NATIONAL TRANSPORTATION SAFETY BOARD

AIRCRAFT ACCIDENT REPORT

APACHE AIRLINES, INC.
DeHAVILLAND DH-104-7AXC, N4922V
COOLIDGE, ARIZONA
MAY 6, 1971

NATIONAL TRANSPORTATION SAFETY BOARD
Washington, D. C. 20591
REPORT NUMBER: NTSB-AAR-72-19

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ADOPTED: JUNE 1,1972

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16. Abstract

Apache Airlines, Inc., DeHavilland Model 104-7AXC, N4922V, crashed near Coolidge, Arizona, on May 6, 1971, at approximately 1315 m.s.t., while en route from Tucson to Phoenix, Arizona.

The aircraft was observed to enter a shallow dive which steepened as the aircraft approached the ground.

The right wing separated from the fuselage just prior to ground impact. The aircraft disintegrated upon impact and all 10 passengers and the two crewmembers were fatally injured.

The National Transportation Safety Board determines that the probable cause of this accident was the inflight failure and subsequent separation of the right wing. This failure was the result of a fatigue fracture in the lower main root joint fitting which propagated from an area of corrosion and fretting damage which, in turn, was probably caused by design deficiencies. These deficiencies remained undetected because surveillance of the supplemental type certification process and the modification programs was not adequate to assure compliance with design and inspection requirements.

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Adopted: June 1, 1972

APACHE AIRLINES, INC.
DEHAVILLAND DH-104-7AXC, N4922V
COOLIDGE. ARIZONA
MAY 6, 1971

SYNOPSIS

At about 1315 mountain standard time on May 6, 1971, an Apache Airlines, Inc., DeHavilland Model 104-7AXC, N4922V, operating as a scheduled air taxi flight, crashed about 5 miles southwest of Coolidge, Arizona. The aircraft was en route from Tucson to Phoenix, Arizona. All 10 passengers and the two crewmembers received fatal injuries.

Witnesses in the area observed the aircraft enter into a dive from which it did not recover.

The National Transportation Safety Board determines that the probable cause of this accident was the inflight failure and subsequent separation of the right wing. This failure was the result of a fatigue fracture in the lower main root joint fitting which propagated from an area of corrosion and fretting damage which, in turn, was caused by design deficiencies. These deficiencies remained undetected because surveillance of the supplemental type certification process and the modification programs was not adequate to assure compliance with design and inspection requirements.

INVESTIGATION

Apache Airlines Flight 33 of May 6, 1971, was a regularly scheduled air taxi flight from Tucson to Phoenix, Arizona. The aircraft was a DeHavilland Dove Model 104-7AXC (Carstedt CH-600A Conversion), N4922V. The flight departed Tucson International Airport, under visual conditions, at 1253 m.s.t.[1] with 10 passengers and a crew of two, and with 140 gallons of fuel on board.

At about 1315, ground witnesses near Coolidge, Arizona, observed the aircraft flying in a northwesterly direction. Some of these witnesses reported that they first heard loud engine noises emanating from the aircraft, and that the engine sound then ceased. According to two of the witnesses, the aircraft initially descended at a "slight angle" which steepened to a 45-to-50 degree dive angle. None of the witnesses saw smoke or fire while the aircraft was in the air, and none saw parts separate from the aircraft. According to the witnesses, there were high scattered cumulus clouds in the area. A pilot, who flew a light aircraft through that general area after the accident, reported that he encountered severe turbulence at about 4,200 feet in the Phoenix area.

The 1258 surface weather observation at Phoenix was reported as, measured ceiling of 4,800 feet broken clouds, 25,000 feet overcast, visibility 40 miles, wind from 250° at 8 knots, and an altimeter setting of 29.78 inches.

[1] All times shown are in mountain standard time (m.s.t.)

The crew was current in the aircraft and qualified for the operation involved.

The wreckage site, a flat plowed field, revealed two distinct craters with no impact marks between them. One crater was oriented north/south. Most of the fuselage and left wing wreckage was found north of that crater and within 200 feet of it. The other crater, which contained remains of the right wing and the right engine, was located approximately 35 feet southeast of the fuselage crater. Scattered parts of the right wing were located northeast of the crater formed by that wing. Only one piece of wreckage was found outside the immediate area of the craters. This piece, a fairing from the lower aft wing root, was located 1,160 feet northwest (downwind) of the fuselage crater.

