362(g) permits placarding a door through which passengers must pass to reach an emergency exit to be latched in the open position during takeoff and landing. They point out that this has been applied to doors in passenger compartment partitions as well as to doors separating passenger areas from galley areas that serve as access aisles to Type I emergency exits.

d. Douglas offers two compensating factors: the main cabin aisle width (below 25 inches from the floor) is 19 inches instead of the minimum of 15 inches permitted, and all armrests except those next to the aisle fold up out of the way. Both of these factors, they contend, facilitate evacuation of the airplane and reduce the time required.

e. We concur with the Western Region that Douglas' suggested addition of a placard does not make their proposal equivalent to the automatic removal required at present.

f. In reply to Douglas' contention that present practice permits the use of placards to require doors in passenger compartment partitions to be latched open for takeoffs and landings, we point out that the recent FAA/Industry Cabin Interiors Task Force recommended the removal of all doors from passenger compartment partitions. Requirements to accomplish this are now in process. We do not wish to add any more items than absolutely necessary to the number of things that crews must accomplish prior to takeoff and landing.

g. In all evacuation tests to date, present minimum aisle widths and seat armrests in place permit passengers to reach exits faster than they are able to go through the exits. Therefore, we are of the opinion that Douglas' slightly wider than minimum aisle and folded armrests cannot improve the time required for evacuation of the airplane.

4. CONCLUSION

It is concluded that the Douglas Aircraft Company proposal to add a placard requiring armrests adjacent to the Type III exits to be folded for takeoff and landing is still not equivalent to present requirements and existing published policy. Our findings in Review Case No. 44, therefore, are not altered.

REVIEW CASE NO. 50 ABBREVIATED FIRE DETECTOR SYSTEM FOR BOEING 707/720 AIRPLANES (Issued 28 December 1964)

1. INTRODUCTION

The Boeing Company, in an effort to find means for reducing the number of false fire warnings, has proposed removing all fire detectors from engine-mounted locations in Boeing 707/720 fan-engine airplanes, leaving only a minimum amount of detection equipment mounted on the fixed-engine supporting structure. The acceptance of this proposal necessitates a change in interpretation of the current fire detector requirement in the Civil Air Regulations. The Western Region position is that the regulations cannot be interpreted as Boeing suggests, and, therefore, the proposal to abbreviate the previously approved fire detector system is unacceptable. The case was referred to Washington for review.

2. CHRONOLOGICAL HISTORY

a. On April 23, 1964, the Boeing Company sent a letter to the Western Regional Office of the Federal Aviation Agency giving data and information about fire detector experience on Boeing turbine-powered aircraft and concluded that substantial gains (in solving the false fire warning problem) could result from system simplification. The letter went on to say that, for pod-mounted engines such as are used on Boeing aircraft, there is no opportunity for primary structure to be damaged by a small fire. Boeing sought adoption of the principle that prompt detection need only be required for fires near fixed structure, not for small fires which Boeing termed "harmless from a flight safety standpoint" or for moderate-sized fires which might be detected by other means. Concurrence with the principle was sought so that Boeing could proceed to reduce the fire detector system to a single loop above the engine on future 707/720 fan-engine aircraft.

b. The Western Region replied to the above letter on May 12, 1964, referring to a subsequent personal discussion with a Boeing representative on the subject, and stated that the proposal was unacceptable. An offer was made to discuss the matter further at Boeing convenience.

c. On June 8, 1964, the Boeing Company again wrote to the FAA Western Region for the purpose of substantiating the Boeing position that the overall safety level of the airplane would not be jeopardized but, in fact, increased by adoption of the new philosophy. Reference was made to the FAA fire tests conducted on the Boeing 707 pod, to service experience on podded military aircraft, on 707/720 commercial aircraft, and to other concepts which have been proposed to deal with the problem.

d. The Western Region replied to Boeing on June 23, 1964, stating that the proposal was unacceptable on the basis of the information presented. The principal objection was the absence of any showing that the shortened detector system would detect promptly enough to constitute adequate compliance with Civil Air Regulations, Section 4b.485. The existing detector system has demonstrated its ability to detect fire; a lesser system might conceivably create a hazard or flight emergency in some unpredictable manner. The possibilities associated with other available means for reducing false fire warnings had not been exhausted.

e. Boeing wrote a letter to the FAA Western Regional Office on July 17, 1964, requesting a case review.

f. The Western Region letter of July 28, 1964, advised Boeing that the request had been transmitted to the FAA Washington Office on July 23, 1964.

g. A meeting between Boeing and FAA was held in Washington, D.C., on September 3, 1964, to discuss the Boeing proposal.

h. On October 30, 1964, Boeing wrote to the FAA Washington Office and supplied additional data on B-52 fire experience.

3. <u>FACTS IN THE CASE</u>

a. In accordance with the established procedures set forth in Order FS P 8110.3, the Western Regional Office requested a review by Washington of a Boeing proposal to delete a substantial part of a previously approved fire detector system from certain future production 707/720 aircraft. The issue involved is whether or not the regulations can be interpreted in the manner suggested by Boeing so that the resulting system could be considered in compliance with the appropriate section of the Civil Air Regulations.

b. The Boeing KC-135 and its commercial counterpart, the B-707, were the forerunners of a new concept in turbojet transport design. The distinguishing feature of the new design was that the engines were mounted in pods under the wings. Fire tests of the pod-mounted engines were conducted by the Civil Aeronautics Administration (now FAA) in early 1957 to determine the likelihood of flammable fluid ignition on the engine, resistance of the structure and engine mounts to fire exposure, and external flame paths after nacelle burnthrough. The results of the fire test program are contained in Technical Development Report No. 357.

c. In the Boeing Company letter dated June 8, 1964, several excerpts from the above technical development report were quoted in an attempt to substantiate the safety of the Boeing design as follows:

(1) "Under the most severe fire conditions, flames did not reach or come close to the under surface of the wing."

(2) "MIL-L-7808 engine oil did not ignite from any of the engine surfaces and was difficult to ignite from a primer fire."

(3) "Fire reached the strut and burned through the strut skin. Flame penetration into the strut, however, did not occur."

(4) "The strut supporting the engine pod is of adequate length to prevent external flames from contacting the wing surfaces."

The Boeing Company also cited the report to show that engine fires could be completely controlled by shutting off fuel with the fuel shutoff valve and allowing drainage from the nacelle and combustion to deplete the source of the fire.

d. The Boeing Company points out that 21 million engine hours have been accumulated by nearly four thousand B-47, B-52, and KC-135 aircraft without a serious pod fire, without damage to primary airframe structure, and without any struts being burned away or any engines being burned off. The majority of this experience was obtained using token fire detection systems or none at all. The fire detector system was completely deleted from the B-47 in February 1958 to eliminate hazards associated with false fire warnings. All late B-52s are equipped with only two spot detectors per engine, KC-135s have less coverage than the equivalent B-707 systems, and current B-707 systems are abbreviated to less than one-half of the original systems.

e. Because of the superior design features of the engine pod configuration as demonstrated in the CAA fire tests of 1957 and the experience gained in the operation of B-47, B-52, KC-135, and B-707 aircraft, the Boeing Company takes the position that it is unnecessary to detect fire promptly. In fact, Boeing suggests that it is not necessary to detect some fires at all if they are small enough or if they could be detected by other means such as powerplant or powerplant system irregularity, instrument indicators, or visual means. Boeing believes that sufficient protection would be afforded by mounting fire detectors above the engine (between the engine and the horizontal firewall).

f. While the fire test results and the extensive experience accumulated over the years may serve to demonstrate quite convincingly that the

Boeing 707 pod design offers some safety advantages with respect to fires in flight, the potential hazard of any powerplant fire must not be overlooked. The following additional quotes from Technical Development Report No. 357 show that fires, initially small, can become extremely large within a few minutes from their start and must be detected promptly to be effectively controlled:

(1) "The fires which were ignited inside the engine nacelle were low intensity fires as long as the nacelle remained intact."

(2) "Due to the low ventilation rate in the nacelle . . . . only a small quantity of fuel could be burned."

(3) "The louvers in the nacelle doors created a chimney effect during a fire and failed when exposed to flames for two minutes."

(4) "The aluminum skin on the nacelle doors resisted burnthrough for two minutes under severe fire conditions."

(5) "Control of the engine was maintained during the entire test even though the P_{t7} decreased during the height of the fire intensity."

It should also be pointed out that the most effective location for fire detectors was found to be under the engine. Even in this location, the response time was comparatively long because of the low ventilation rate. The significance of these items is that (1) when a fire starts in the Boeing pod, it remains small because of the low ventilation rate for approximately two minutes, after which it may be expected to burn through and grow much larger, (2) other powerplant instruments do not provide sufficient warning to be used as an alternate means for detecting fire, and (3) it is questionable whether or not detectors mounted above the engine only would provide prompt detection of all fires in the pod.

g. To provide a basis for updating the fire protection requirements for turbine engine pod configurations, the FAA recently launched a fire test research and development project using a full-scale Boeing 720B pod with a fan engine. Boeing assistance in carrying on this project has been solicited. The program includes studies of structural integrity and fire detector systems.

h. The FAA is aware of certain deficiencies of the current fire protection standards in the Civil Air Regulations, Part 4b, but needs data to substantiate changes to the requirements. In addition, the Regional Offices need guideline material to assist them in evaluating type certification projects for conformance with fire protection requirements. These are the reasons the FAA has established the fire test project.

i. Without data which will be forthcoming from the above tests, it is extremely difficult to judge whether or not equivalent safety is provided, the trade-off being between a detection system which detects fires but which occasionally gives false warnings, and a system which is comparatively trouble-free (supposedly) but which may permit the existence of fires for extended periods.

j. It was established at the September 3, 1964, meeting that Boeing was attempting to show compliance with CAR, Section 4b.485. To show compliance, it is necessary to provide quick-acting fire or overheat detectors. (Any detector approved, either as part of the airplane or under the technical standard order system, would meet this requirement.) The detectors must be provided in all designated fire zones and in the combustion, turbine, and tailpipe sections. (If mounted on the horizontal firewall, the detectors would have to extend into each of these zones in order to comply.) The question is whether detectors mounted only on the horizontal firewall can be considered sufficient in number and location to assure prompt detection.

k. The Western Region (and the FAA in general) has always interpreted this regulation as a requirement for a sufficient number of detectors properly distributed throughout a designated fire zone or protected section to detect any fire likely to occur therein within a few seconds from its start.

l. The Western Region interpretation is in accordance with the preamble to CAR Amendment 04-1 (of Part 04 as promulgated on November 9, 1945), effective November 1, 1946. The opening paragraph states "The requirements hereinafter set forth are intended to aid in preventing any fire from starting, to detect at the outset any fire which has started, to prevent the spread of any fire, and to extinguish any fire." It will be noted that the words "any fire" are repeated several times.

m. The exact words used in CAR, Section 4b.485, are "to assure prompt detection of fire." In the absence of any qualifying words with respect to size or location of fire, it must be presumed that the intent is to assure that any fire occurring outside the engine within any of the specified zones will be detected promptly. A corollary of this is that, if any fire were permitted to exist in the zones undetected, this would be contrary to the regulation.

n. Notwithstanding the arguments presented by Boeing to show that the safety of the pod installation in B-707 aircraft makes prompt detection less urgent than on some previous types of aircraft, the fact remains that the regulations as presently written require detection of any and all powerplant fires, not only those which might damage primary structure.

o. Boeing has provided some data on B-52 experience dating from April 22, 1960, when the simplified detector system for this airplane was adopted in accordance with T.O. 1B-52-1176. These data indicate that the particular system used on the B-52 is 70 to 80 percent effective in detecting fire (not necessarily promptly). The system used on the B-52, however, is probably not the one intended for the 707/720 models. Boeing has not indicated it is like the one on the B-52 and has not presented a detailed description of the proposed system. The only similarity in the two systems is probably their brevity. The B-52 pod is different from the 707/720 pod in that it is a dual pod, whereas the 707/720 pod is a single pod. No data exists to show that a fire in a single pod would react in the same way as a fire in a dual pod. There may be other differences also which would have to be considered in evaluating the relative safety of the designs. For these reasons, the fire data on the B-52 are not fully applicable.

p. In this review case, we are dealing with a concept rather than a specific system and our concern is not, in reality, limited to making a determination of compliance or noncompliance with Civil Air Regulations, Section 4b.485, Boeing request notwithstanding. There are three considerations in this instance; namely, (1) is the proposed (undefined) abbreviated system in compliance with the regulations, (2) is it in compliance with the intent of the regulations, and (3) will it provide equivalent safety.

q. Elsewhere in this document it has been shown that an abbreviated system of the type visualized would not comply strictly with the requirements for a fire detector system. It has also been shown that such a system would not comply with the intent of the requirements. In combination with the pod concept, however, it might be possible to show that a level of safety can be achieved that is equivalent to that intended by Civil Air Regulations, Part 4b, provided the applicant demonstrates by fire tests that:

(1) The consequences of small fires, which might go undetected for relatively long periods, would not compromise any of the fire protective features such as compartmentation, sealing, fuel shutoff reliability, or extinguishment.

(2) Large fires, detected late, would not endanger the aircraft. (This would provide an assessment of the inherent isolation of pod-mounted engines.)

4. CONCLUSIONS

It is concluded that:

a. The Boeing proposed abbreviated system would not comply with the current fire detector system requirement for "prompt detection."

b. The Boeing proposed abbreviated system would not comply with the intent of the requirement because it would not be designed to detect all fires in the protected zones.

REVIEW CASE NO. 51 COOLING TEST FOR APPROVAL OF TURBOSUPERCHARGER INSTALLATION (Issued 7 January 1965)

1. INTRODUCTION

a. In connection with an application for a supplemental type certificate by Alcor Aviation Inc. for the installation of a turbosupercharger in Cessna 180 and 182 airplanes, the Southwest Regional Office has advised Alcor that it will be necessary to comply with Civil Air Regulations, Section 3.586 "Cooling test procedure for single engine airplanes."

b. Alcor Aviation Inc. contends that the above-mentioned regulation does not apply to their installation in that they are not changing the demands on the engine and that the addition of the turbosupercharger should be treated as would an engine accessory.

2. CHRONOLOGICAL HISTORY

a. On February 28, 1964, Alcor Aviation Inc. advised the Southwest Regional Office of their intent to apply for an STC to have a turbosupercharger installed on a Cessna 180 and 182 airplane equipped with a Continental 0-470 engine. They also stated that other aircraft would be added to the STC. A proposed plan for obtaining certification was enclosed.

b. The Southwest Regional Office, in a letter dated March 6, 1964, advised Alcor that their proposal to substantiate a turbosupercharged engine had been evaluated, provided detailed comments, and requested that the program be revised in accordance with these comments.

c. Alcor "Proposal on Turbosupercharging Horizontally Opposed Engines" dated March 23, 1964, stated that detonation and/or cooling limited altitude would be established for each make and model of engine for both climb and cruise configuration. It was further indicated that, after cooling limits had been established, Alcor would recommend the minimum fuel grade to be used. It was also mentioned that cooling tests would be conducted on each aircraft configuration different enough to warrant additional data.

d. Alcor letter to Southwest Region, dated September 23, 1964, enclosed the STC applications and, further, pointed out that, from the standpoint of aircraft safety, since they were only seeking Federal Aviation Agency approval for turbosupercharging at powers up to 75 percent, there will be no question about exceeding the manufacturers power limits.

e. The Southwest Region made the following comment, among others, regarding the STC application and its applicability to all models of Cessna 180 and 182 airplanes: "This will require cooling tests of the various cowling configurations which have been produced for these aircraft. To our knowledge, there have been at least three different cowling configurations on the aircraft. You should investigate to determine if there are others which will require testing."

f. In commenting on the quote referenced in subparagraph e above, Alcor questioned the need to conduct cooling tests for each cowling configuration for which a given model engine is utilized. They concede that with the turbosupercharger the aircraft can attain an altitude at which engine temperatures will be exceeded when full available power is used at the angle established for best rate of climb. They state, however, that certification is being sought only for operation at altitude and that cowling configuration changes will only affect the altitude at which temperature limits will be reached at maximum climb power and at the attitude for maximum rate of climb.

g. Note 2 of the Type Inspection Authorization dated October 8, 1964, states "The cylinder head temperatures will vary for the different engines on which this turbosupercharger is to be installed. Cooling tests will be conducted when each type of engine is approved for a particular airplane."

h. SW-210 letter dated October 12, 1964, advised Alcor the engine cooling tests would be required for each different configuration for which approval was requested and that it would probably be necessary to conduct cooling tests from a near sea level condition to the maximum altitude to which certification was desired.

i. Alcor letter of October 13, 1964, to Southwest Region speaks to the cooling tests from the aspects of the turbosupercharger installation, per se, in the airplane. It is noted, however, that no mention is made of the turbosupercharger's affect during climb cooling tests.

j. The minutes of a meeting between Southwest Region and Alcor held October 16, 1964, reflect that Alcor questioned the necessity of the engine cooling tests and that it was pointed out by the Southwest Region that it was necessary to determine that the engine would cool or establish the climb speed required for cooling. The region indicated that a fairly comprehensive cooling test program on the first installation might make it possible to establish a minimum test procedure for subsequent installations.

k. Mr. Hundere, Alcor Aviation Inc., visited the Washington Office on October 19, 1964, to discuss the requirements for approval of the turbosupercharger. A record of conference, copy to Alcor, indicated the he objected to a full series of cooling tests. He mentioned that they would solve the cooling problem by establishing appropriate operating limits and procedures. It was pointed out to him by FS-140 personnel that any operational limitations that might be altered would have to be covered by adequate instructions and information to enable the new limits to be consistently observed by the calibre of pilots expected to operate the aircraft involved.

l. Alcor Aviation Inc. letter to FS-140 dated October 21, 1964, stated that the turbosupercharger would not be used during takeoff and initial climb. They do imply, however, that the turbosupercharger will be used for altitude climb as is evidenced by their suggested placard "UNDER CRUISE AND ALTITUDE CLIMB, CLOSE WASTE GATE TO MAINTAIN MANIFOLD PRESSURE UP TO 24" HG MAX." They further state that they are aware of the overtemperature and detonation dangers that could result from turbosupercharging.

m. Alcor Aviation Inc. letter to Southwest Region dated October 27, 1964, again reiterated that they did not feel cooling tests should be required other than to substantiate that the installation of the turbosupercharger does not affect engine cooling when the waste gate is open. They state "If the FAA is going to require such data, we need to know what use will be made of said data."

n. Minutes of a meeting between Southwest Region and Alcor again reflected Alcor objection to cooling tests in accordance with CAR 3.586. Alcor was advised that, if they disagreed with the regional position, a review case procedure could be considered.

o. Alcor Aviation Inc. letter to Southwest Region dated November 25, 1964, again mentions that they consider cooling tests in accordance with CAR 3.586 unnecessary for an installation where the turbosupercharger will be utilized at altitude only. In this letter the various explanations as to why the information on cooling is required were disputed. In expressing their objection, Alcor noted that "the pilot would, no doubt, not use the information," "For us to conduct tests for the FAA to show that the cooling limited altitude for particular aircraft at 65% power is 15,20,25,30 or 35,000 feet would only fill the FAA files," "we are going to conduct sufficient flight tests . . . . . . but we see no reason why such tests should be an FAA requirement," "we should not be handicapped by the unnecessary requests made of Rajay by another region." In this letter, Alcor asked whether the matter could be referred to Washington for a ruling.

p. Southwest Regional memorandum dated December 7, 1964, forwarded a summation of the problem and requested a Washington evaluation of the matter.

q. Alcor Aviation Inc., by their letter of December 9, 1964, advised FS-140 that the Southwest Region informed them that the difference of opinion regarding CAR 3.586 was referred to the Washington Office for review. Alcor again voiced an objection to conducting cooling tests, stating in response to a regional explanation for the tests "As to how the pilot keeps his engine temperatures in line under turbocharging (sic) conditions at altitude will be a matter of pilot descretion and not for FAA to dictate. He may choose to use richer mixtures or reduced power instead of increased air speed."

3. FACTS IN THE CASE

a. Originally, Alcor Aviation Inc. had a cooling test included in their test proposal.

b. It was only after the Southwest Regional Office pointed out to Alcor that a cooling test would be required for each different cowling configuration, that any objection to cooling tests was raised.

c. The applicant has maintained that the turbosupercharger will be used only at altitude and will not be used for takeoff and initial climbs. He has, however, made a distinction between initial climb and altitude climb and has indicated that the turbosupercharger will be used for climb to altitude (see above: 2.1.).

d. The applicant has recognized that, as a result of the turbosupercharger installation, overtemperature and detonation during climb and at altitude could be a problem (see above: 2.c, f, k, 1, q). They have mentioned that they fully intend to investigate this in order to determine what corrective measures must be taken (see above: 2.c, g, o).

e. The applicant questions the use that the FAA will make of the information obtained from cooling tests and objects to furnishing such information. This appears to be a prime basis for his objections. His arguments have not raised any points of technical substance that relate to the basic requirement of CAR 3.586 for cooling tests.

4. ANALYSIS

a. Subpart E of Part 21, Federal Aviation Regulations, prescribes procedural requirements for the issue of a supplemental type certificate (STC). Section 21.115 of this subpart requires that each applicant for an STC must show that the altered product meets applicable airworthiness requirements as specified in the regulations incorporated by reference in the type certificate. In the case in question, this would be Part 3 of the Civil Air Regulations.

b. Cooling tests prescribed in CAR 3.586 are propulsion installation requirements and are applicable in the certification process regardless of whether or not there is a turbosupercharger and without reference to whether it is furnished separately or as part of an engine package.

c. CAR 3.586 requires that, after engine temperatures have been stabilized, the climb is to be started at the lowest practicable altitude and continued for one minute with the engines operating at the takeoff rating. The climb then is required to be continued at maximum continuous power until at least five minutes after the occurrence of the highest temperature recorded.

d. Alcor has chosen to divide climb into two regimes, initial and altitude. It appears that initial climb is intended to culminate at that altitude referred to as the "lowest practicable" or, in other words, that altitude necessary to avoid obstructions peculiar to a particular area. It is not clear, however, that any specific altitude is representative of that which would terminate initial climb. For instance, in Death Valley, initial climb could terminate at a pressure altitude near or below sea level; whereas, in Denver, Colorado, the termination point would be several thousand feet above sea level. It does not appear, therefore, that there would be any persuasive argument for omitting or modifying cooling test requirements based upon the proposed operating procedure which is intended to limit use of supercharging to a particular portion of climb or cruise.

e. While the turbosupercharger would not be used during takeoff, it could be used soon thereafter for climb to altitude. It must be recognized that, in operation out of certain high altitude airports, the turbosupercharger would be used in climb to produce a power considerably above that which the engine would have experienced during initial certification of the aircraft without a turbosupercharger at the same altitude. The cooling characteristics of the powerplant installation, under these circumstances, would be unproven until verified by cooling tests.

f. Past operating experience with turbosupercharger installations has demonstrated that the engine is affected in many ways by the addition of a turbosupercharger. The air intake temperature is raised due to the compression action of the turbosupercharger. There is a reduced mass airflow for cooling because of the lower density of ambient air at altitude. Mixture distribution patterns are altered. The pressure on the exhaust valves is increased and the pressure and temperature pattern may be altered because of the increased exhaust manifold pressure necessary to operate the turbosupercharger. The production of heat by the turbosupercharged engine does not fall off as quickly with altitude because the turbosupercharger retains a much more constant power output with increasing altitude than is the case with a naturally aspirated engine. These factors are all present regardless of whether or not the maximum takeoff power of the engine is altered and without reference to whether the airframe or engine manufacturer supplied the turbosupercharger installation.

g. The effect of the factors discussed in paragraph f is recognized by the applicant in his proposal to establish detonation limits, cooling limited altitude, and to recommend the minimum fuel grade to be used.

h. One specific technical point not covered in information submitted by the applicant concerns the manner by which he intends to establish how his procedures are effective to ensure the operation of the engine within established cooling limits without conducting the required cooling tests. The effects of the addition of the turbosupercharger raise substantial doubt that the temperature pattern of the engine is unaltered from that of the unsupercharged engine. Without conclusive evidence on this aspect, there is no basis upon which to conclude that any particular procedure based upon observance of cylinder head temperature is adequate.

i. Cylinder head temperature limits for any basic engine are available to all airframe manufacturers. If the approach of approving an installation without cooling tests were accepted in this or any other case on the basis of operating procedures to keep cylinder head temperatures within limits, any airplane might be certificated without cooling tests. The many effects of the installation upon the overall cooling situation have shown clearly that cooling tests are needed and must be repeated whenever there are changes to the installation that might affect cooling and/or the procedures set up to keep the temperatures within limits.

j. The applicant has questioned the use that is made of data by the FAA. In the certification of aircraft, the FAA must perform a duty to the public in determining that an aircraft design is airworthy before a type certificate is issued. It cannot adequately discharge this duty without examining all data needed to establish that the airworthiness requirements are satisfied. It is recognized that, where data show satisfactory compliance with a requirement, there is no further active need for such data unless service difficulties necessitate that they be reexamined. In this sense, after certification, the various reports and analyses for the most part do little more than occupy file space, but the Agency could not grant approval to an aircraft without having the opportunity to evaluate the manufacturer's substantiation of airworthiness represented by this material.

5. CONCLUSIONS

In consideration of the foregoing, it is concluded that:

a. In accordance with FAR 21.115, the applicant must comply with airworthiness requirements specified in the regulations incorporated by reference in the original type certificate: CAR, Part 3.

b. CAR 3.586 sets forth the requirements for demonstrating cooling capabilities during takeoff and climb.

c. The applicant states that he intends the turbosupercharger to be used after initial climb for further climb to altitude. The altitude representing culmination of initial climb is undefined and, depending on the airport in question, could actually be a pressure altitude representative of sea level or below; therefore, it is necessary to demonstrate the cooling capabilities of the engine during climb irrespective of the phase of climb during which the turbosupercharger will be used.

d. Based on the results of the cooling test, appropriate limitations and procedures could be established if they are shown to be practical and effective.

REVIEW CASE NO. 52 ROTOR CONTAINMENT FOR AIR TURBINE MOTOR AND REFRIGERATION COOLING TURBINE IN LOCKHEED MODEL 382(C-130E) AIRCRAFT (Issued 27 January 1965)

1. ORIGIN AND PROBLEM

The Southern Region, in a memorandum dated November 9, 1964, advised that Lockheed-Georgia Company has requested approval of the Air Turbine Motors and refrigeration cooling turbines based upon their showing of equivalent safety, recognizing that these installations do not literally comply with CAR 4b.659. The Southern Region concludes that the refrigeration cooling turbines are in compliance with CAR 9a as it pertains to and references CAR 4b.659, but the Air Turbine Motors are not in compliance with these requirements and should be limited to ground use only.

2. REGULATIONS AFFECTED

a. CAR 9a effective January 10, 1964, in which paragraph 9a.2(b)(2)(i) establishes the type certification requirements which are CAR 4b as amended to December 31, 1953, including Amendments 4b-1 through 4b-11, and in which paragraph 9a.2(h) states:

> In cases where the applicant has shown to the satisfaction of the Administrator, with respect to a particular aircraft being submitted for type certification, that strict compliance with a specific provision of this section would impose a severe burden on the applicant, the Administrator may accept such compliance as he finds will provide substantially the same level of airworthiness as is provided by the specific provisions of the requirements. In such cases, evidence of satisfactory military service experience may be considered in determining whether the level of airworthiness is substantially the same as that which would be provided by strict compliance with the specific provisions of the applicable requirements.

b. CAR 4b.659 states:

> Equipment incorporating high energy rotors shall be demonstrated as capable of containing a failed rotor or shall be so located that failure will not affect the ability of the airplane to continue safe flight.

3. FACTS IN THE CASE

a. Lockheed-Georgia Company's letter dated October 16, 1964, reference E-05-690-64, presented service history information and quality assurance practices on the Air Research Gas Turbine Compressor, and refrigeration cooling turbine and the Stratos Air Turbine Motors.

b. A review of this letter (reference paragraph 3.a) produces the following facts considered significant:

(1) The ATM turbine design speed is 43,000 RPM. An overspeed control device shuts off system at 54,000 plus or minus 2000 RPM. Every turbine wheel is spin tested to 56,000 RPM for three minutes. Qualification testing included a turbine wheel spinning at 57,200 RPM for 20 hours, one inadvertently going to 95,000 RPM and a three minute period elapsed prior to shutdown. Neither of these two wheels failed. A destructive test was conducted and the turbine wheel failure occurred at 103,000 RPM. The design margin of safety is 140 percent over its normal operating speed at 860°F as compared to its normal operating temperature of 500°F.

(2) Lockheed states that over 1000 similar units have accumulated "extensive running time and, to date, there has not been a single reported case of turbine wheel failure. Total ATM operations in C-130's is approximately 27,400 hours." Although this is a small number of hours as compared to the millions of hours accumulated on the refrigeration cooling air turbine, we consider that this number of hours without a failure is a significant factor.

(3) Lockheed proposed to recommend the following during overhaul of repair processing of commercial units: Magnetic particle inspection, dimensional inspection before and after balance and spin test, and X-ray inspection.

(4) No failures occurred on which a positive assessment could be made of the resultant damage; however, the location of the ATM is such that failure of the rotor or the resulting damage would not cause catastrophic damage to structure or occupants.

c. The regulation in question, CAR 4b.659, was added by amendment 4b.-8, effective May 17, 1958, for the purpose of protecting personnel on the ground and in flight, as well as vital aircraft structure and system components, from the potentially lethal fragments of a failed high energy rotor. With improved quality (reliability) control available in the advancing state-of-the-art, it was considered appropriate to delete 4b.659 by amendment 4b.12, effective May 3, 1962. This amendment came immediately after those applicable to the type certification basis for this aircraft. The deletion was based on the provisions of 4b.606 as adequately covering the reliability of all equipment, systems, and installations.

d. A finding of equivalent safety per CAR 4b.10 should give full consideration to reliability control and service experience demonstrating the effectiveness of such reliability control. If no rotors have failed to date, and the probability of such failure is shown to be extremely remote, we cannot support the requirement for containment.

4. CONCLUSIONS

a. We conclude that the Air Research refrigeration cooling turbines comply with applicable requirements and concur with the findings of the Southern Region in this regard.

b. We conclude that the Stratos Air Turbine motors comply with applicable requirements based on equivalent safety with compensating factors of reliability control, including special inspections indicated in paragraph 3.b.3, and absence of any unsatisfactory Service experience, notwithstanding the findings of the Southern Region.

REVIEW CASE NO. 53 BOEING PROPOSAL FOR ACCESS AISLEWAY AT TYPE II EMERGENCY EXITS IN 707-300C AIRCRAFT (Issued 29 April 1965)

1. INTRODUCTION

The Boeing 707-300C airplane is a cargo/passenger version of the 707-300, and is so designed that cargo is carried in the fuselage forward of the passengers. The bulkhead separating cargo from passengers is movable such that the ratio between cargo and passengers can be varied from all cargo to all passenger. In certain mixed configurations forward exits are blocked by cargo; because of this Boeing has added a Type II exit on each side approximately halfway between the wing trailing edge and the aft Type I exits. Boeing has proposed an arrangement which the Western Region considers does not provide the access to these Type II exits required by CAR 4b.362(g). Boeing, therefore, has requested that the Western Region request a review case. This was done by WE-400's memorandum to FS-100 dated March 5, 1965.

2. CHRONOLOGICAL HISTORY

a. The Western Region in their memorandum dated March 5, 1963, to the Engineering and Manufacturing Division, presented the Boeing request for a review case.

b. The Western Region further stated that they do not consider that Boeing's proposal meets the access and assist space required by CAR 4b.362(g).

3. FACTS IN THE CASE

a. CAR 4b.362(g) requires that passageways leading to Type I and Type II emergency exits must be unobstructed and at least 20 inches wide.

b. CAR 4b.362(g) requires enough additional space at exits, which must have emergency evacuation means, for a crewmember to assist in the evacuation of passengers without reducing the unobstructed width of the passageway below that required for that exit.

c. A minimum acceptable size of the assist space has never been contained in the regulation. Since May 5, 1958, however, Federal Aviation Agency published policy contained in Civil Aeronautics Manual 4b.362-6(b) has stated that this should be a 12" x 20" area with the long dimension parallel to and clear of the required 20 inch exit approach passageway.

d. The Western Region considers that the Boeing proposal does not provide a 20 inch access aisle to the Type II exit because of interference on the aft side with the attendant assist space or on the forward side with the seat backs. Boeing considers that the required access is provided by a dog-leg in the aisle to fit an angled attendant assist space, and by providing seats with breakover backs which can be rotated out of the 20-inch passageway.

e. The Western Region interprets CAR 4b.362(g) to require the 20-inch access passageway unobstructed at all times except for seated passenger legs and feet. They also consider that the angled assist space is not practical, because it is natural that the assisting attendant would station herself such that the assist space would be normal to the plane of the door. Since Boeing has not provided any compensation for these encroachments, the Western Region recommends against the Boeing interpretation of CAR 4b.362(g).

f. Up to the present time, seat backs have been allowed to encroach on required exit area only at overwing exits. This has been permitted in cases where the seat backs could be pushed sufficiently forward to clear the exit area, even though the seat was occupied. The only encroachment which has been permitted in the assist space leading to Type I and Type II exits is that flight attendant seats have been allowed in the passage provided that they are spring loaded and automatically clear the passage when the flight attendant stands up. Thus, with occupants on their feet the full passageway is assured. This has the advantage of stationing the flight attendant right at the exit. Encroachments such as Boeing proposes lower the overall safety level.

g. We, therefore, concur with the Western Region, that the intent of CAR 4b.362(g) is to provide a minimum unobstructed 20-inch aisle leading to Type I and Type II exits, plus an additional 12" x20" assist space clear of the 20-inch passage with the long dimension normal to the plane of the exit where emergency evacuation slides are required.

4. CONCLUSIONS

It is concluded that the Boeing proposal to angle the assist space and rely on breakover seat backs to clear the passageway to the Type II exits in their Model 707-300C airplane does not meet the intent of CAR 4b.362(g) and is, therefore, unacceptable.

REVIEW CASE NO. 54 COMPLIANCE WITH FAR 23, SECTION 23.735 BRAKES: AMERICAN TURBINE ENGINE COMPANY (Issued 16 May 1965)

1. ORIGIN

The Western Region on March 9, 1965, requested a review case decision on a single tailed twin Beech airplane (PAC Aero) with Pratt & Whitney PT-6 turbo propeller engines owned by the American Turbine Engine Company. Working toward a supplemental type certification, American Turbine Engine Company has found that the subject aircraft will not meet the static brake test required under Federal Aviation Regulations Part 23, Section 23.735.

2. REGULATIONS AFFECTED

FAR, Part 23, Section 23.735 states:

> There must be brakes that are adequate to (a) prevent the airplane from rolling on a paved runway with takeoff power on the critical engine; and (b) provide adequate speed control during taxiing without excessive pilot loads.

3. FACTS IN THE CASE

a. FAR Section 23.735 brake systems are evaluated to ascertain that the brakes can prevent the wheels from turning with full power on the critical engine. This is necessary to achieve an evaluation of the holding ability of the brakes during type certification.

b. With brakes and wheels locked on the subject airplane when demonstrating compliance with FAR, Part 23, and under full runup power, the brake and wheels do not rotate but the tires will skid along the surface of the runway.

c. The Western Region's proposal is to "accept the brakes based on a satisfactory qualitative demonstration of being able to skid the tires or prevent the airplane from moving."

d. A condition similar to the above occurred in 1961 when Hamilton Aircraft Company applied for an exemption from Civil Air Regulations, Section 3.363, because their T28R-2 aircraft would slide with the brakes locked before full power was reached.

e. The Safety Regulations Division at that time stated that if the wheels of the Hamilton Aircraft Company T28R-2 airplane were locked and did not rotate during application of full takeoff power, the intent of CAR 3.363 would be satisfied in this respect. Based on this, no exemption was necessary.

f. In view of the above, a certification precedence has been established for instances wherein a Part 23 airplane, with the brakes and wheels locked, developed a tire skid condition due to a high thrust-to-weight ratio.

4. CONCLUSIONS

The Western Region should advise American Turbine Engine Company that their single tailed twin Beech (PAC Aero) with Pratt & Whitney PT-6 turbo propeller engines meets the intent of FAR, Section 23.735, if the wheels do not rotate with the brakes locked when takeoff power is applied to the critical engine.

REVIEW CASE NO. 55 HILLER REQUEST FOR APPROVAL OF OPERATIONS UNDER FAR 133 CLASS B ROTORCRAFT-LOAD COMBINATION AT A MAXIMUM WEIGHT OF 3500 POUNDS (Issued 12 May 1965)

1. INTRODUCTION

a. Hiller Aircraft Company, Inc. desires approval of an external jettisonable cargo operation for their model UH-12L4, UH-12L, and UH-12E-L helicopters at a maximum total weight of 3500 pounds which is 400 pounds in excess of the currently certificated weight under CAR 6. Hiller originated a project with the Western Region to approve this configuration for operation under FAR 133 having shown compliance at the higher weight with the structural flight loads and main component service life determinations of CAR 6. Hiller notes that approval was granted to the Bell Helicopter Company to operate their Model 204B helicopter as a Class B rotorcraft-load combination wherein the maximum total weight exceeded the CAR 7, Category B certificated weight by 1000 pounds. (Review Case No. 37)

b. Hiller operators propose to show compliance with the Flight Characteristics Requirements of Section 133.41 at the maximum weight for which authorization is requested up to a maximum of 3500 pounds.

c. The Western Region has denied approval under FAR 133 on the basis that Part 133 requires the applicant to meet all the provisions of CAR 6 or 7, rather than only structural substantiation of the main rotor and drive, for certification at the increased weight.

2. CHRONOLOGICAL HISTORY

a. January 25, 1965. Hiller letter to FS-100 requesting that FAR 133 be reviewed and that an interpretation of this regulation be issued to enable operation under this regulation, provided adequate structural substantiation is furnished, but that complete certification requirements under the applicable Part 6 need not be applied.

b. February 2, 1965. WE-412 letter to Hiller disapproving their proposal for operations at a 3500 pound maximum weight because they interpret FAR 133.43(c)(1) and (2) to require that other sections of CAR 6 be met, rather than just the structural flight load requirements.

c. February 19, 1965. FS-100 memorandum to WE-416 requesting recommendations on Hiller letter dated January 25, 1965.

d. March 9, 1965. WE-416 memorandum to FS-100 answering FS-100 memorandum of February 19, 1965. This memorandum explained that disapproval was made to Hiller because FAR 133.43(c)(1) definitely states that "the total weight of the rotorcraft-load combinations must not exceed the maximum certificated weight established for the ro rcraft during its type certification under Part 6".

e. March 17, 1965. FS-100 memorandum to WE-400 stating that urgent workload and critical manpower shortage precluded completion of subject review case prior to May 15, 1965.

f. April 26, 1965. Hiller letter to the Administrator petitioning an exemption to specific paragraphs in FAR 133 and CAR 6 prior to May 12, 1965.

3. FACTS IN THE CASE

a. A regulatory project was initiated as a result of Review Case No. 37 to clarify Section 133.43(c) to allow a rotorcraft-load combination weight for Class B and C loads to exceed the certificated weight providing structural substantiation of the applicable provisions of Part 6 or 7 was shown. The Western Region concurs with this philosphy.

b. Hiller believes that the satisfactory demonstration of the flight characteristics requirements of Section 133.41(c) are acceptable and that it is not necessary to conduct the flight test demonstration proposed by the Western Region which includes hover, climb, cruise at the proposed V_{NE}, approach, and return to hover conditions at a gross weight of 3500 pounds with a 1200 pound Class B load. There is no evidence to date that the flight characteristics requirements of Section 133.41(c) are inadequate for the type of operations proposed.

c. The Western Region points out that Hiller models UH-12L and UH-12L-4 equipped with the T1VO-540-A2A engines cannot meet the hover cooling requirements if the test is conducted out-of-ground-effect at a maximum gross weight of 3100 pounds and, therefore, they feel that Hiller should substantiate the out-of-ground-effect hover cooling regimes at the requested weight of 3500 pounds. The type of operation proposed requires considerable prolonged out-of-ground-effect hovering.

d. The Western Region further points out that with the T1VO-540-A2A engines, hover out-of-ground-effect cannot be accomplished on a sea level standard day at weights in excess of 3245 pounds with the UH-12L-4. Also the VO-540-C2A, hover out-of-ground-effect at sea level on a standard day is limited to 3020 pounds on the UH-12L and to 3000 pounds for the UH-12L-4. Since most sling load operations require that the helicopter hover out-of-ground-effect in order to pickup or deposit the sling loads the Western Region feels that the out-of-ground-effect operating capabilities of the helicopter should be shown in the rotorcraft flight manual at all operating weights. (The Bell 204B can hover in ground-effect at the increased gross weight of 9500 pounds at 1500 feet above sea level and out-of-ground-effect at 9350 pounds at sea level in standard atmosphere.)

e. Review Case 37 permitted Bell to exclude the jettisonable external load weight under the structural landing loads and emergency landing conditions of CAR 7 in evaluating compliance with the weight and center of gravity paragraphs of 133.43(c), for their Model 204B helicopter as a Class B rotorcraft-load combination.

4. CONCLUSIONS

a. In view of Review Case No. 37 and the regulatory project discussed in 3(a) an exemption is not considered necessary to approve Hiller's request.

b. Hiller Model UH-12L series helicopters should be permitted to operate under FAR 133 at a jettisonable rotorcraft-load combination weight, in excess of 3100 pounds wherein the release of the Class B load would return the rotorcraft gross weight to 3100 pounds or less, provided:

(1) The affected structure has shown compliance with CAR 6 at the increased weight requested, and

(2) The flight characteristics of Section 133.41(c) for Class B loads has been satisfactorily demonstrated, and

(3) The applicable sections of CAR 6, Subpart E powerplant installation have been substantiated at the increased weight requested, particularly as pertains to cooling.

(4) The rotorcraft-load combination flight manual contains adequate hover out-of-ground-effect performance information to show what loads are possible at various temperatures and altitudes.

REVIEW CASE NO. 56 LOCKHEED MODEL 382 (C-130E); DELETION OF FIRE SHIELDS FROM ALUMINUM ENGINE MOUNT SUPPORT BEAM (Issued 7 July 1965)

1. INTRODUCTION

a. The Southern Region has type certificated the Lockheed-Georgia Company Model 382 (Military C-130E) airplane under Part 9a, Civil Air Regulations, Aircraft Airworthiness; Surplus Military Aircraft, which specifies the requirements of Part 4b, CAR, effective on October 1, 1959, including Amendments 4b-1 to 4b-11. The applicable Part 4b required protection of the structure against deterioration or loss of strength in service.

b. During certification of the Lockheed Model 382 (C-130E), the aluminum engine mount support beam or rails were required to be protected by fire shields. The first Model 382 produced had these fire shields and now Lockheed desires to delete these shields on subsequent aircraft. Their request is based on recent acceptance of their high energy rotor review case, which was accepted in part due to satisfactory service history and acceptance of their anti-ice system.

c. The Southern Region does not consider a negative service history of engine fires as substantiating data for nacelle fire shield removal. Also, review cases for other components (rotors) do not have a bearing on the fire shield removal, nor does the acceptance of the anti-ice system.

2. CHRONOLOGICAL HISTORY

a. The Southwest Regional Office letter dated May 5, 1960, advised Lockheed-Georgia Company that a review of the report of the C-130A nacelle fire test conducted by the Federal Aviation Agency Technical Development Center at Indianapolis, Indiana, disclosed that the damage to the engine mount structure during the fire tests indicated that the mount would not retain the engine in position in case of severe Zone 2 fires in flight. As firewall integrity is dependent upon the engine remaining in position, revisions to the mount structure must be incorporated.

b. Lockheed letter dated May 25, 1960, advised Southwest Regional Office of a proposal to install stainless steel shields to protect the engine mounts.

c. Lockheed letter dated July 30, 1964, advised Southern Regional Office that the data on the C-130A nacelle would be compared to that on the C-130E nacelle and submitted to that office.

d. Lockheed letter dated August 10, 1964, advised Southern Regional Office that a review of the fire test (reference item 2.a.) and the new design features incorporated in the C-130B and C-130E engine mounts as compared to the C-130A mounts made shielding of the aluminum mounts unnecessary.

e. Southern Regional Office letter dated September 3, 1964, to Lockheed replied to the claims made in the August 10, 1964, Lockheed letter and advised that negative fire experience is not acceptable to substantiate the engine mounts.

f. Southern Regional Office letter dated September 15, 1964, to Lockheed elaborated on the September 3, 1964, letter (item 2.e., above) concerning the catastrophic results which could result from a fire with aluminum engine mounts

g. Lockheed letter dated October 23, 1964, advised Southern Regional Office that fire shields, substantially the same as submitted by Lockheed in 1960, would be released for the Model 382 (C-130E).

h. Lockheed letter dated February 23, 1965, to Southern Regional Office made the point that, based upon trouble-free service history (negative service experience), approval was granted in the review case on high energy rotors and approval, on this basis, was requested for the Model 382 ice protection system.

i. Southern Regional Office letter dated February 26, 1965, advised Lockheed that approval for dispatch and flight into known icing conditions was granted on the basis of the technical data submitted and not on negative service experience.

j. Lockheed letter dated March 15, 1965, requested Southern Regional Office to reevaluate the request for approval of the aluminum engine mounts to the same criteria used in the review case on high energy rotors and the approval to fly into known icing conditions (negative service experience).

k. Southern Regional Office letter dated March 22, 1965, advised Lockheed that no justification was found to permit removal of the fire shields from the engine mount and, consequently, a review case would be requested from the Washington Office.

l. Lockheed letter dated April 21, 1965, requested Southern Regional Office to submit the review case to Washington since compliance with Section 4b.487(e), CAR, is met by the type design without fire shields.

3. FACTS IN THE CASE

a. During the proposed certification of the Lockheed Model 282A (C-130B) by the Southwest Regional Office, it was determined that fire shields for the engine mount beams would be required (item 2.a.). The shields were required to maintain nacelle fire integrity. Lockheed agreed to install the shields in the Model 282A (item 2.b.). A full scale nacelle fire test was conducted November 5, 1958, by the Technical Development Center, which indicated the need for additional fire protection.

b. During certification of the Model 382 (C-130E), the engine mount beam fire shields were accepted to show nacelle fire protection adequacy (item 2.e. and f.). Lockheed now wishes to delete the fire shields (item 2.j.) on the basis of results of their review case on high energy rotors which they understand was based, in part, on trouble-free service history and acceptance of the anti-ice system. Lockheed was advised (item 2.k.) that approval for fire shield removal could not be granted on the basis of their letter dated March 15, 1965, (item 2.j.) and that a review case was being forwarded to the Engineering and Manufacturing Division, FS-100, for evaluation as they requested.

c. Sections 9a.2(i) and 4b.10, CAR, provide for special conditions and/or requirements to require an adequate level of airworthiness where specific existing regulations are not adequate.

d. Section 4b.300, CAR, requires that the airplane shall not incorporate design features which experience has shown to be unreliable or hazardous and that the suitability of all questionable design be established by tests. Currently there are no civil aircraft in the transport category that incorporate an aluminum engine mount, and the evaluation of the fire tests on the Model 382 does not show compliance with this section due to the hazardous condition created by an engine fire.

e. Section 4b.301, CAR, requires that materials used in the aircraft structure be established on the basis of experience or tests until such materials will ensure the proper strength assumed in the design data. Aluminum structures without proper protection could not show compliance with this requirement when considering an engine fire.

f. Section 4b.304(a), CAR, states that all members of the structure shall be suitably protected against deterioration or loss of strength in service due to weathering, corrosion, abrasion, or other causes. The removal of the fire shield on the Model 382 would not show compliance with this requirement since there would be no protection afforded against loss of strength under conditions of high heat, and it has always been required to provide adequate protection in hot areas of an engine. (e.g., Stainless steel has been required in areas where exhaust gases impinge on the structure.)

g. Section 4b.486(a), CAR, requires that firewalls be constructed in such a manner that no hazardous quantity of flame can pass from one compartment to another portion of the airplane. Since the engine did sag as a result of a nacelle fire causing the firewall to fail, compliance with Section 4b.486(a) could not be shown. The engine will sag when the mount beams are distorted or damaged by heat and fire. This sagging condition was shown by the full scale nacelle fire tests.

h. Section 4b.487(e), CAR, requires that the nacelle be constructed such that the probability is extremely remote for an accessory section fire to pass to another nacelle zone. During the nacelle fire test, pictures taken of the test showed that the upper cowling separated at or near the aft side of the firewall. This separation could allow flame passage from one zone to another; therefore, compliance with Section 4b.487(e) has not been shown when the engine sags. During the fire test, the engine was prevented from falling by guy wires. Lockheed contends that the cowl splitting would have no adverse affect on fire protection as fire did not pass into Zone 3 during the fire test. Lockheed also states that firewall integrity is not dependent upon retaining the engine. It appears, after review of the nacelle fire test pictures, that the fire did not pass from Zone 2 to Zone 3 due to the location of the artificially induced fire and the fact that the nacelle angle of attack was fixed.

i. Lockheed objected to the requirement for engine mount fire shields during Model 382 type certification. After several conferences and letters dated September 3 and 15, 1964, however, Lockheed agreed to install the shields per their letter dated October 23, 1964. It is evident that Lockheed agreed to the fire shield installation to expedite approval of this aircraft.

j. The basis of acceptance of the review case for the high energy rotors does not have any bearing on the fire shield removal as non-use of the air conditioning rotors does not compromise safety. Also, these components are not related to the fire shields in any way. Acceptance of the argument relating to service experience as a basis for removal of the fire shield could logically be extended to cover all aspects of the fire protection provided for this aircraft.

k. Section 4b.490(b), CAR, requires that consideration be given to the effect on adjacent parts of the airplane of heat within designated fire zones within the combustion, turbine, and tailpipe sections of turbine engines. The Lockheed Model 382 will not be in compliance with this requirement with the deletion of the fire shield which provides the only protection for the engine mount in the event of a fire.

l. The approval of the aircraft ice protection referenced in Lockheed letter of March 15, 1965, was granted due to compliance with Section 4b.460, CAR. Compliance was shown by test data obtained during Air Force Category II flight test; therefore, ice protection acceptance has no bearing on fire shield removal as suggested by Lockheed letter. Although in the March 15, 1965, letter, Lockheed requested a finding by March 19, 1965, the request is no longer pertinent since Lockheed has recently initiated construction of two aircraft sets of shields using temporary tooling.

4. CONCLUSIONS

The Lockheed request to delete Model 382 engine mount beam fire shields cannot be granted for the following reasons:

(1) There are no compensating factors which provide an equivalent level of safety; therefore, fire protection would be compromised with fire shield removal.

(2) Acceptance of the high energy rotor review case has no bearing on the requirements for the nacelle fire shields, since fire experience service history is not available. Also, non-use of air conditioning system (high energy rotors) does not compromise safety.

(3) Aircraft ice protection was substantiated satisfactorily in compliance with Section 4b.640, CAR, and has no bearing on fire shield removal.

(4) Nacelle fire test data indicates that a satisfactory level of safety could not exist without the fire shields.

(5) The economic aspects of this installation, as suggested by Lockheed, should not be a consideration in this case due to the safety factors involved.

(6) Nacelle fire test pictures show that compliance with Section 4b.487(e), CAR, has not been met for simulated nacelle angles of attack.

(7) Lockheed has not considered related requirements of safety, design, and construction; i.e., Sections 4b.10, 4b.300, 4b.301, 4b.304(a), and 4b.490(b).

REVIEW CASE NO. 57 USE OF AUTOPILOT AS STABILITY DEVICE WHEN PITCH TRIM COMPENSATOR IS INOPERATIVE - DC-8 (Issued 16 September 1965)

1. ORIGIN AND PROBLEM

On April 28, 1965, the Douglas Aircraft Company requested that the Agency issue a review case which would permit ". . . allowing the DC-8 to be certificated to the present operating placards using the autopilot when the pitch trim compensator is inoperative . . .". To justify their contention that the autopilot could be used in lieu of the PTC, Douglas presented proposed modifications to the autopilot system and a description of the flight test program which they felt would demonstrate that the autopilot could be used as the equivalent of a stability device.

2. CHRONOLOGICAL HISTORY

a. April 28, 1965. The Douglas Aircraft Company, submitted to the Western Region a request for a review case to use the autopilot as an equivalent stability device in the high transonic flight regime when the PTC is inoperative on their DC-8 aircraft.

b. April 30, 1965. Western Region memorandum to Washington, transmitting the Douglas request with the recommendation that favorable consideration be given to the proposal.

c. June 2, 1965. The Douglas Aircraft Company, submitted to the Western Region the data obtained from the flight tests. In addition, Douglas proposed certain revisions to their original request, based on the flight test experience.

d. June 11, 1965. Western Region memorandum to Washington, transmitting the June 2 Douglas letter and copies of the flight test data.

e. July 16, 1965. Western Region memorandum to Washington, transmitting recommendations based on their review of the Douglas flight test data.

3. REGULATIONS AFFECTED.

CAR 4b as effective December 31, 1953, and Amendment 4b-1 through 4b-11. These were the regulations under which the DC-8 was type certificated.

a. CAR 4b.150 - Stability, General

This section states, in part; "suitable stability and control feel (static stability) shall be required in other conditions normally encountered in service if flight tests show such stability to be necessary for safe operation."

b. CAR 4b.151 - Static Longitudinal Stability

This section requires certain characteristics of the elevator control forces and friction, namely:

(1) A pull is required to obtain and maintain speeds below the trim speed and a push is required to obtain and maintain speeds above the trim speed. (Stable slope of the stick force curve.)

(2) Friction band within 10 percent of the trim speed.

(3) A substantial change in speed is indicated to the pilot by a perceptible change in stick force.

c. CAR 4b.155(a) - Stability during Cruising; Landing Gear Retracted

This paragraph requires a stable slope of the stick force curve at all speeds, between 1.3 V_{S1} and V_{NE}, obtainable with a stick force not in excess of 50 pounds, with the airplane trimmed for level flight with maximum cruising or 75 percent maximum continuous power, whichever is greater.

4. BACKGROUND INFORMATION

a. The present generation of swept-wing turbojet transport airplanes cruise in normal operation at speeds which carry them into the transonic speed range. It is in this range that compressibility begins to have an effect on the flight characteristics of the airplane. If the speed increases from a trimmed condition in this regime, the effect of compressibility begins to significantly alter the pressure distribution over the wing, due to shock wave formation and flow separation. This usually begins to occur at Mach numbers of approximately 0.82, depending on the configuration.

b. As a result of compressibility effects, none of the U.S. certificated large swept-wing turbojet transports, except the B-727, have been able to meet the high speed, static longitudinal stability requirements, without using some type of stability augmentation. The overall affect of compressibility is a rearward shift of the wing aerodynamic center. At a certain speed above the trim speed the aerodynamic center has moved so far aft that the pitching moment caused by lift has become unstable. At this point, and at speeds above this, an increase in speed is met with a decreasing push (or increasing pull) stick force. This is known as the Mach tuck; an increase in speed produces an unstable pitching moment tending to increase the speed even further. This occurs on most of the presently U.S. certificated swept-wing turbojet airplanes. Those on which it occurs have had to use some type of stability augmentation to compensate for this condition.

c. The Douglas Aircraft Company chose to use a stability augmentation system known as the pitch trim compensator (PTC) on their DC-8 airplanes. The PTC provides the required stable stick force curve and control feel by exerting a speed sensed, incremental pull force on the control column. In this manner the pilot input remains in the correct sense. He has to exert an increasing push force (against the pull input of the PTC) to obtain increasing speeds.

d. Shortly after the DC-8 entered service, a campaign was begun, primarily by the Airline Pilots Association (ALPA), to have the PTC removed from the airplane. The ALPA has been, and is, the primary opponent of the PTC. They contend that the major problem is the erratic, unwanted operation of the device. Investigation of the mechanical reliability reports indicates there have been very few reported incidents involving malfunction or unwanted operation of the PTC. The ALPA, however, is of the opinion that many cases of unwanted PTC operation occur, but very few are reported. A study of the problem was undertaken not only by the Agency, but also by Douglas and Giannini Controls Corporation, the manufacturer of the PTC, in an effort to make the device more reliable. The results of this study included:

(1) Issuance of alert information to our inspectors to assure that all DC-8 operators are complying with the Douglas rigging specifications.

(2) Verification that all DC-8 operators have complied with Douglas Service Bulletins 27-160 and 27-161 covering installation of the 80 percent PTC extension warning light and relocation of the nose-down trim stop.

(3) Incorporation of the requirement for a one-time inspection of all the pitch trim compensator computers in accordance with the procedures contained in Giannini Service Bulletin No. 27-20 into the carrier's operation specification and inclusion in the carrier's maintenance manual of the inspection and test procedures contained in the Service Bulletin No. 27-20.

(4) Investigation and approval of a Douglas modification to change the elevator trail angle to the stabilizer.

e. The Agency feels that these modifications represent a satisfactory correction to the problem. The ALPA, however, did not share this opinion and continued their efforts to have the PTC installation removed. On April 16, 1965, the ALPA submitted to the Agency a petition for rulemaking. This proposal would amend the FARs such that the use of any stability augmentation or stall prevention device which acts on the primary control or trim surfaces would be prohibited. The ALPA also submitted proposed airworthiness directives which would require the aforementioned devices to be removed from any aircraft using such systems. It is important to note that the ALPA proposal covers not only the PTC, but the Mach trim device and stick pusher installations as well. It does not mention yaw dampers, power units to operate flight controls, or autopilots.

5. FACTS IN THE CASE.

a. The Douglas request is a result of this pressure. The letter requesting the review case states: "Due to the many inputs from various airlines together with organizations such as ALPA, your immediate consideration of allowing the DC-8 to be certificated to the present operating placards (M_{MO}/V_{MO}) using the autopilot when the PTC is inoperative would be greatly appreciated."

b. The original Douglas request stated that, once certain modifications are made to the existing autopilot system, it will achieve the same goal as that of the PTC. The modifications Douglas proposes are:

(1) Aural warning upon any autopilot disconnect in the clean configuration;

(2) Adding the autopilot disconnect light to the master warning light;

(3) Elimination of the automatic cutoff feature; and

(4) Restriction of the airplane nose-down pitch commanded authority of the autopilot to approximately 10^{o}, in lieu of the presently approved limit of 15^{o}.

In addition to these autopilot modifications, Douglas proposes to add an aural stabilizer-in-motion warning.

c. Douglas also submitted a proposed flight test program which they contend will demonstrate the adequacy of the autopilot to perform the intended function of the PTC. The flight test program includes:

(1) Hardover protection - Hardovers will be conducted throughout the speed range.

(2) Upset protection, high and low altitude - longitudinal and lateral upsets to the maximum autopilot capacity for various configurations and altitudes.

(3) Performance and handling qualities at different airplane and autopilot configurations. The autopilot will be demonstrated to be capable of flying the DC-8 at (a) maximum climb speed, (b) high speed cruise (0.82M to 0.85M), and (c) descent at M_{MO} and slowdown. Disconnects will be conducted during these tests and turbulent flight will be demonstrated, with the autopilot disconnected and connected.

d. The above flight test program has been completed and the resulting data reviewed. The tests were either witnessed or performed by Agency personnel. The important results or features of the flight tests are as follows:

(1) Hardover Protection

Hardovers were conducted about all axes. Recovery was initiated three seconds after recognition, and effected within the presently approved speed envelope of the DC-8.

(2) Upset Protection

Upsets were conducted at forward and aft c.g. positions. The airplane was upset to the maximum autopilot nose-down position (10°) and recovery initiated 15 to 20 seconds later. The design speed was not exceeded. Recovery techniques consisted of thrust reduction, (resulting in additional nose-down pitching) and minimal use of stabilizer trim (recovery effected by elevator alone). Upsets from the maximum bank positions (35°) were in no way critical.

(3) Performance and Handling Qualities

The airplane was flown into and out of the tuck region with the autopilot, which was disconnected at the pilot's discretion. No control problems were encountered. Maximum stick forces at disconnect were approximately 10 pounds, push or pull.

(4) Turbulence

Mild turbulence was encountered during the test program. No major trim changes were experienced upon disconnect.

(5) Emergency Descent

Emergency descents were not considered. The emergency descent procedure requires pitch attitudes in excess of 10^{o}, which is the maximum pitch attitude commanded by the autopilot.

e. As a result of the flight tests, Douglas in their June 2 letter, proposed a revision and an addition to their original request. The revision is to retain the automatic cutoff feature (A.C.O.). The addition is a request to allow DC-8 operation to 0.82M with both autopilot and PTC inoperative.

(1) The flight tests were conducted with the ACO disconnected. It was found that with this feature eliminated, the pilot could mistrim the stabilizer by manually overriding the autopilot with the control column, due to the trim followup circuit in the autopilot. The Western Region stated, in their July 16 memorandum, that serious control problems can result from this mistrim at high Mach numbers and cruise altitudes. Douglas contends that equivalent safety is provided by the aural stabilizer-in-motion warning and the aural autopilot disconnect (pilot or ACO) warning.

(2) The Douglas request to have the DC-8 approved to 0.82M with the PTC and autopilot inoperative is based on the results of the upset tests. Any autotrim movement during the upsets was found to be destabilizing. Douglas, therefore, contends that the maneuver would be less critical with the autopilot disengaged. In their memorandum dated July 16, 1965, the Western Region states that "Based on the longitudinal stability data obtained during the type certification of the DC-8, the DC-8 can demonstrate cruise stability to 0.83 Mach number with the PTC and autopilot inoperative." The PTC provides stability augmentation from 0.83M, where tuck is first encountered, to M_{MO}, which is 0.88M. At the time the DC-8 was certificated, it was possible for M_{NO} and M_{NE} to be the same, due to the provision that a speed spread between V_{NO} and V_{NE} was not required at altitudes where V_{NE} was limited by compressibility. The aural overspeed warning activated at M_{NO}/M_{NE}, which was 0.05M into the tuck regime. For this reason, at the time the DC-8 was certificated, it was felt that an additional incremental Mach number should be imposed on jet aircraft when the stability augmentation device became inoperative. This is the basis for the present 0.78M limitation for the DC-8 with the PTC inoperative. Two points should be indicated here. First,

there is no regulatory basis for the imposition of an incremental Mach number as was done on the DC-8. Second, had Douglas selected M_{NO} as 0.83M, rather than 0.88M, the PTC would not have been necessary. The regulations to which the DC-8 was certificated required cruise stability up to M_{NE}, but M_{NE} could be equal, and in the case of the DC-8 was equal, to M_{NO}. Since the airplane was stable to 0.83M, no stability augmentation would have been required.

(3) If the Douglas request is granted, M_{MO} for the PTC and autopilot inoperative configuration will become 0.82M. This will require the aural overspeed warning to be reset. Special Regulation 450A requires that the aural warning activate at M_{MO} + 0.01M, or 0.83M for the configuration being discussed. Since tuck begins to occur at 0.83M, setting the aural speed warning at this value would provide the pilot with warning at the initial tuck Mach number and is therefore considered to be satisfactory.

f. A similar request, for the use of the autopilot as a speed stability device in lieu of an automatic stability device, has been previously received by the Agency. Convair made this request in May 1960 and then again in October 1961 for their Model 880-22 aircraft. This request was denied in both instances. The basis for denial was that the autopilot was not considered to be a satisfactory device for use in producing the control feel required in CAR 4b.150 or the static longitudinal stability of 4b.151 or 4b.155. It is interesting to note that the Western Region recommended accepting the Convair request, on the basis that the autopilot would do the equivalent job of a speed stability system.

g. Since the time of the Convair request, however, more operating experience in the transonic speed range has been gained. In the justification for their regulatory proposals, the ALPA states that the airline pilots can fly in this tuck regime, with no augmentation, with little or no difficulty and often do so. This statement is qualified by the provision that the pilot is aware that he is in this regime, and has been given instruction as to the control characteristics which can be expected and the proper techniques to be used.

h. The Boeing and Convair models use the Mach trim system. In this system the adjustable stabilizer is used, rather than the elevator, as the PTC does. The system is automatically actuated at a certain Mach number and retrims the stabilizer to counteract the tuck tendency. If the speed increases, the amount of nose-up trim increases; if the speed decreases, the amount of trim decreases. In this manner it is only the elevator control which is affecting the increase or decrease of speed, since the pitching moment due to tuck is compensated for by the Mach trim system.

i. As indicated in the basis for denying the previous Convair request, an autopilot cannot perform the function performed by either the PTC or the Mach trim system, since it does not provide the required control feel or stable stick force curve. This, however, applies only to the stability regulations aspect of the problem. The PTC and the Mach trim devices perform essentially two functions. First, they provide the required control characteristics of the stability requirements. (proper control feel and stable stick force curve.) Second, during high speed cruise operation, they automatically and continually provide opposing trim or control inputs to nullify the Mach trim effect. The autopilot can perform this function, in that it is constantly providing positive control to hold a given altitude or attitude, and will act against the tuck tendency to do this.

j. The Sud-Aviation (French) Caravelle, which is U.S. approved under CAR 10 and has an M_{MO} of 0.775M, does not have a Mach trim system, although a slight tuck exists. The S.G.A.C. Approved Airplane Flight Manual contains a note in the normal operating procedures section which states: "A slight reduction in elevator force may be experienced. This fact, localized between Mach .75 and Mach .79, is not critical."

k. The deHavilland (British) Comet 4C, also U.S. approved under CAR 10 has an M_{MO} of 0.76M, and has a Mach trim system, but it does not operate until a Mach number of 0.79 is attained. This is, of course, above the normal operational envelope, and is used only as a safety device against inadvertent overspeeding.

l. If the Douglas requests are approved, the autopilot will become a required piece of equipment for normal cruise operation above 0.78M. The controllability characteristics of the airplane in the event of failure of the autopilot, therefore, must be given consideration. It would appear that the operating limitations and restrictions applied to an inoperative PTC would be applicable also to an inoperative autopilot being used in the same capacity.

m. If approval is granted, experience gained through the use of the autopilot as the equivalent to a stability device, and its affect on DC-8 operations, will be given consideration in Regulatory Project 1673, which was established in response to the ALPA petition. This project will also consider the applicability of the stability requirements in the transonic speed range, based on the experience gained by the present generation of swept-wing transport jet aircraft.

6. <u>CONCLUSIONS</u>

a. All of the presently U.S. certificated large swept-wing turbojet airplanes, except the B-727, have had to use stability augmentation to meet the high speed cruise static longitudinal stability requirements.

b. Although the Agency is of the opinion that the corrective action taken as a result of the study conducted by Douglas, Giannini, and the Agency, represents a satisfactory correction to the problem, the ALPA continues to maintain that the PTC is a potential hazard to safe operation, and has petitioned the Agency for regulatory action prohibiting such devices and their removal from existing aircraft.

c. The modifications to and use of the autopilot as proposed by Douglas should make the system eligible to use in lieu of the PTC device. The modifications incorporate features to provide the pilot with an adequate warning of unwanted disconnects, and an aural stabilizer-in-motion warning.

d. The Douglas flight test program has successfully demonstrated the capability of the autopilot to fly the DC-8 in the high speed cruise regime. The pilot has ample time to recover from upsets and can easily effect recovery from malfunction or disconnect within speed and acceleration limits. Additional safety is provided by the aural stabilizer-in-motion and autopilot disconnect warnings.

e. The Douglas proposal to retain the automatic cutoff feature is acceptable, as it will prevent mistrimming in the high Mach altitude regime should the pilot overpower the autopilot.

f. The Douglas request to allow operation of the DC-8 at an M_{MO} of 0.82M with the PTC and autopilot inoperative is acceptable. The DC-8, under the regulations to which it was certificated, can legally operate at 0.82M without any stability augmentation. The 0.05M increment presently imposed is not necessary from a safety standpoint, provided that an aural overspeed warning is provided to activate at a lower value corresponding to an M_{MO} equal to 0.82M.

g. The autopilot will not provide the stable slope of the stick force curve nor the control feel required by the regulations. It will, however, perform the operational function of the PTC and the Mach trim; providing constant, automatic trim to compensate for the effect of compressibility.

h. Both the Caravelle and the Comet 4C, which are U.S. approved under Part 10 do not use Mach trim systems within the normal flight envelope (M_{MO}).

i. When used in lieu of the PTC, the autopilot will be required equipment for normal operation above 0.78M, and appropriate operating limitations and restrictions must be established for operation with the autopilot inoperative.

7. <u>FINDINGS IN THE CASE</u>.

a. The Douglas request to allow the DC-8 to be certificated to the present operating limitations (V_{MO}/M_{MO}) using the autopilot as a stability device when the PTC is inoperative is approved, subject to the following conditions:

(1) The autopilot is modified in accordance with that system used on the test aircraft, namely;

(a) Aural warning upon any autopilot disconnect in the clean configuration;

(b) Adding the autopilot disconnect light to the master warning light; and

(c) Restriction of the airplane nose-down pitch commanded authority of the autopilot to approximately 10°.

(2) In addition to the above modifications to the autopilot, the stabilizer-in-motion aural warning must be incorporated.

(3) Information for use of the autopilot is incorporated in the appropriate section of the airplane flight manual as follows:

(a) The present operating limitations which apply to an inoperative PTC apply also to an inoperative autopilot being used as a stability device.

(b) Emergency procedures, including

1 Recovery from malfunction or unwanted disconnect. This should include expected airplane response, control forces and sense of movement, and effect of Mach tuck; and

2 Recovery from upsets due to gusts and turbulence.

(c) Normal operating procedures.

(4) In addition to the above changes to the airplane flight manual, a note is to be added to the emergency descent procedures to inform the pilot that if the autopilot is being used in lieu of the PTC, it must be disconnected prior to initiating an emergency descent.

(5) An evaluation is made of the capability of the autopilot to operate the DC-8 in severe turbulence. This may be accomplished by simulator studies.

b. The Douglas request to allow the DC-8 to be certificated to an M_{MO} equal to 0.82M with the PTC and autopilot inoperative is approved, subject to the following conditions:

(1) Whenever both the PTC and the autopilot are disengaged or inoperative an aural warning shall occur at speeds greater than 0.83M.

(2) Appropriate limitations, procedures and information must be included in the appropriate section of the airplane flight manual.

(3) If the above conditions are not met, the present 0.78M limitation for PTC inoperative operation will apply to operation with both the PTC and autopilot inoperative.

REVIEW CASE NO.58 FIRE RESISTANT FUEL LINES IN THE ENGINE COMPARTMENT OF THE CESSNA MODEL 188 AIRPLANE (Issued 10 December 1965)

1. INTRODUCTION

a. It was found during the verification of compliance program for issuance of a provisional type certificate on the Cessna Model 188 airplane that an aluminum alloy fuel line was installed between the fuel pump inlet and the fuel strainer in an area where an engine fire could exist. Cessna agreed to replace the aluminum alloy line with a fire resistant flexible hose assembly. Later, during the Type Certification Board meeting held at the Cessna Aircraft Company, Wichita, Kansas, on August 13, 1965, for the provisional Model 188, Cessna objected to the Central Region interpretation of FAR 23.1183 which made this replacement necessary and requested that Washington Headquarters review the case.

b. A difference of opinion about the fire resistance of aluminum alloy lines has existed for well over two years and has resulted in changing the fuel lines of at least one other Cessna airplane model.

c. As defined in FAR, Section 1.1, "Fire Resistant..... (2) With respect to fluid carrying lines, other flammable fluid system parts, wiring, air ducts, fittings, and powerplants controls, means the capacity to perform the intended functions under the heat and other conditions likely to occur at the place concerned."

2. CHRONOLOGICAL HISTORY

a. During type certification of the Cessna Model 206 airplane, the Federal Aviation Agency Central Regional Office was concerned that alluminum alloy lines carrying fuel might burn through within a short time after exposure to flames or high temperatures. In a letter to Cessna dated July 2, 1963, the region advised it would be necessary to fabricate the fuel injector return line of material shown to be capable of withstanding a five-minute exposure to a flame with a temperature of 2,000 $\pm$ 50^{o} F.

b. In a reply dated July 8, 1963, Cessna attempted to show that, by quoting from Part 3 of the Civil Air Regulations, aluminum alloy was a satisfactory material inasmuch as fire is not "likely" in Part 3 aircraft powerplants.

c. Verbal discussion followed, and, in a letter dated July 22, 1963, the Central Regional Office quoted Cessna as advising that several Cessna airplane models had been manufactured with nonfire resistant lines and requested that Cessna submit a program for replacing the lines on these airplanes with fire resistant lines and fittings.

d. In their reply of August 9, 1963, Cessna stated they had advised the FAA that a large number of models with installations similar to Model 206 had been built and had accumulated thousands of hours of satisfactory service experience. Cessna said that they yielded to the Regional interpretation of fire resistance for the 206 airplane only to obtain a type certificate.

e. The Regional Office, in a letter dated September 6, 1963, suggested that if Cessna did not believe that fire tests in accordance with the generally applied standard of exposure to a 2,000 $\pm$ 50° F. flame for a period of five minutes were representative, they should conduct investigations to substantiate less severe conditions.

f. Cessna's letter of September 20, 1963, reiterated a previous contention that the Region was making interpretations contrary to the written regulations without giving substantiating data.

g. In their letter of October 9, 1963, the Regional Office stated they could find no basis supporting the Cessna objection to complying with CAR 3.638(c) relative to fire resistant lines and asked Cessna to comply.

h. Cessna wrote to FAA Headquarters, Washington, D.C., on October 21, 1964, presenting its case.

i. Washington replied to Cessna on May 7, 1965, pointing out that the requirements for lines and fittings carrying flammable fluids in any area subject to engine fire conditions had their background in the 1956 Annual Airworthiness Review. Part 3, Amendment 3-2, resulted because of adverse experience with powerplant fires due to difficulties in fuel system plumbing. Currently accepted procedure for determining fire resistance was reviewed but it was noted that the applicant was free to substantiate alternate test fire conditions similar to those likely to exist in the actual installation. Washington further indicated the FAA planned to conduct actual fire tests of aluminum alloy tubing.

j. The Regional Office wrote to Cessna on September 7, 1965, about Cessna Model A185E and asked what action was proposed to show compliance with the fire resistant requirements since aluminum alloy lines in the fuel system were not considered acceptable. The Region also advised Cessna that a review case was being processed.

k. In a letter dated September 9, 1965, Cessna replied that over 900 aircraft of the 185 series and over 22,000 other model Cessna aircraft were built with aluminum alloy lines. The company commended the Regional Office for processing the review case.

l. In a memorandum dated September 14, 1965, the Regional Office requested a review case on this subject.

m. The Regional Office wrote Cessna again on September 22, 1965, about lines in the A185E and other Model 185 aircraft.

3. FACTS IN THE CASE

a. There are two basic points raised by Cessna relative to their position that aluminum alloy lines should be accepted to carry flammable fluids in engine compartments of normal-category aircraft:

(1) Fire is not a condition which should be considered as one "likely to occur" in the engine compartment of normal-category aircraft.

(2) The wording of the regulation makes it clear that aluminum is acceptable in satisfying requirements for fire-resistant components.

b. In consideration of the first point, it is noted that the requirements of FAR 23 relating to fire protection were based upon the premise that the consequences of fire necessitate the incorporation of protective features in the design of all aircraft. Since the inception of these requirements, the Flight Standards Service has always intended that the protection provided be referred to the conditions of heat likely to occur when there is a fire. It is further considered by the FSS that lines carrying flammable fluids within the engine compartment and which are not capable of performing their intended functions under the conditions of heat likely to occur when there is a fire in that compartment would not be acceptable as providing adequate fire protection.

c. It should be noted that non-containment of flammable fluids can have particularly drastic results in the case of normal category aircraft. Because there is no fire detection, fires may burn for an appreciable period before being recognized. Failure of lines during this period could increase the severity of a fire to an uncontrollable magnitude. In addition, since extinguishment means are not required, there may be no way to arrest a fire once sufficient fuel has been provided.

d. A survey of normal category aircraft approved by the various regions indicates that, apart from the Cessna Company, there has been no report of misunderstanding of the intent of the regulations requiring "fire resistant" lines and fittings by any region or manufacturer, either with aircraft approval directly by the Agency or through the delegation option procedure.

e. The foregoing notwithstanding, the issue raised by Cessna indicates that the language of the definition of "fire resistant" is not entirely clear. To preclude any further misunderstanding of the intent of the regulations relating to fire protection, this definition will be revised to indicate, specifically, that conditions associated with fire must be considered.

f. The second point raised by Cessna, that aluminum lines should be accepted as being fire resistant in any case, rests upon the circumstance that, as related to sheet or structural material, fire resistant is defined to mean a material which will withstand heat as well as aluminum alloy. The point is raised by Cessna that the type of application of items to which this term is applied should not govern the meaning of "fire resistant."

g. Concerning this point, it is noted that the regulation clearly intended to specify a different criterion for the determination of fire resistance as it applies to sheet and structure as contrasted to its application to flammable fluid lines and fittings. This intent is self-evident because pains are taken in the regulation to present two different definitions, and the language is clearly different for each.

4. CONCLUSIONS. The arguments presented by Cessna in support of their view that aluminum lines should be accepted as being in compliance with requirements for fire resistance cannot be concurred in as a basis for accepting such lines for the following reasons:

a. The term "fire resistant" has always been intended by FSS to apply to the conditions of heat likely to occur when there is a fire in the engine compartment. Lines and fittings carrying flammable fluids in that compartment which are not capable of performing their intended function under heat conditions likely to occur when there is a fire would not be acceptable.

b. The requirement pertaining to lines and fittings was clearly intended to be different from that applying to sheet and structure and such distinction is plainly evident in the language of the definitions in which the definition applicable to lines and fittings pointedly omits reference to material. Since material is not specified in the regulations, aluminum lines may be accepted if shown to be capable of performing their intended functions under the conditions of heat likely to occur when there is a fire in the engine compartment.

c. Recognizing the problems that have been introduced because of the existing language of the definition of the term "fire resistant" as it applies to systems carrying flammable fluids, a revision to the definition is planned to make it clear that this term is to be applied as discussed in this review case.

REVIEW CASE NO. 59 FIRE PROTECTION OF OIL SYSTEM OF SWEARINGEN SA-26T AIRPLANE (Issued 15 December 1965)

1. INTRODUCTION.

a. Swearingen Aircraft applied for a type certificate on their Model SA-26T aircraft on January 23, 1964. The type certification basis is Civil Air Regulations, Part 3, effective May 15, 1956, including Amendments 3 through 8 plus special conditions. The airplane will be powered by two United Aircraft of Canada Limited (UACL) PT6A-20 engines. The gross weight will be 9,000 pounds for takeoff and 8,050 pounds for landing.

b. The UACL PT6 engine incorporates a 2.3 gallon oil tank integral in the engine accessory case at the aft end of the engine. External engine oil lines are utilized to carry pressure and scavenge oil to and from the forward end of the engine. When the engine is installed, additional oil system components and plumbing are added. These items include an oil radiator (connected into the engine oil scavenge system), fuel/oil heat exchanger (furnished as an optional item by the engine manufacturer and connected into the engine oil pressure system), oil pressure and torque pressure transmitters, and plumbing connecting these components to the engine. These components were to be fabricated of aluminum alloy and located in the engine accessory compartment aft of the induction system plenum. Air for the oil radiator comes from the lower aft induction system plenum bulkhead, through the oil radiator, and is ducted out the top of the nacelle forward of the primary firewall.

2. REGULATIONS AFFECTED. Federal Aviation Regulations, Section 23.1189, (CAR, Section 3.637) which contains requirements relating to means to prevent flammable fluids from flowing into, within, or through engine compartments.

3. HISTORY.

a. CAR, Sections 3.637(a) and (c), were revised under Amendment 3-2 effective August 12, 1957, to incorporate the language essentially as it exists now in FAR, Section 23.1189. The preamble to Amendment 3-2 stated that the intent of the changes to this section was to improve fire protection provisions and set forth specific design criteria. One substantive change was the replacement of the word "into" with the words "into, within, or through."

b. On June 9, 10, and 11, 1965, representatives of the Southwest Regional Office, SW-214, visited Swearingen Aircraft to examine the powerplant installation of the Model SA-26T airplane. It was noted that the entire engine oil system is forward of the firewall, but no shutoff valves or equivalent were provided in the system. Swearingen was advised that, on the basis of this examination, the oil system did not appear to comply with the provisions of CAR, Section 3.637. Mr. Swearingen expressed the opinion that his installation was equivalent to that of other CAR, Part 3, aircraft certificated under the same fire protection requirements and that further substantiation or major design changes were not necessary.

c. Southwest Regional Office, SW-214, confirmed its position by a letter of June 30, 1965, to Swearingen Aircraft.

d. The Eastern Regional Office memorandum of July 12, 1965, to SW-214 confirmed that engine oil lines installed by the engine manufacturer had satisfactorily passed appropriate fire tests.

e. On August 11 and 12, 1965, SW-214 representatives visited Swearingen Aircraft and reiterated their earlier statement that the oil system installed by Swearingen was not in compliance with CAR, Section 3.637. They also informed Swearingen that the engine manufacturer's oil lines were considered fireproof. Swearingen agreed to revise the installation to incorporate fireproof components, except that he felt that he should not have to change the oil radiator as it is part of the engine. He contended that radiators similarly located have been approved on CAR, Part 3, aircraft with wet-sump reciprocating engines and that the oil system of the Swearingen SA-26T is equivalent to the Beech 65-90 King Air which uses the same engine.

f. On September 23, Swearingen was again advised in writing that the oil system in the SA-26T installation must comply with the intent of CAR, Section 3.637. It was agreed, however, that the question would be submitted as a review case.

4. FACTS IN THE CASE.

a. It is clear that the current requirement of FAR, Section 23.1189, does require a shutoff valve for systems carrying flammable fluids within the engine compartment and that this requirement is applicable to this case. The airframe manufacturer may comply either by providing a shutoff valve, or by providing compensating factors to attain an equivalent level of safety.

b. A precedent has been established for accepting oil systems without shutoff valves in turbine-engine-powered aircraft. Equivalency was based in such cases upon provision of a fireproof system and protection of fittings against leakage.

c. Another basis for acceptance of oil systems without shutoff valves could be a demonstration by actual fire tests that the burning of the entire quantity of oil would not endanger the aircraft. Such a demonstration was made, for instance, to establish the acceptability of an arrangement without shutoff valves in a podded engine.

d. Inasmuch as approval may be considered on an equivalent safety basis, it is reasonable to evaluate each component of the oil system with respect to its capability to achieve the applicable safety objective. In this case, this objective is considered to be the containment of oil under fire conditions for a sufficient period to achieve equivalence to the conditions associated with fire tests for fireproof lines and fittings.

e. It should be noted that acceptance of materials other than those generally recognized as being fireproof will impose upon the applicant the burden of providing acceptable substantiation that the oil system will contain the oil for a period of fifteen minutes under the fire conditions each component is likely to be exposed to in the event of a fire in the engine compartment.

f. The applicant, in this case, has agreed to provide fireproof lines and fittings and to isolate the pressure transmitters from the engine compartment. The only point remaining to be settled is whether additional protection in some form must be provided for the aluminum alloy oil radiator.

g. Swearingen refers to Amendment 3-6 of CAR, Part 3, and contends that Part 3 aircraft equipped with turboprop engines need maintain only the same level of safety as that required for reciprocating engine-powered aircraft. It should be noted, however, that CAR, Section 3.11(d), requires a modified airplane to comply with later airworthiness requirements applicable to the modification if it involves a substantially complete redesign of a component or installation.

h. The flow of oil in reciprocating engines would normally be stopped, even without a shutoff valve, if the propeller were feathered. Feathering the propeller in this turbine engine will not stop the flow of oil through the oil system. There is, therefore, a difference related to changes in engines and, since no oil shutoff valve is to be employed, additional steps must be taken to achieve equivalent safety.

i. In evaluating an article, such as an oil cooler, it is reasonable to make a determination of the fire conditions to which it is likely to be exposed in the event of a fire in the engine compartment. If such conditions are established as being less severe than the standard 2000° F. flame, the overall level of safety would not be lowered if these less severe conditions were used as a basis for evaluating the article. As an alternative, the oil radiator could be isolated.

j. Swearingen contends that the SA-26T oil system is equivalent to the Beech Model 65-90 system from the standpoint of compliance with FAR, Section 23.1189, (CAR, Section 3.637). With regard to this point, the approval of the Beech 65-90 does not bear upon the acceptability of the Swearingen installation. The same options, comply or show equivalent safety, apply to any approval under the current regulations.

5. CONCLUSIONS.

a. FAR, Section 23.1189, (CAR, Section 3.637) is specific in requiring means to prevent the flow of hazardous quantities of flammable fluids within, as well as into and through, the engine compartment.

b. Precedent has been established for the acceptance of turbine engine oil systems without shutoff valves on the basis of equivalent safety.

c. Equivalent safety in the past has been based upon showing that, despite the absence of an oil shutoff valve, either the oil system will continue to contain the oil, or release of all oil in the system into an existing fire would not further endanger the airplane.

d. Determination of the ability of the system to continue to contain oil may be based upon an evaluation of the system relative to conditions that have a reasonable probability of occurrence during an actual fire.

e. Although the reciprocating engine and the turboprop engine are considered to employ the same principles of propulsion, there are in fact differences introduced by virtue of the fact that feathering in the first case cuts off the flow of oil in the oil system whereas, in the case of the turboprop, the oil will continue to flow.

f. The burden of establishing equivalent safety rests with the applicant. If a showing of equivalent safety is not made, there is no alternative to requiring a shutoff valve.

g. Based on the information available, Swearingen Aircraft has not demonstrated that the aluminum oil radiator would be acceptable in establishing safety equivalent to that provided when an oil shutoff valve is installed.

REVIEW CASE NO. 60 CESSNA CRAFTED FULL-FLOW OIL FILTERS USED ON CESSNA AIRCRAFT (Issued 27 December 1965)

1. INTRODUCTION.

a. Recently there have been a number of cases of oil filter failures involving Cessna Crafted full-flow oil filters installed on various models of Cessna-built airplanes. Several of these oil filter failures involved complete loss of the oil supply and consequent inflight engine stoppage.

b. Taking cognizance of this situation, the Central Region concluded that a hazardous situation existed and that corrective action should be undertaken.

c. Pursuant of this conclusion, the Central Region developed and proposed an airworthiness directive which would require replacement of the offending Cessna Crafted filter with an improved model developed by Cessna or the original oil screen supplied by the engine manufacturer.

d. Cessna expressed objection to the issuance of an AD, maintaining that the problem could be resolved more effectively through the company service system.

e. Consideration of the various factors involved in this situation led to a conclusion that the best way to present the conclusions reached would be through a review case.

2. CHRONOLOGICAL HISTORY.

a. December 22, 1964. Noting reports of leakage or loss of oil from filters, Cessna issued Service Letter 64-61 covering inspection and installation of the filter.

b. February 16, 1965. Cessna Service Letter 65-17, Item 2, was issued covering installation of a new oil filter gasket.

c. March 16, 1965. Cessna Service Letter 65-3 was issued concerning importance of proper torquing of oil filter adapter nut.

d. May 4, 1965. Cessna Service Letter 65-48 was issued concerning inspection of oil filter adapter nut, adapter oil return hole, and proper torquing of adapter nut.

e. June 8, 1965. Cessna Service Letter 65-49 was issued announcing a new filter design to be available about August 15, 1965.

f. August 11, 1965. CE-210 wrote letter to Cessna noting design of new filter adapter and requesting information relative to number of failures experienced and asking about basis for continuing old filter in service in view of service record.

g. August 13, 1965. Supplement No. 1 to Cessna Service Letter 65-59 announced no-cost replacement of original filters.

h. August 25, 1965. Cessna replied to CE-210 letter (Ref. f), noting that the filter in question is optional equipment and that Cessna does not have a complete record of number of installations in aircraft in service. This letter also pointed out that the company is urging early replacement of all old parts.

i. September 20, 1965. CE-210 wrote to Cessna announcing the Central Region intention to process an AD based upon Cessna Service Letter 65-59.

j. October 12, 1965. Meeting was held between personnel of the Central Region and Cessna to discuss the proposed AD. Cessna expressed objection to its issuance and outlined the modification program being undertaken by the company.

k. October 15, 1965. CE-210 advised Cessna that the proposed alternative was being evaluated.

l. October 26, 1965. Installation of redesigned oil filter was extended to Cessna 310 and Skyknight by Supplement No. 2 to Service Letter 65-59.

m. November 9, 1965. CE-210 letter to Cessna gave notification of the intent of the Central Region to proceed with issuance of the AD.

n. November 12, 1965. Cessna replied to the November 9, 1965, letter from CE-210 enumerating the company objections to the issuance of the AD.

o. November 19, 1965. CE-210 forwarded a proposed AD to FS-100. The Central Region recommended issuance of this AD as an adopted rule.

p. December 2, 1965. Mr. Obed Wells of the Cessna Company visited FS-100 to discuss the proposed AD.

q. December 3, 1965. A letter from Cessna to FS-100 detailed the reasons for Cessna belief that the company program would achieve an acceptable level of effectiveness.

3. FACTS IN THE CASE.

a. The issue in this case arises because the Central Region believes that a hazardous condition exists and its correction demands the immediate issuance of an AD, while the Cessna Company disagrees, maintaining that the problem can be satisfactorily corrected through the company service system.

b. The specific problem involves the Cessna Crafted oil filter which has been installed as optional equipment on a substantial number of Cessna-built aircraft. This filter installation, as originally designed, appears to have been of marginal strength and of questionable serviceability. There have been more than seventy reports of partial or complete oil losses as a result of failures involving this filter assembly and, approximately, twelve of these are understood to have resulted in emergency landings.

c. The circumstance that more than 1,600 of these filters are installed in single-engine aircraft when combined with the fact that loss of oil is very likely to result in engine stoppage results in a situation that must be regarded as potentially hazardous. Cessna does not disagree with this point.

d. Recognizing that the design of the oil filter installation has not proven to be entirely satisfactory, Cessna redesigned it with respect to both its strength and maintenance aspects.

e. The Central Region, becoming concerned that a hazard existed and that it should be corrected as quickly as possible, proposed an AD to require that the original filter be replaced either with the new design, or the oil screen normally furnished as part of the engine. Because the failure rate of the filter installations appeared to be higher during the winter months, presumably, on account of higher oil pressure associated with cold starts, it was proposed that this AD be issued as an adopted rule to make it effective as soon as possible.

f. In argument against the AD, Cessna contended that none of the failures had occurred until after there had been servicing of the filter. This strongly suggests that improper reassembly of the installation is the major factor in these failures. Even though the design might be conducive to wrong assembly, it is suggested that the most logical step to cure the immediate problem would be in the maintenance area.

g. Cessna further points out that, because the company recognized the possible contribution of a marginal design to the occurrence of these failures, the design improvement change mentioned above was carried out. Appreciating the potential hazard involved, Cessna implemented a program of replacement of the old type filters at no cost to the owners. For their own internal reasons and to encourage early accomplishment of the changes, a deadline of January 1, 1966, was set on the no-cost replacement for single-engine aircraft. Because the program was initiated later for them, affected twin-engine aircraft are given until March 1, 1966, to complete the no-cost replacement program.

h. To provide incentive to their authorized service agencies for carrying out the replacements, the normal labor charge associated with doing the job is paid to the agencies upon application after the job has been done. It is felt by Cessna that this will assist materially in producing an active and prompt program of filter replacement.

i. Cessna reports that, as of December 1, 1965, 1,401 kits for installation of the new design filter assembly had been ordered, and that this is running somewhat ahead of the rate of ordering that had been anticipated by the company.

j. With respect to the number of aircraft that are modified, Cessna advised that their procedure of supplying kits only on order and reimbursing the service agencies only after installation will provide an accurate and effective method of the extent of compliance with Service Letter 69-59. It is expected that, because of processing time involved, the exact number of installations involved will not be known until somewhat after January 1, 1966, probably, in late February.

k. With respect to obtaining compliance with the change, it is Cessna's contention that the company service system provides for making the owners aware of the need for the change more effectively than the AD system. In this connection, it is appreciated that the AD system is not very effective in providing individual owners of small aircraft with information relating to mandatory changes. Such owners, for the most part, neither read the Federal Register nor subscribe to the AD card distribution. Thus, in many cases, the only knowledge that these people gain of the existence of an AD comes when their aircraft undergo annual inspection. In a case like this, many of these people would certainly be in unintentional violation of the AD before their annual inspections became due.

l. The 50-hour compliance time proposed for the AD would represent calendar time of two to three months based upon a normally expected utilization of the small aircraft. It also represents two oil change periods based upon Cessna servicing recommendations. For some of the affected aircraft, the actual period of time might well be in excess of a year, in which case nearly two years might pass before compliance would be verified by an annual inspection following the specified 50-hour period. On this basis, it appears that compliance to avoid possible winter hazards would not be very effectively achieved with an AD, even published as an adopted rule.

4. CONCLUSIONS.

a. A potentially hazardous situation requiring early correction exists.

b. This condition appears to result more from a susceptibility to improper installation than from the admittedly marginal design; hence, there is some basis for doubt that the proposed AD speaks to the most appropriate corrective action.

c. An AD issued as an adopted rule would not produce compliance by a sufficiently early date to insure modification prior to the coldest winter weather.

d. It is considered that there is an adequate basis to accept the confidence of Cessna that the company program will result in modifications to a substantial majority of aircraft involved by January 1, 1965. The incentives incorporated into the Cessna program would appear to provide a reasonable basis for this confidence.

e. On this basis, it is recommended that further processing of the AD be suspended and that Cessna be given an opportunity to implement the needed modifications through the company program.

f. It is further recommended that, as soon as the results of the Cessna no-cost replacement program are available after January 1, 1966, this program be resurveyed with respect to single-engine aircraft to determine if it has achieved an acceptable level of effectiveness.

REVIEW CASE NO. 61 REVIEW OF CAR 6.328 WITH PARTICULAR REGARD TO LOCKHEED CL-286 HELICOPTER POWER CONTROL SYSTEM (Issued 31 May 1966)

1. INTRODUCTION.

The Lockheed California Company has requested a review and interpretation of CAR 6.328 with particular regard to the Lockheed Model CL-286 helicopter. Lockheed contends that the CL-286 helicopter power control system complies with CAR 6.328 without backup provisions or complete separation of the dual hydraulic systems. The Western Region contends that the power control system presented by Lockheed does not comply with CAR 6.328.

2. CHRONOLOGICAL HISTORY.

a. October 1, 1964, Preliminary Type Certification Board meeting minutes CL-286 helicopter, Item S-5 power control systems. The Western Region advised Lockheed that there was a common point in the dual hydraulic system in which a single failure such as a crack or rupture of an actuator could result in a loss of both hydraulic systems. It was further stated that Lockheed may be required to provide a manual backup system, unless complete hydraulic system redundancy is provided.

b. In a letter of July 30, 1965, Lockheed requested approval of the CL-286 control system based on past experience and structural integrity of components which would be based on life cycle testing.

c. At Lockheed's request, a team of region personnel visited the Lockheed facility at Rye Canyon on August 8, 1965, to become more acquainted with the hydraulic actuator assembly details.

d. In the letter of September 14, 1965, Western Region stated that they were unable to find compliance with 6.328 because the CL-286 power control system did not provide the capability to land safely in the event of a single failure in the power portion of the control system.

e. Meeting held on September 20, 1965, requested by Lockheed in order that the subject be further discussed. Lockheed's interpretation of CAR 6.328 was stated as being that the mechanical portion statement of the regulation should include the spool of the control valve and the actuator housing. No agreement was reached.

f. Subsequent to the meeting of September 20, 1965, Lockheed submitted data changing their control system by installing additional actuators in series with the existing actuators and providing complete separation of the hydraulic systems. This overall concept was considered satisfactory by the Western Region.

g. On October 7, 1965, Mr. P. Hollowell of Lockheed met with FS-120 personnel. He reviewed the history of the problem and requested guidance on avenues available to Lockheed to obtain approval of the original control system. He was advised that they could show compliance on an equivalent safety basis, or request a review case or apply for an exemption.

h. Lockheed on February 11, 1966, requested a meeting to further discuss the CL-286 control system. Lockheed stated that the addition of the additional hydraulic actuators has resulted in down-grading of the handling characteristics of the helicopter and requested that the Type Board reconsider the original design concept for compliance with CAR 6.328.

i. Lockheed's letter dated February 24, 1966, submitted a detailed failure analysis and service history in attempting to show compliance with CAR 6.328 and requested a review of CAR 6.328 with particular application to the CL-286.

j. Upon learning that the Western Region had not reversed its previous decision based on their submission of a detailed failure analysis, Lockheed informally requested a review case in a meeting on March 9, 1966.

k. WE-130's letter dated March 10, 1966, to Lockheed restated their past position on this matter.

l. WE-100's memo of April 7, 1966, requested a review case.

m. On April 14, 1966, Mr. A. Turner of Lockheed reviewed Lockheed's position on the review case with FS-100 personnel.

3. <u>BACKGROUND</u>.

a. The Lockheed CL-286 helicopter is being certificated under CAR 6 dated December 20, 1956, Amendments 6-1 through 6-7. The development of CARs 6.328 and 7.328 paralleled and was concurrent with the development with CAR 4b.320 with most of the discussion and analysis concentrated on CAR 4b.320. As the wording of the final rules (CARs 4b.320(b)(1) and 6.328) is almost identical, the regulatory history of CAR 4b.320 is pertinent and must be considered.

b. In Draft Release 58-1 of February 24, 1958, it was proposed that rotorcraft be required to have the capability of continued safe flight and landing after any single failure in the power actuation system. The same requirement was proposed for fixed wing transports except the requirement was extended to cover the entire control system and a specific requirement for an independent standby power source was proposed. As a result of industry objections to standby systems and complete duplication of load paths, an immediately available alternate system was specified and the proposal was restricted to the power portion of the control system. The power portion was defined as including such items as valves, lines, cylinders, etc.

c. The revised fixed wing transport proposal and the original rotorcraft proposal were published in Draft Release 58-1C dated December 22, 1958.

In the comments on this proposal, the ATA proposed that only "reasonably probable single failures" be considered in the requirement and that failures such as jammed pistons or broken links be excluded. The AIA proposed that the failure of structural elements not be considered if expected to be remote. The ALPA contended that the power portion included the control surface drive mechanism such as jack screws. The proposal to consider only "reasonably probable single failures" was rejected. The proposal to exclude jamming of power pistons and failure of mechanical elements was accepted, provided they were shown to be extremely remote. The contention that the drive mechanism or jack screw was a part of the power portion was rejected as it was considered to be part of the mechanical portion system.

d. The following proposal was adopted in CAR Amendment 6-4 dated August 24, 1959, and a corresponding CAR 4b amending (4b-11) was adopted with almost identical wording:

"CAR 6.328 Power boost and power-operated control systems. When a power boost or power-operated control system is used, an alternate system shall be immediately available, such that the rotorcraft can be flown and landed safely in the event of any single failure in the power portion of the system or in the event of failure, of all engines. Such alternate system may be a duplicate power portion or a manually operated mechanical system. The power portion shall include the power source (e.g., hydraulic pumps), and such items as valves, lines, and actuators. The failure of mechanical parts (such as piston rods and links) and the jamming of power cylinders need not be considered if such failure or jamming is considered to be extremely remote."

e. Lockheed contends that their system complies with CAR 6.328 on the premise that failure of mechanical parts in the power portion of the control system is extremely remote as detailed in Lockheed's report No. 19089. Lockheed's position is based on the following reasoning:

1. Had it been the intent of the formulators of the regulation to permit proof of improbability of failure of only piston rods and links, and the jamming of power cylinders, the words, "mechanical parts," "such as" and parentheses would have been ommitted leaving only a specific list of components subject to proof.

2. Control system failure modes may be classified into three categories:

   (a) Fatigue or structural failure

   (b) Jamming

   (c) Loss of power (if a portion of the system utilizes power-assist to pilot input)

   System integrity against fatigue failure, or jamming is insured by suitable tests and demonstrations including cycle load and operation tests on the complete vehicle. It is Lockheed's position that all elements of the control system (including the power portion) wherein single load paths exist must be subjected to the same scrutiny for insurance against the probability of a structural failure. It is inconsistent to require duality of certain portions of a system (the power sections) for insurance against structural failures while exempting those elements of the system from the pilot to the power system and from the power system to the rotor from the same duality requirement.

(4) It is reasonable, however, to require duality of the source of power, the delivery circuitry to the actuators and duality of actuators such that a single failure of the power source will not result in loss of control. It is Lockheed's position that the duality requirement applies to prevention of power failure only and does not apply insofar as structural failures are concerned inasmuch as the latter category may be demonstrated to be extremely remote by suitable tests and substantiation.

f. The Western Region's position that the control system proposed by Lockheed for the CL-286 does not comply with provisions CAR 6.328 and/or 6.10 and is premised on literal interpretation of the regulation. Their position is based on the following reasoning:

(1) The CL-286 helicopter control system does not provide the capability to land safely in the event of any single failure in the power operated control system. Cracking of the actuator housing or jamming of the control valve constitutes a single failure in the power portion of the power operated control system. The CL-286 can only be flown on hydraulic power as no effective manual reversion is available. The regulations specifically define, "the power portion shall include such items as valves, lines, and actuators." The last sentence of the regulation, which lists specific items which need not be considered because failure is considered extremely remote, does not negate the main context which considers any single failure in the power portion definition of the regulation. The employment of a single load path for the mechanical portion of the control system is acceptable; however, the actuator assembly housing and the spool of the control valve is not considered part of the mechanical portion of the control system.

(2) The literal intent of CAR 6.328, as well as the equivalent regulations in CAR 3 and 4b to require redundancy has been the objective in certification projects in the Western Region. It is believed that the soundness of this approach has been confirmed by service experience.

(3) Cyclic testing to verify actuator integrity is not considered an equivalent level of safety as improper maintenance can negate laboratory findings by introducing stress risers which would be impractical to detect once an actuator is assembled.

4. <u>ANALYSIS</u>.

a. In the development of the CARs 6.328 and 4b.320, the proposals were made to require capability of continued safe flight and landing after any single failure in the entire control system and after any single failure in the surface drive mechanism. These proposals were rejected and the requirement limited to the power portion of the control system which was defined as including the power source and such items as valves, lines, and actuators. Proposals were also made to limit the requirement by excluding consideration of certain failures in the power portion (failure of mechanical parts such as piston rods, and links and the jamming of power cylinders) if considered to be extremely remote. These proposals were accepted. The proposal to limit the requirement to consideration of only "reasonably probable single failures" was also rejected.

b. In view of the foregoing, it is evident that "any single failure" referred to in CAR 6.328 is not limited to "reasonably probable single failures" (such as loss of power source, failure of seals, leaky junctions, etc.) but does not include all single failures of the power portion of the control system. For example, jamming of the power cylinders and the failure of mechanical parts are excluded if considered to be extremely remote. The wording of the regulation makes it clear that the jamming exclusion is limited to power cylinders as it is a specific exclusion and not an example or modified by the phrase "such as." The regulation also clearly excludes purely mechanical parts (such as piston rods and links) whose essentially sole primary function is to directly transmit a control force or motion. The primary question is whether the regulation excludes parts which have functions (such as involved in converting an energy source into a control force) in addition to or in lieu of a purely mechanical function.

c. To answer this question, the intent of the regulation must be considered. The fact that CAR 6.328 is limited to the power portion of the control system indicates that the basic intent was to preclude catastrophic results from the failures anticipated on the basis of past unreliability of this portion of the system. Service experience (Lockheed report 19089) indicates that the basic cause for this unreliability was the additional failure modes over and above those associated with purely mechanical systems. Such failure modes primarily involve loss of power and include leaks from structurally insignificant cracks or yielding; leaky seals, valves and junctions; stuck valves; clogged passages; fluid contamination; lack of fluid; electrical shorts; open circuits; overheating; etc. These additional failure modes result in additional inherent risks for power system parts which have nonmechanical functions. Consequently, such parts cannot be considered mechanical parts even though they also have a mechanical function and it is considered to be reasonable and the intent of the regulation to require the capability of continued safe flight and landing after the failure of such "nonmechanical" parts.

d. If the additional risks associated with these "nonmechanical" parts are ignored or compensated for, it can be argued that it is inconsistent to require safe flight capability after failure of a "nonmechanical" part and not after failure of a purely mechanical part. However, service experience (including that presented in Lockheed Report No. 19089) indicates that no part is immune from failure and it can be argued that the requirements for "nonmechanical" parts, while perhaps inconsistent with mechanical system requirements, are more appropriate.

e. Lockheed Report 19089 indicates that failure of the power actuator housing or jamming of the power actuator valve would be catastrophic as these parts are common to both hydraulic systems and both systems would be lost. Although fixed wing aircraft power control requirements are essentially the same as rotorcraft, an independent trim system is required which may permit continued safe flight and landing after such a failure. No such backup control system is required on rotorcraft or provided on the CL-286.

5. CONCLUSIONS

a. CAR 6.328 requires the capability of continued safe flight and landing after any single failure in the power portion of the control system (including cracking of actuator housings, and jamming of control valves) except the failure of mechanical parts and (the specific case) of power cylinder jamming need not be considered if shown to be extremely remote. Mechanical parts are considered to be only those parts (such as piston rods and links) which have essentially a sole primary function of directly transmitting a control force or motion. Parts such as hydraulic actuator housings which have other primary functions, such as containing hydraulic fluid, are not considered mechanical parts.

b. Based on the evidence presented, the CL-286 control system (described in Lockheed Report 19089 dated January 3, 1966) does not provide capability of continued safe flight and landing after failure of an actuator housing or jamming of an actuator valve and does not comply with CAR 6.328.

REVIEW CASE NO. 62 LABELING OF FIRE PANEL - BOEING 707-321C
(Issued 11 September 1967)

1. INTRODUCTION

a. Boeing requested approval for relocating of the fire control panel in the Model 707-321C airplane. Boeing advised that they proposed no change in the labeling of the fire panel. The change merely involved relocation of the existing fire control panel.

b. Western Region in their evaluation of the project for relocating the fire panel found that the control system marking did not comply with CAR 4b.737(c)(2).

c. Boeing requested a review case to determine if the general requirement of CAR 4b.737 is applicable to emergency controls in lieu of the specific requirement of CAR 4b.737(c)(2).

2. CHRONOLOGICAL HISTORY

a. Western Region letter dated April 6, 1967, advised Boeing that a review of the pilot's overhead control panel design for the relocation of the fire handles in the Model 707-321C Pan American Airlines (PAA) airplane did not comply with CAR 4b.737(c)(2). This rule requires emergency controls to be colored red and marked to indicate their function and method of operation. The specific areas of concern are as follows:

(1) The fire panel is not labeled as a fire panel.

(2) The fuel shutoff handles do not indicate their function and method of operation, such as, Fuel and Hydraulic Shutoff - Pull.

(3) The bottle discharge switches are not labeled.

b. Boeing letter dated April 17, 1967, submitted additional data stating the reasons why relabeling of fire switch panel is unnecessary are as follows:

(1) Boeing advised that the only change in the fire switch panel was the relocation of a previously approved panel.

(2) Training of pilots is both extensive and comprehensive and past service experience also has indicated no problems due to lack of labeling.

(3) Excessive labeling can cause confusion.

4. BACKGROUND.

a. During type certification of the Piper Model PA-35 aircraft in the Southern Region, information was requested from the Eastern Region as to the landing gear warning system incorporated on the PA-25, 30, and 31 airplanes. SO-210 was advised that the system used on both the PA-23 and PA-30 involved a blinking red light which functions when one throttle is retarded and a horn which sounds when both throttles are retarded. This system was approved in accordance with CAR dated 1956. The system incorporated on the PA-31 on the other hand was approved under CAR 3.359 Amendment 3-7, effective May 3, 1962. This amendment required an aural or equally effective warning device be provided which functions continuously when one or more throttles were closed.

b. Subsequent to discussions with the Southern Region, the Eastern Region observed that the PA-31 maintenance manual stated that the "warning horn should not operate when only one throttle is retarded." On the basis of this finding, Piper was requested to submit comments and a proposal for correcting this noncompliance. Piper, in their response, contended that the system was consistent with that installed on the PA-23 and 30. They indicated that this system met the intent of the rule and was preferred since single engine operation could be practiced without being annoyed by a loud horn and yet have adequate warning to prevent a wheels-up landing. Piper also maintains that the warning light is equally effective as an aural device and is actually safer. The region disagreed and pointed out that the blinking light was not equally effective as an aural device. The region also indicated that the lights covered in a note following CAR 3.359 Amendment 3-7 refers to the landing gear position indicator system and not to the warning device as assumed by Piper.

5. ANALYSIS.

a. The provisions of CAR 3.359 Amendment 3-7 specifically require that an aural or equally effective warning device function continuously when one or more throttles are closed. The blinking red light used in the PA-31 installation which functions when one throttle is retarded cannot be considered equally effective to an aural device. A light, to be effective, must be physically observed by the pilot. An aural device, on the other hand, provides an audible sound which can be perceived by hearing and does not demand direct observation on the part of the pilot. The light used on the PA-31 requires the attention of the crew within the cockpit and is therefore not considered as effective as an aural device.

REVIEW CASE NO. 63 APPLICATION OF CAR 4b.356(e) TO THE AERO COMMANDER MODEL 1121 (Issued 14 November 1967)

1. INTRODUCTION

Aero Commander has installed a baggage compartment in the Model 1121 airplane in the unpressurized area aft of the passenger cabin. The cargo compartment is being approved under a "D" classification. Access to the compartment is through an exterior door hinged at the top with four quick-acting fasteners and a simple rotary tab and a slot-type lock. At present, Aero Commander proposes to provide a visual means to signal the pilot that the door is closed and fully locked on the rotary tab lock only. The Southwest Region has requested that Aero Commander either (a) show by flight that the compartment can be classified as "D" with only this lock closed, or (b) install a signaling device that will indicate to the pilot (or copilot) when all fasteners are in the locked position, since this is the configuration tested for "D" compartment, or (c) that the compartment be tested with any given number of locks all of which would give a signal to the pilot and copilot when locked.

2. REFERENCE REGULATIONS

CAR 4b.356(e) - "In addition, visual means shall be provided to signal to appropriate crewmembers that all normally used external doors are closed and in the fully locked position."

3. CHRONOLOGICAL HISTORY

a. April 14, 1966 - During a compliance inspection of a prototype installation of the cargo compartment, Aero Commander was advised that the single switch on the lock did not comply with the regulation. Aero Commander agreed to review the installation to determine a method of compliance.

b. February 3, 1967 - At the time of a familiarization with the Models 1121A and 1121B airplanes, the alternatives given in the introduction were formally established by the Southwest Region.

c. April 28, 1967 - Aero Commander requested a case review based on the arguments presented in their letter 603/G/1121/3161 of the same date.

d. May 19, 1967 - Southwest Region forwarded Aero Commander's request.

4. BACKGROUND

Aero Commander's Position

Aero Commander indicates that the door in question was used for access prior to installation of the baggage compartment, and was normally opened during each preflight inspection. Aero Commander contends that the latches are readily visible during preflight and that the micro-switch on the security lock extinguishes the cockpit warning light, thus indicating to the crew that the door is closed and fully locked as required by CAR 4b.356(e). This configuration was certificated when used only as an access door. With installation of the baggage compartment, straps are provided across the door opening within the compartment to secure the baggage. They believe that no change exists in the certification basis for the aft fuselage access door whether or not a baggage compartment is installed and contend that the proposal configuration complies with CAR 4b.356(e).

Southwest Region's Position

The Southwest Region believes the intent of the regulation is to assure that the normally used external doors must be provided with a signaling device that insures the pilot or copilot (the only crewmember in this case) will be advised of the degree to which a door is secured. Although in the case of initially outward opening doors, inspection means are provided for crewmembers for determining if the locks are completely secured, they do not believe a walkaround inspection is comparable or necessarily a guarantee that the door has not been opened and closed after crew inspection. For example, the crew may make their inspection and start ground checkoff. Meanwhile, the door is opened to load late baggage but due to the difficulty of closing the fasteners, they may be only partially closed or not at all. Observation of tabs at night is not easy and their position may not be noticed.

When the compartment door was used for access to equipment only no signaling device was required. With installation of a class "D" baggage compartment this door now becomes a normally used external cargo door, and its fastening should comply with the regulations for normally external used doors. Ventilation in the compartment should not exceed class "D" limits when the signaling device indicates that the door is fully locked.

From the above, Southwest Region recommends that Aero Commander Bethany Division be required to proceed with one of the alternatives suggested in the "Introduction" or one equally satisfactory.

5. ANALYSIS

Prior to installation of a baggage compartment, the access door in question was not considered a "normally used external door" and was not evaluated under the requirements of CAR 4b.356(e). With installation of a class "D" baggage compartment this door is a cargo door and is obviously "a normally used external door" and consequently is now subject to the requirements of CAR 4b.356(e). Further, while this compartment is unpressurized, the regulatory history of CAR 4b.356(e) shows that is was intended to apply to both pressurized and unpressurized doors.

CAR 4b.356(e) requires that, "In addition, visual means shall be provided to signal appropriate crewmembers that all normally used external doors are closed and in the fully locked position." This requirement is further amplified by CAM 4b.356-6, which states in part, "The objective herein is to be able to ascertain by visual means that the door and/or locking means is sufficiently engaged to eliminate hazards emanating from an improperly closed door."

With the existing baggage compartment configuration, it is possible to secure the baggage compartment door and extinguish the cockpit door warning light by engaging only the door rotary security lock, while leaving the four overcenter latches unfastened. The words "closed and in the fully locked position" in CAR 4b.356(e) must be read as describing the door condition that provides the basis for approving the type design. In the case of Model 1121 with the class "D" baggage compartment, this means not only that the door must remain closed under all flight conditions but also that all class "D" baggage compartment requirements are met. Aero Commander has not shown that engagement of only the rotary security lock will assure that the door is in this condition. Aero Commander must show that the warning light is not extinguished unless the door is in this condition with respect to remaining closed and maintaining class "D" baggage compartment integrity.

6. CONCLUSION

Aero Commander has not shown that the Model 1121 complies with CAR 4b.356(e).

advised Boeing that if the APU is to be operated inflight, the APU is considered an essential unit. As an essential unit the APU must meet all applicable powerplant installation standards and the APU fuel system should be protected against formation of the crystals in the fuel. Assuming there are no alleviating factors in the APU fuel systems, provisions for maintaining fuel flow under fuel icing conditions should be as effective as for the main engines.

c. Boeing letter, September 22, 1967, (reference 2.d.) reasoned that the APU-generator need not have a higher degree of reliability than that of the main engine generator. We concur with the Boeing position; however, if the minimum equipment list (MEL) would permit dispatch of the aircraft with one generator inoperative, the APU driving the generator replacing it would have to meet the same criteria as the main engine and the APU fuel systemwould have to meet the same degree of reliability on the engine fuel system.

d. The reference by Boeing (reference 2.d) to military service is not applicable since military aircraft are generally serviced with fuel containing a controlled amount of anti-icing additive. This could explain the reason that no icing of the APU fuel filter occurred in military service.

e. The reference by Boeing (reference 2.d.) to service experience on the Boeing 727 is not applicable since the APU on the Boeing 727 is not operated inflight and is not an essential unit.

f. Concerning the statement made by Boeing (reference 2.d.) that the BAC 1-11 was certificated for inflight use without a fuel heater, we find that compliance with CAR 4b.435, amendment 4b-11 was determined by tests and application of operating limitations as specified in the aircraft flight manual. Under these limitations, the filter was not susceptible to icing.

4. CONCLUSIONS

a. The Western Region has correctly applied the requirements of FAR 25.1309.

b. It is concluded that if the MEL permits dispatch of the aircraft with an engine generator inoperative provided the APU generator is operating, the APU fuel system must meet all those powerplant installation standards that are necessary to insure that the APU, as installed, has a level of reliability equivalent to that of the engines.

REVIEW CASE NO. 65 PIPER PA-31 AIRCRAFT/CAR 3.359 AMENDMENT 3-7
(Issued 22 March 1968)

1. INTRODUCTION.

a. The Piper Aircraft Corporation contends that the landing gear warning device incorporated on the Model PA-31 aircraft meets the intent of CAR 3.359 Amendment 3-7. This system, which is similar to that previously approved on the PA-23 and PA-30, consists of a blinking red light that operates when one throttle is retarded and a horn that sounds when both throttles are retarded.

b. The Eastern Region maintains that the PA-31 warning system does not comply with CAR 3.359 Amendment 3-7 because (1) the landing gear warning horn (aural device) does not sound when only one throttle is retarded, and (2) the blinking red light is not considered to be an equally effective warning device.

2. REFERENCE REGULATIONS. CAR 3.359, Amendment 3-7 - "Position Indicator and Warning Device. When retractable landing wheels are used means shall be provided for indicating to the pilot when the wheels are secured in the extreme positions. In addition, landplanes shall be provided with an aural or equally effective warning device which shall function continuously when one or more throttles are closed until the gear is down and locked." This amendment was made applicable to the PA-31 by Exemption No. 460 dated December 9, 1965.

3. CHRONOLOGICAL HISTORY.

a. September 18, 1967, EA-212 letter to Piper Aircraft Corporation informing them that their Model PA-31 was not in compliance with CAR 3.359 Amendment 3-7.

b. September 21, 1967, Piper letter to EA-212 contending that their Model PA-31 did comply with CAR 3.359.

c. October 10, 1967, EA-212 letter to Piper disagreeing with Piper's claim and requested compliance.

d. November 22, 1967, Piper letter to EA-212 maintaining that the PA-31 meets the intent of CAR 3.359.

e. December 7, 1967, EA-200 memorandum to FS-100 requesting a review case.

4. BACKGROUND.

a. During type certification of the Piper Model PA-35 aircraft in the Southern Region, information was requested from the Eastern Region as to the landing gear warning system incorporated on the PA-25, 30, and 31 airplanes. SO-210 was advised that the system used on both the PA-23 and PA-30 involved a blinking red light which functions when one throttle is retarded and a horn which sounds when both throttles are retarded. This system was approved in accordance with CAR dated 1956. The system incorporated on the PA-31 on the other hand was approved under CAR 3.359 Amendment 3-7, effective May 3, 1962. This amendment required an aural or equally effective warning device be provided which functions continuously when one or more throttles were closed.

b. Subsequent to discussions with the Southern Region, the Eastern Region observed that the PA-31 maintenance manual stated that the "warning horn should not operate when only one throttle is retarded." On the basis of this finding, Piper was requested to submit comments and a proposal for correcting this noncompliance. Piper, in their response, contended that the system was consistent with that installed on the PA-23 and 30. They indicated that this system met the intent of the rule and was preferred since single engine operation could be practiced without being annoyed by a loud horn and yet have adequate warning to prevent a wheels-up landing. Piper also maintains that the warning light is equally effective as an aural device and is actually safer. The region disagreed and pointed out that the blinking light was not equally effective as an aural device. The region also indicated that the lights covered in a note following CAR 3.359 Amendment 3-7 refers to the landing gear position indicator system and not to the warning device as assumed by Piper.

5. ANALYSIS.

a. The provisions of CAR 3.359 Amendment 3-7 specifically require that an aural or equally effective warning device function continuously when one or more throttles are closed. The blinking red light used in the PA-31 installation which functions when one throttle is retarded cannot be considered equally effective to an aural device. A light, to be effective, must be physically observed by the pilot. An aural device, on the other hand, provides an audible sound which can be perceived by hearing and does not demand direct observation on the part of the pilot. The light used on the PA-31 requires the attention of the crew within the cockpit and is therefore not considered as effective as an aural device.

b. The note following CAR 3.359 Amendment 3-7 refers to the use of lights for indicating the position of the landing gear. This is evident by the use of the phrase "all lights out" to advise the pilot when the wheels are up and locked. The only portion of this note which does refer to the warning device is in the last sentence which states that a throttle stop cannot be considered an acceptable alternative for the aural warning device.

c. The horn system incorporated on the Piper PA-31 which sounds when both throttles are closed would be acceptable to this extent only. To comply with CAR 3.359 Amendment 3-7, however, the horn should also function when one throttle is closed.

c. Piper's contention that their system is a safer system than that required to comply with the regulations is incorrect. For example, after the loss of an engine, the only warning available to the pilot would be a blinking light on the instrument panel in cases where the power setting on the remaining engine would not be reduced to the extent necessary to activate the aural device. This type of indication, in relation to the other demands required of the pilot during final approach for landing with an engine out, would not be as effective as an aural device. In addition, during approach the pilots' primary attention is normally directed along the flight path rather than on the instrument panel.

6. CONCLUSION. We concur with the Eastern Region's findings that the aural warning device installed on the Piper PA-31 does not comply with CAR 3.359 Amendment 3-7. The aural device used on this aircraft to be acceptable must function when one or more throttles are closed.

REVIEW CASE NO. 66 INTERPRETATION OF FAR 25.857(e)(1) FOR THE BOEING 747 AIRCRAFT (Issued 9 April 1968)

1. INTRODUCTION

   a. The Western Region requested a review case to determine if the "fire-resistant lining" required in FAR 25.857(e)(1) may serve both as primary structure and cargo compartment lining.

   b. WE-100 concludes that a separate fire-resistant lining is required.

2. CHRONOLOGICAL HISTORY. WE-100 memorandum to FS-100 dated October 13, 1967, transmitting the region's review and recommendations and transmitting the Boeing Company proposal to the Western Region, dated August 15, 1967.

3. REGULATION AND PROBLEM.

   a. The regulation in question, FAR 25.857(e)(1), states that "a Class E cargo compartment is one on airplanes used for the carriage of cargo and in which ... there is a fire-resistant lining."

   b. The Western Region contends that the cargo compartment fire-resistant lining must be separate from primary structure. The Boeing Company contends that the primary structure may also serve as the fire-resistant lining.

4. BACKGROUND MATERIAL.

   a. Fire-resistant lining is required for Class B, C, D, or E compartments. Fire-resistant material, in the past, has been used to line cargo compartments. This lining has completely enclosed the compartment including floor, ceiling, sides, and ends. The exception to this has been floors of compartments in the normal passenger area when used as a Class B, C, D, or E compartment. The upper floor surface has been accepted as both fire-resistant liner and primary structure.

   b. The presence of fire-resistant lining offers primary structure a degree of protection from the effects of heat and damage from baggage or cargo.

   c. Reference the Annual Review, Airworthiness Civil Air Regulations Meeting - August 1950, Appendix IV, "CAA's Comments and Proposals on Part 4b of CAR." On page 39, 4b.383(d)(3), it is stated that "The compartment shall be completely lined with at least fire-resistant material and sufficiently insulated to protect surrounding structure and equipment from unsafe temperatures." This proposal, from which the current FAR 25.857(e)(1) evolves, clearly states the intent that the surrounding structure and equipment be protected from unsafe temperature.

5. ANALYSIS.

a. The Boeing Company believes that a separate fire-resistant lining serves the following purposes:

(1) To help support and confine baggage.

(2) To protect the basic airframe structure from rough treatment from baggage handling and from impact by baggage.

(3) To insure that the lining itself will not ignite, help spread, and thereby add to the severity of the fire.

(4) To protect other cumbustible materials on the outside of the liner.

b. The Boeing Company offers that all cargo will be either completely containerized and/or palletized and covered by nets and that the presence of any separate fire-resistant lining material is required only for the purpose of achieving proper temperature environment in the aircraft.

c. The Western Region concurs with the Boeing Company that a lining of fire-resistant material essentially serves two purposes; namely, protection of structure from physical damage and fire. Further, they agree that a fire-resistant container and pallet combination would be satisfactory to demonstrate compliance with the regulation.

d. Boeing concludes that because a Class E cargo compartment will have a smoke detection system and means for cutting off ventilation airflow, it is reasonable to assume that fires which may occur will be confined to those of relatively small magnitude, thus negating the need for a separate lining. The Boeing Company offers this opinion but has not attempted to substantiate it with either tests or data.

e. Agency research at the NAFEC facility relative to cargo compartment fire protection indicates that the closing off of ventilation sources is ineffective, of itself, in the control of the magnitude of a compartment fire. Therefore, sole dependence on ventilation control without a fire-resistant lining is inadequate.

f. As noted in Item 4.a., the upper surface of the floor in the normal passenger area, when used as a Class B, C, D, or E compartment, has been accepted as both fire-resistant lining and primary structure.

(1) With reference to Item 4.c., the present rule is intended to reflect the objective set forth in that Annual Review.

(2) A separate lining has not been required on the upper floor surface because a fire originating in the upper compartment will first affect the ceiling and side walls due to the higher temperature gradient near the ceiling. Since the integrity of the upper fuselage skin and support structure is affected first, the fire's affect on the floor becomes a consideration of secondary importance.

(3) In the remote case whereby the fire does burn through a portion of the floor prior to loss of ceiling or side wall integrity, the separate fire-resistant lining below the floor serves as the barrier against further extension of the flame front.

6. CONCLUSIONS. In consideration of the foregoing, wherein FAR 25.857(e)(1) includes a requirement for a fire-resistant lining, it is concluded:

a. That a fire-resistant containerized system is a separate fire-resistant lining which satisfies the requirement.

b. That on the basis of equivalent safety, the floor of the B-747 compartment in the normal passenger area, when used as a cargo compartment, satisfies the requirement for a fire-resistant lining while in no case may the underside of this flooring be used as the lining for below the floor cargo or baggage compartment.

c. That except as noted in a and b above, each cargo compartment classified as a Class E compartment must have a fire-resistant lining which lines the primary structure.

d. That a through c above applies, as well, to each cargo or baggage compartment classified as B, C, or D in FARs 25.857(b)(4), 25.857(c)(5), and 25.857(d)(4).

e. That a cargo compartment which depends upon a fire-resistant containerized system to comply with these requirements must be properly identified through incorporation into the type design with reference to appropriate limitations and/or other information.

REVIEW CASE NO. 67 FIRE RESISTANT REQUIREMENTS FOR THE OIL COOLER LOCATED IN A FIRE ZONE OF THE WINDECKER MODEL AC-7 AIRPLANE (Issued 3 January 1969)

1. INTRODUCTION

Windecker Research,Inc., Midland, Texas, is undertaking a type certificate program for their Model AC-7 airplane. During the preliminary type board meeting held on January 16 and 17, 1968, Windecker was advised, in effect, that the fire resistant qualities of the oil cooler would have to be considered. It was pointed out, however, that if the engine installed in this aircraft had the oil cooler located on the front of the engine, the danger of damage to this component from engine compartment fires would be minimized. Recently, it was discovered that the engine being provided had the oil cooler mounted in a cutout in the rear vertical baffle; therefore, Windecker was told that it would need to be shown that the cooler is fire resistant to the extent that it can withstand a flame temperature of 2000° F. for five minutes.Windecker did not agree with this and by their letter of September 3, 1968, requested that a review case be initiated.

2. REGULATIONS AFFECTED

a.1.1, General Definitions: "Fire resistant," as the term applies to fluid-carrying lines, other flammable fluid system parts, wiring, air ducts, fittings, and powerplant controls, means the capacity to perform the intended functions under the heat and other conditions likely to occur at the place concerned.

b.21.21, Issue of Type Certificate: Normal, utility, acrobatic, and transport category aircraft; aircraft engines; propellers. This rule states, in part, that an applicant is entitled to a type certificate if the product meets the applicable airworthiness requirements of the Federal Aviation Regulations and any special conditions prescribed by the Administrator and the Administrator finds, for an aircraft, that no feature or characteristic makes it unsafe for the category in which certification is requested.

3. HISTORY

a. During the Preliminary Type Certification Board Meeting held on January 16-17, 1968, there was a discussion regarding the fire resistant qualities of the oil radiator. At that time, it was believed that the engine being furnished for this aircraft would have the oil radiator at the front end. Agreement was reached that, if this was the case, the hazard from a fire in the engine compartment would be minimized.

b. On August 7, 1968, representatives from SW-214 visited Windecker to witness fuel tank tests. During the visit, other aspects of the powerplant installation were discussed. It was learned that the engine being furnished had the oil cooler located in a cutout in the rear vertical baffle. This location places the cooler such that flames from a fire in the powerplant compartment could directly impinge on the oil cooler core. The Southwest Region reiterated its position that, as located, the oil cooler would have to be fire resistant to the extent that it must withstand a flame temperature of 2000° F. for five minutes.

c. On August 21, 1968, a representative of SW-214 visited Windecker. The subject was again raised regarding the need for a fire resistant oil cooler. SW-214 reaffirmed its position. Windecker was advised of the review case procedure where there was a disagreement between the certificating region and the applicant in the interpretation or application of the rules.

d. Windecker Research, Inc., by their letter of September 3, 1968, expressed disagreement in the need for a fire resistant oil cooler and requested a review case if the Region persisted in its stand.

4. BACKGROUND

a. The following represents Windecker's position:

(1) The requirements for fire resistance of sheet or structural components as defined in FAR 1 is that they be equivalent to aluminum. The part in question (Continental Part No. 634063) is made of aluminum so equivalency is obvious.

(2) The oil cooler, being a fluid-carrying component, would have the ability to carry heat away from the source. Therefore, the use of aluminum in this application should be conservative.

(3) The cooler is part of a type certificated engine. As such, the applicant would, in effect, be "tampering" with the T.C. of another manufacturer by testing or evaluating that manufacturer's component. In addition, the cooler, per se, is manufactured by a third party, precluding any control by Windecker of design changes.

(4) The cooler as installed is similar to the installation in aircraft manufactured by others and, consequently, has a long service record. Because of this similarity, one installation should be considered as good as the other.

(5) There are no known cases where this oil cooler was the cause of a fire; therefore, the safety record, together with a good service record, should adequately substantiate the design.

b. The following represents the Southwest Region's position:

(1) FAR 1.1 defines "fire resistant" as the term applies to flammable fluid system parts as the capacity to perform the intended functions under the heat and other conditions likely to occur at the place concerned. Presently, there are many rules written on the assumption that fire is a likely occurrence. NPRM 68-18 published in the Federal Register on August 22, 1968, further indicates, by the revised definition of fire resistant, that fire should be considered as a likely occurrence.

(2) FAR 25.1183(a) requires that lines and fittings carrying flammable fluids must be fire resistant. If such lines and fittings must be fire resistant, it is only logical that other parts of the same system and located in the same area should be capable of withstanding fire at least as well as the lines. NPRM 68-18 proposes to amend FAR 25.1183 to speak to other flammable fluid system components and explains that this is the intent of the present rule.

(3) AC 20-15A dated March 24, 1966, par. 4.b, states in part that, under FAR 23.901(a), the powerplant installation of an aircraft includes each component (engine, propeller, and associated parts, appurtenances, and accessories) that satisfies certain stated conditions. It is further stated that each such component is subject to the powerplant installation requirements set forth in Subpart E of the applicable aircraft airworthiness part even when the component (as for example fuel pumps, lines and valves) is supplied by the engine (or propeller) manufacturer as an integral part of a type certificated engine (or propeller). The fact that a component is furnished to the applicant as part of a type certificated engine does not relieve the applicant from showing compliance with the airworthiness requirements for the aircraft.

(4) Notwithstanding the fact that there are similar installations in airplanes certificated under delegation option procedures,

it is the conviction of SW-214 that the absence of fire in no way substantiates the fire resistant capabilities of a component. A good service record indicates that the components are durable and in all probability will not contribute to the start of a fire. The concern, however, is their susceptibility to failure from an existing fire and their contribution to that fire subsequent to failure.

(5) While oil flow through the cooler may dissipate heat from a fire rapidly enough to prevent burnthrough, unless this is demonstrated by test, it is only speculation.

(6) It is considered that the provisions of FAR 21.21 that speak to an unsafe feature can be applied. The oil cooler is mounted on the left side of the rear vertical baffle in direct line with the cylinders. Cooling air over the cylinders would be directed through the cooler. The fuel manifold is located on top of the engine with one high pressure fuel line going to each cylinder. Should a fire result from the failure of one of the lines on the left side, it is quite likely that the flames from that fire would, as the result of normal cooling air flow, be directed at the core of the oil cooler. Failure of the core and subsequent dumping of oil would contribute significantly to the intensity of the fire.

5. DISCUSSION

a. FAR 1.1 contains a definition of the term "fire resistant." This definition is not a requirement by itself. There is no regulation in FAR 23 requiring the oil cooler to be fire resistant in this airplane.

b. FAR 25.1183(a) and NPRM 68-18 do not contain the certification requirements for approval of this airplane. The applicable regulation for this airplane is FAR 23, as amended.

c. The definition of the term "fire resistant" (FAR 1.1) is in two parts: One part defines the meaning of fire resistant with respect to sheet or structural members; the other part defines fire resistant with respect to flammable fluid-carrying lines, other flammable fluid system parts, etc. In the latter case, fire resistant is defined as the capacity to perform the intended functions under the heat and other conditions likely to occur at the place concerned. The oil cooler, being a flammable fluid component, would be subject to this latter definition of fire resistance (if the component was required to be fire resistant,) and not to the definition applicable to sheet or structural members.

The fact that the component is constructed of aluminum may or may not be important to the establishment of fire resistant capability by this latter definition. Experience indicates that an oil cooler would be likely to remain intact during fire conditions as long as oil continued to circulate through it. Oil circulation would continue as long as the engine continued to operate. In the event of a fire, however, shut down of the engine might be deemed desirable, in which case circulation would cease and the oil cooler might fail. The recommended procedure for combating fire in flight in an aircraft engine is to shut down the engine, and shut off the flammable fluids to the engine compartment. This procedure will normally reduce a fire to minimum proportions immediately. Consequently, the hazard of oil being released from the oil cooler when flame impingement causes a cooler failure would be greatly reduced.

d. While it is possible to theorize that the hazard of fire aggravation associated with an oil cooler failure is nominal whether or not the engine is in operation, this is not to say one oil radiator location would not be safer than another. The Regional Office recognized this, as the history indicates, when it agreed to accept the oil radiator without further substantiation if it were located at the front end of the engine.

e. The Regional Office has commented that they believe the cooler is located in a hazardous position within the power plant installation. They cite the possibility that the high pressure fuel manifold, located over the top of the engine, can fail such as to result in a fire, and then reason that this fire will be directed at the oil cooler core as a result of the normal airflow within the engine compartment. If the oil cooler is located in the main cooling airstream (and, accordingly, the main fire path), substantiation of its fire resistant capability would certainly be in order to prevent a hazard. This does not appear to be the case, however, in this installation as it has been described. While some air will be directed through the oil cooler core, the largest amount of air will flow between the engine cylinders to cool the cylinders. If a fire occurred within the power plant installation, the fire would, most likely, follow the airflow and be largely directed between the cylinders. Very little fire, therefore, would be expected to be directed to the oil cooler itself.

f. With respect to the problem cited by Windecker, that they would be "tampering" with the T.C. of another manufacturer by testing or evaluating the oil cooler, we cannot accept this logic. If this thought were to be accepted, then a flight test program which evaluated the performance and cooling of the engine in an airplane installation could also be considered as tampering. FAR 21.21(b)(2) clearly indicates that an applicant is entitled to a TC providing no unsafe feature or characteristic is present in the article to be type certificated. If the airplane incorporates an engine with a feature found to be unsafe, then the airplane cannot be type certificated. It should also be noted that a feature, if found to be unsafe in one airplane installation, may not necessarily be unsafe in another installation.

g. Windecker cites the long service record of safe operation of this oil cooler similarly installed in a great many other airplane models as reason to believe that their installation will be just as safe. This argument would be a good one if backed up with experience in demonstrating the integrity of the cooler during fire conditions. Lacking such experience, this argument can only be used to establish the durability and reliability of the cooler. The region makes this point in their discussion, stating that the service experience simply demonstrates that the cooler is durable and probably will not, therefore, fail and be the cause of a fire. It in no way demonstrates the capability of the cooler to remain unhazardous in the presence of an external fire. It appears, however, that such fires as may have occurred have not served to indicate that the installation configuration is unsafe.

h. The region cites the current NPRM 68-18, FAR 1.1 wording which intends to clarify that fire be considered likely in defining fire-resistant capability. Further, the region cites the requirements of 25.1183(a) and NPRM 68-18 which propose to amend FAR 25.1183(a), requiring that flammable fluid components, in addition to lines and fittings, be fire resistant. These are proposals and, as such, should not be applied as a current applicable regulation. In addition, the NPRM 68-18 proposal regarding lines, fittings, and flammable fluid system components applies to FAR 25.1183(a), and was not similarly proposed for FAR 23.1183(a) at the time. The certification requirements for the Windecker airplane are FAR 23.

6. <u>CONCLUSIONS</u>

Windecker Research, Inc., need not demonstrate the fire-resistant capability of the oil cooler since:

a. There is no requirement in the applicable FAR regulations.

b. The region has not substantiated the presence of an unsafe feature in the proposed installation under the requirements of FAR 21.21(b)(2).

REVIEW CASE NO. 68 CUTOUT SWITCH FOR ELEVATOR TRIM SYSTEMS ON TED SMITH AIRCRAFT COMPANY MODELS 600 AND 601 (Issued 18 August 1969)

1. INTRODUCTION

This review case results from the Ted Smith Aircraft Company, Inc. request to omit the additional electrical circuit protection, a cutout switch, for the trim tab control system in their Models 600 and 601 since they believe the system complies with pertinent regulations without such a cutout switch. WE-100 disagrees, contending that compliance is not shown and an unsafe condition would exist without the cutout switch.

2. REGULATIONS AFFECTED

a. FAR 21.21(b)(2), unsafe feature or characteristic.
b. FAR 23.143, pilot effort.
c. FAR 23.161, trim.
d. FAR 23.409, tabs.
e. FAR 23.677, trim systems.
f. FAR 23.1351, electrical systems.
g. CAR 3.337 and CAM 3.337-1, trimming controls.

3. HISTORY

a. December 31, 1968, a letter from Ted Smith Aircraft Company, Inc. to WE-100 enclosed a draft of a proposed review case to support their position.

b. January 13, 1969, TSA letter to WE-100 submitted an addition to their draft review case.

c. March 20, 1969, WE-100 letter to FS-100 submitted TSA draft review case, with addition, and their analysis of the TSA position with their reasons for a difference of opinion.

d. April 18, 1969, FS-100 letter to WE-100 requested clarification of WE-160 flight participation in the TSA demonstrations which were claimed to show compliance following various pitch trim runaways, as well as other significant factors to be considered.

e. May 14, 1969, WE-100 letter to FS-100 provided system operating characteristics and flight test data relevant to this case.

4. BACKGROUND

a. CAM 3.337-1 presents FAA interpretations which apply to CAR 3.337, including the statement: "Each trim control system will be reviewed on the basis of its individual merits."

b. FAR 21.21(b)(2) requires the Administrator to find, prior to type certification of an aircraft, "that no feature or characteristic makes it unsafe for the category in which certification is requested."

c. The "strength of pilots" criteria of FAR 23.143 includes the statement: "In no case may the limits exceed those prescribed in the following table." The table shows, for the pitch axis, 75 pounds for temporary application and 10 pounds for prolonged application.

d. FAR 23.161 presents normal trim criteria without regard to failures.

e. FAR 23.409 presents tab design criteria including "the most severe combination" which is applied to account for runaway trim situations.

f. FAR 23.677 (formerly CAR 3.337) presents the trim system criteria most directly related to this review case: "Proper precautions must be taken to prevent inadvertent, improper, or abrupt trim operation."

g. FAR 23.1351(b)(1)(i) requires each electrical system to be "free from hazards in itself, in its method of operation, and in its effects on other parts of the airplane."

h. TSA's draft review case contains the following "history": "Pitch trim runaways have occurred in the past on transport category aircraft even with dual circuit protection guarding against a double failure. Major damage has occurred to the aircraft involved and in some instances injury to passengers and loss of life. In these aircraft, a pitch trim runaway in either direction, without a means to quickly deactivate the circuit before reaching the mechanical tab stops, would place the aircraft beyond control to safely return and land."

i. TSA's draft review case contains the following "fact in the case": "When the trim tab system is installed in the aircraft, as proposed by TSA without the added circuit quick-deactivation switch located adjacent to the elevator trim switch, a dual electrical failure and a consequent trim tab runaway will produce full tab deflection and the pilot control forces will exceed those prescribed by FAR 23.143(c) for temporary and prolonged application.

j. WE-100 in their letter of March 20, 1969, to FS-100 added the following background comments:

(1) The airplane in question may become unsafe to operate following a runaway elevator trim, particularly in IFR conditions. A force exceeding 75 pounds would be required at some airspeeds with full nose-up trim within the operating envelope for the airplane. Even with power reduced to permit continued flight at a maximum force of 10 pounds, an excessive and unsafe workload is imposed on the pilot. One of the basic cues for safe IFR flight is pilot recognition of a trim change from any cause. A constant control pressure of 10 pounds would tend to mask any trim change.

(2) Approval of elevator trim systems for TSA Models 360, 400 and 600 was based on torque limited trim motors which stalled before excessive pilot effort is required to overpower the trim system. TSA Model 601 was found to have trim motors which were not torque limited and would not stall. As a result of this finding, a reexamination of the TSA trim systems was made which also revealed that the elevator trim circuit breaker is near the extreme right end of the instrument panel and cannot be manually tripped or pulled (tripped only by an electrical overload). Therefore, if a trim switch failure caused an elevator trim runaway, there would be no way to manually stop the runaway trim motor, which would continue to run until stopped either by the limit switch or by the mechanical stop.

(3) Flight tests were discontinued on TSA Model 601 when, following flight evaluation of the trim system and a later review, WE-100 required TSA to include an additional switch on the pedestal to disable the elevator trim in the event of a runaway. TSA added the switch to Model 601 which is currently approved WITH this switch. TSA is now proposing the removal of this switch from the Model 601 and they do not intend to add this switch to the Model 600 if they are successful with this review case.

(4) A single mechanical switch failure could cause a pitch trim failure and service experience shows this type of failure to be probable. Demonstrated continued safe flight and landing following such a failure are considered by WE-100 to be valid only for VFR cruise conditions at time of failure. A runaway during IFR or during approach and landing could be beyond the pilot's capability to cope safely with the resultant control forces and power setting change requirements.

k. WE-100 in their letter of May 14, 1969, provided the following additional information:

(1) Total elevator pitch trim travel is from $7^{\circ} \pm 1^{\circ}$ up to $37^{\circ} + 9^{\circ} - 1^{\circ}$ down. Time for full travel is 10 to 12 seconds. This yields a trim change rate of $4.3^{\circ} \pm .1^{\circ}$ per second; that is, for example at 130 m.p.h. IAS, a stick force change (dF/dt) of 10 pounds per second.

(2) The 10 pound continuous force limit per FAR 23.143 would be exceeded under the following conditions:

Aft c.g., full nose down trim, 120 m.p.h. IAS and greater.
Aft c.g., full nose up trim, 95 m.p.h. IAS and greater.
Fwd c.g., full nose down trim, all airspeeds.
Fwd c.g., full nose up trim, 118 m.p.h. IAS and greater.

(3) The 75 pound temporary force limit per FAR 23.143 would be exceeded under the following critical conditions at takeoff gross weight of 5719 pounds:

Aft c.g., full nose up trim, clean, 184 m.p.h. IAS and greater.
Aft c.g., full nose up trim, 25^{o} flaps gear up, 177m.p.h. IAS and greater.

5. DISCUSSION

a. The basic issue is whether or not the additional cutout switch for the elevator trim tab control system in Ted Smith Aircraft Company Model 600 and 601 airplanes is required for compliance with applicable regulations.

b. It has been shown that a runaway elevator trim, resulting in control force change at an abrupt rate, can result from a single failure, the occurrence of which must be considered probable.

c. It has been further shown that, without the additional cutout switch under discussion, such a runaway can continue until the tab reaches the extreme end of its travel in either direction.

d. TSA's draft review case admits that, in such a runaway, the pilot control forces will exceed those prescribed by FAR 23.143(c) for temporary and prolonged application. This is confirmed by regional correspondence and flight test data.

e. The position taken by WE-100 is that TSA Models 600 and 601 may become unsafe to operate without a separate electrical cutoff switch in the event of a runaway elevator trim, particularly under IFR conditions. It is further contended by WE-100 that a constant control pressure of 10 pounds would tend to mask any trim change and would render unsafe continued IFR flight following such a runaway. This is confirmed by flight test data.

6. CONCLUSIONS

a. Ted Smith Aircraft Company, Inc. Models 600 and 601 airplanes without a separate electrical cutoff switch in their elevator trim system do not comply with applicable Federal Aviation Regulations.

b. The TSA request to omit the additional electrical circuit protection for the trim tab control system should be denied.

REVIEW CASE NO. 69 INDUCTION SYSTEM ALTERNATE AIR DOOR REQUIREMENT FOR WINDECKER MODEL AC-7 AIRPLANE WITH FUEL INJECTION ENGINE (Issued 9 October 1969)

1. INTRODUCTION.

Windecker Research, Inc., Midland, Texas, is in the process of type certificating their Model AC-7 airplane. In January 1969, Southwest Regional Propulsion Section personnel visited Windecker to witness tests and to conduct a review of certain propulsion items. One observation made during the review was that the alternate air door of the engine induction system, when opened, permitted alternate air to be mixed with primary air. Regional personnel expressed the opinion that this arrangement was unacceptable. Windecker was so advised officially by letter dated 20 January 1969. In their letter of reply dated 29 March 1969, Windecker disagreed with the Regional interpretation of the pertinent rules and requested a review case.

2. REGULATIONS AFFECTED.

a. FAR 23.1091(b) requires that each engine must have at least two separate air intake sources except that an engine with a fuel injection pump need have only one air intake source if the air intake, opening, or passage is not obstructed by a screen, filter, or other part on which ice might form and restrict the airflow so as to adversely affect engine operation.

b. FAR 23.1093 requires that each engine air induction system must have means to prevent and eliminate icing.

3. HISTORY.

a. On 7 and 8 January 1969 representatives of the Propulsion Section, SW-214, visited Windecker Research, Inc., where they conducted a cursory examination of the powerplant in the Model AC-7 airplane. They later advised Windecker of possible problem areas including the alternate air source arrangement which permitted the mixing of primary air with the alternate air.

b. SW-214 letter of 20 January 1969 officially advised Windecker that the alternate air source would have to be arranged so that primary air would not mix with the alternate air.

c. Subsequently, the regional position was reaffirmed verbally on a number of occasions when representatives of Windecker visited the Regional Office.

d. On 1 April 1969, a Pre-Flight Type Board meeting was held at the Windecker facility. At the meeting, the regional representative again advised Windecker that the alternate air arrangement was unacceptable. Windecker chose to exercise their prerogative and request a review case. Pending resolution of the matter, Windecker agreed to a day-VFR restriction on the airplane.

e. On 3 April 1969, the Windecker letter dated 29 March 1969 requesting a review case was received in the Regional Office.

4. BACKGROUND.

a. Windecker views, as expressed in their letter of 29 March 1969, are as follow:

(1) The Continental IO-520C engine induction system is arranged so as to preclude icing. As proof of this, reference is made to a Continental Motors Corporation telegram dated 24 March 1969 to the Federal Aviation Administration Southwest Region stating in part: "Our fuel injection systems have been ice free with 33,000 units in service."

(2) FARs 23.1091(b) and 23.1093(d) seem to conflict with each other except that 23.1093(d) clearly states carburetor rather than a fuel injection system.

(3) The primary air source on the airplane is so located (sheltered) that the possibility of impact icing in the primary system is minimal.

(4) Thousands of aircraft in service today utilize the same powerplant as used in the AC-7 and have essentially the same induction system arrangement.

(5) The request to redesign the system so that alternate air is completely divorced from primary air is reasonable from the safety standpoint because of unsatisfactory service experience in other makes of aircraft; however, since there are other aircraft in operation using essentially the same system, it is unfair to require redesign in this case.

b. The Southwest Region's position is as follows:

(1) FAR 23.1091(b) requires an alternate air source if there is any obstruction in the induction system on which ice could form. The proposed Windecker induction system incorporates a filter in the inlet and a "butterfly" in the throttle housing, both of which are obstructions on which ice could form. An alternate air source therefore is required,

and Windecker has attempted to provide one. The objection to the design is that it permits cold moist primary air to mix with the warm alternate air, contrary to the intent of the rule.

(2) There is on record at least one instance of throttle valve icing in a fuel injection engine. Corrective action was required by AD 66-18-3. The engine was of different design and it appears that the throttle valve design may have been more susceptible to icing.

(3) While FAR 23.1093(d) does not specifically mention fuel injection engines, the rationale to be applied should be the same as for engines using carburetors which tend to prevent ice formation.

(4) Service experience has shown another problem may exist where mixing occurs. Ice forms on the downstream side of some air filters and, when warm alternate air is introduced, ice breaks free, enters the engine, and causes stoppage.

5. DISCUSSION.

a. Continental's statement that their fuel injection systems have been ice free with 33,000 units in service may be true but it neglects to make clear whether this record was achieved with the type of arrangement proposed by Windecker. Specifically, the statement gives no indication that it takes account of the type of alternate air source provided by Windecker.

b. Windecker believes there is a conflict between Section 23.1091(b) and Section 23.1093(d) except that the latter section uses the term "carburetor" rather than "fuel injector." FAR 23.1091 sets forth the requirements for the air induction system of each engine. Subsection (b) of this rule is applicable in this instance. FAR 23.1093 sets forth the requirements to prevent and eliminate icing in induction systems. None of the subsections in this rule are considered directly applicable to the Windecker Model AC-7 installation.

c. The region has indicated a concern with respect to the design of the Model AC-7 engine air induction system since 1) they believe ice can form on induction components of both primary air and alternate air systems and 2) service experience available to them has shown that ice did form on the induction system components of an airplane having an engine fuel injection system similar to the Model AC-7 system necessitating that mandatory action be taken to correct the deficiency on in-service airplanes.

d. Windecker states that it would be unfair to require modification of the AC-7 engine air induction system since other makes of airplanes use essentially the same design configuration. We have informed our other Regional Offices of the unsatisfactory service experience and pitfalls of injector air induction systems wherein cold moist primary air which cannot be shut off can reduce the effectiveness of alternate air systems. Experience has shown that the other injector air system designs in service do not incorporate this design deficiency.

e. In accordance with Section 23.1093, each engine air induction system must have a means to prevent and eliminate icing. The remaining subsections of this paragraph outline acceptable means of achieving this for certain specific carburetor types. The type of carburetion which the Windecker AC-7 airplane model employs is not specified in these subsections. Therefore, it is necessary that Windecker make a showing that the AC-7 has an induction system which does "prevent and eliminate icing." In this regard, the region must assure compliance with the rule by reviewing whatever design and test data is presented to assure that the provisions of the rule are met since no alternative heat rise requirements are included for the particular induction system design proposed. It would be logical for the region to consider, in the evaluation of this design, the possibility of ice to form in the induction system passages and the experience of other airplane designs wherein ice clogging was found and corrections were necessary. Windecker has offered no proof of compliance with the rule, contending only that other makes of aircraft use essentially the same configuration as proposed for the AC-7. While it is of interest that other makes of aircraft use similar induction air systems, this statement does not assure that compliance with this rule has been achieved. Compliance with 23.1093 is required for this engine air induction system design and a finding of compliance would require that information and data be presented by the manufacturer for the particular design.

f. FAR 23.1091 includes the general design requirements for induction system installations. FAR 23.1093 covers the specific requirements for induction system ice protection. It has been noted that these two rules have been misinterpreted since 23.1091 includes a specific statement pertaining to air induction systems employing a fuel injection pump and 23.1093 does not include a similar specific rule covering icing requirements for engine air induction systems having fuel injector pumps. Recent amendment to Part 23, Amendment No. 23-7, has corrected paragraph 23.1091 by deleting the specific reference to injector pump systems, thus the general design requirements for all engine air induction systems will be the same.

g. To further improve the wording of FAR 23.1093 since sufficient experience is now available upon which to base a specific requirement for induction system heat rise of engines incorporating injector fuel systems, it is planned that specific requirements for heat rise be recommended in this section.

6. CONCLUSIONS.

1. The Windecker Research, Inc., Model AC-7 engine air induction system has not been shown to comply with the requirements of FAR 23.1093.

2. FAR 23.1091 as revised by Amendment 23-7 clarifies the requirements pertaining to injector pump systems.

3. Since sufficient experience is now available, a change to paragraph 23.1093 can be proposed to include more specific requirements for heat rise of engine air induction systems employing fuel injectors. This would assist in assuring uniformity in the application of this rule by regional and DOA personnel.

GPO 904.247

REVIEW CASE NO. 70 HUGHES TOOL COMPANY, AIRCRAFT DIVISION, REQUEST FOR FLIGHT TEST PROCEDURES FOR DEMONSTRATING THE ABSENCE OF FLUTTER AND EXCESSIVE VIBRATION FOR THE HUGHES, MODEL 369 HELICOPTER

1. INTRODUCTION

Hughes Tool Company (Aircraft Division) using Delegation Option Authority provided the Western Region with a statement of compliance and was issued a type certificate for the Hughes Model 369H helicopter. During a subsequent audit of the certification procedures used by Hughes, Western Region pilots found the Model 369H in noncompliance with CAR 6.140 and 6.711. Hughes disputes this finding. The essence of the dispute is not over the requirements of the regulations but rather in the means of determining compliance. The point of contention is that the Hughes Company accepted a bank angle of less than 30^{o} at V_{NE} while the FAA pilot contends that if the helicopter is not capable of maneuvering to 30^{o} bank angles at V_{NE} without encountering excessive vibration the V_{NE} must be lowered until this maneuver can be performed.

2. REGULATIONS AFFECTED

a. CAR 6.140 (FAR 27.251)
b. CAR 6.711 (FAR 27.1505)

3. HISTORY

a. Originally, tests to determine compliance with CAR 6.140 consisted of steady unaccelerated flight in smooth air to V_D and gentle but undefined maneuvers at V_{NE}. These checks were made throughout the altitude envelope. Helicopters that were tested in this manner were characterized by the fact that their V_{NE} was usually well above V_H. The maximum demonstrated speed was normally a design or controllability limit rather than a roughness limit.

b. With the introduction of turbine engines, helicopter speeds were increased to the point that V_{NE} could be achieved in level flight and in some cases, a climb. Roughness limits were encounterd prior to reaching design or controllability limits during tests on some helicopters. As a consequence these helicopters were being consistently maneuvered at speeds much closer to a roughness limit.

c. In 1965, in recognition of the need to provide the pilot with some maneuver capability at V_{NE} and further to provide the pilot some margin away from roughness when operating in turbulence (since tests are necessarily done in smooth air) it was agreed among the regions that future tests would require 30^{o} banked turns at V_{NE} without encountering excessive roughness.

d. While the 30^{o} banked turn has been used consistently since that time, there was no consistent power setting used. Recently, policy was issued which specified the use of maximum continuous power at V_{NE} in conjunction with the 30^{o} banked turn.

4. FACTS IN THE CASE

a. CAR 6.120(b) requires that a helicopter possess flight characteristics such that it is possible to maintain a flight condition without requiring an exceptional degree of pilot skill, alertness, or strength under all conditions of operation probable for the type.

b. CAR 6.120(a) requires that such determinations be made for all speeds, power, and rotor rpm conditions for which certification is sought.

c. CAR 6.121 states that the rotorcraft must be SAFELY controllable and maneuverable during any maneuver appropriate to the type.

d. CAR 6.140 requires that the rotorcraft be free from flutter and excessive vibration under all speed and power conditions appropriate to the operation of the type.

e. CAR 6.711(a) requires that V_{NE} be not greater than 0.9 times the design maximum forward speed or 0.9 times the speed demonstrated in accordance with 6.140, whichever is less.

f. The procedures used in demonstrating compliance, including the 30^{o} banked turn at V_{NE} were shown to the Hughes pilots and used by them under FAA pilot supervision. The procedure was used in certification of the Model 369A, the Lockheed Model 286, and the Fairchild Hiller, FH-1100 in 1966.

g. On the basis of satisfactory experience demonstrated during the 369A certification tests, Western Region approved Hughes' application for Delegation Option Authorization on 28 January 1966. WE-160, on the basis of satisfactory experience with the pilots proposed by Hughes for authorization to conduct qualitative flight evaluations, concurred in the list designating pilots to make such findings.

h. Hughes, by letter to WE-100 dated 21 May 1969, provided the statement of compliance required by FAR 21.253(a)(3), and was issued a type certificate for the Model 369H.

i. During the audit conducted in July and August 1969, Western Region flight test personnel discovered excessive roughness in 30^{o} bank turns at V_{NE} at 6000 feet density altitude. Subsequent investigation disclosed that Hughes pilots were aware of the helicopter's inability to meet the 30^{o} banked turn criteria since they established V_R in straight flight and V_{NE} at .9 of V_R and then maneuvered the helicopter to some bank angle (less than 30^{o}) at V_{NE}.

5. DISCUSSION

Hughes, in their petition for this review case, makes several statements to which we take exception:

a. We disagree that excessive roughness in a helicopter with a fully articulated rotor is not related to safety of flight. Other helicopters with fully articulated rotors have experienced rapid pitch-up and roll left maneuvers which are further aggravated by application of right cyclic, when retreating blade stall was encountered.

b. We disagree that excessive roughness due to retreating tip stall presents no controllability or stability problems because the blade moment change does cause changes in control forces (cyclic and collective) as well as increased oscillatory feedback.

c. We disagree that the 30° bank test procedure is "recent", in that all pilots authorized to make qualitative findings were familiar with the procedure and standards prior to the issuance of the Delegation Option Authorization; if they were not, they could not have qualified for the DOA. Although we consider the 30° bank continuous turn a test procedure and, therefore, one acceptable method of determining compliance with CAR 6.140, we do not agree that either increased roughness or decreased bank angles constitute acceptable methods of determining compliance.

d. Regarding Hughes contention of noncompliance and interregional differences in the application of the procedures in question, we have checked with each region involved in helicopter certification and each has used the 30° banked turn criteria at V_{NE} since its inception in 1965.

6. CONCLUSIONS

In consideration of the foregoing it is concluded that:

a. The applicant must comply with the requirements of CAR 6.140 and 6.711(a).

b. The demonstration of compliance should be in accordance with the latest policy and procedures (i.e., at maximum continuous power (MCP) at V_{NE} and with a 30° bank angle) as described to the applicant by the Western Region.