Although the entire airframe of N4922V was fragmented, the degree of fragmentation was greater on the right side of the fuselage. Both engines were recovered, still attached to portions of their wing attach structure. No evidence of any preexisting damage was noted on the engine mounts. The engines revealed no evidence of malfunction or failure prior to impact. Neither engine was producing power at impact. Both propellers were found in the feathering range: Disassembly and examination of both fuel controls, the fuel pumps, and the propeller governors revealed no discrepancies except for impact damage.

The remains of the inboard end of the right wing revealed general crushing in aft and outboard direction. The right wing front spar upper and lower main attachments were both failed. The front spar upper attachment was fractured at the lug ends of the center section boom, which is part of the fuselage. The ends of the boom lugs outboard of these fractures were wedged firmly in the upper main root fittings and were found with the remains of the right wing. When the fracture surfaces of the lugs are held together, the right front wing spar is deflected upward approximately 80" with respect to its fuselage attach structure. The front spar lower attachment sustained a transverse fracture through the attach bolt hole in the wing lower main root joint fitting. This fitting is a single lug designed to transmit major tensile loads between the wing and fuselage during flight. The Board's metallurgical examination of this fitting revealed fatigue markings over 95 percent of the section of the lug aft of the wing attach bolt. The remaining 5 percent of the fracture in that section (the lower aft corner) and the entire fracture in the forward section of the lug were typical ductile overload separations. The fatigue markings were partially obscured by numerous small gouges that were determined to have been produced when the inboard end of the fitting impacted the ground. The origin of the fatigue fracture at the upper aft wall of the bolt hole in an area of fretting and corrosion pitting. A remanent of what appeared to be a small surface pit was found at the origin. A similar area of fretting was found on the lower forward wall of the hole. Further metallurgical examination revealed that the chemical composition of the fitting material was within the limits prescribed by the applicable material specification and that the microstructure of the steel in the origin area was normal for the specified material (4130 alloy steel, hardened and tempered). However, a series of hardness measurements taken on sections of the fitting indicated that the tensile strength of the material varied widely, with an average value near 157,000 pounds per square inch (p.s.i.). The manufacturing drawing required that the part be heat treated to a tensile strength of 180,000 to 200,000 p.s.i.

The aircraft had been maintained on a progressive maintenance cycle and the inspections had been performed at the designated time intervals. An eddy current inspection of the right wing lower main fitting had been performed 1,651 hours prior to the accident.

At that time, no evidence of a fatigue crack was noted. This inspection was performed in compliance with Airworthiness Directive (AD) 70-15-6. That AD, which resulted from a prior accident involving a standard Dove aircraft, required inspection of all DH-104 wing fittings at 2,500 hour intervals. The last visual inspection of the fitting was made on March 24, 1971, approximately 2 weeks prior to the accident. This inspection did not require removal of the attached bolt.

The aircraft had been modified in accordance with engineering approved by Supplemental Type Certificate (STC) SA1747WE. The STC, dated July 23, 1968, was issued to Von Carstedt Corporation, C-Air, Long Beach, California.

The aircraft modification consisted primarily of the installation of two AiResearch TPE 331 series engines, an increase in fuselage length, and relocation of the wing fuel tanks, Von Carstedt subcontracted the engineering associated with this modification to Strato Engineering Company, Burbank, California. The heat treat of various fittings was subcontracted to Comet Steel Treating Company, Signal Hill, California.

One significant aspect of this modification was the redesign of the wing lower main root joint fittings to accommodate the new engine installation and the relocation of the fuel tanks. The new fitting, part number CPD-2004, was structurally similar to the original fitting. This similarity was the basis upon which design approval was issued without a requirement for substantiating fatigue tests. The fatigue life of the CPD-2004 fitting was predicated upon the life of the original DeHavilland fitting, provided that the new fitting maintained the same precise tolerances and joint sealing procedures employed in substantiating the life of the original DeHavilland fitting. The critical nature of these procedures and tolerances was reported by DeHavilland in 1964 after that company failed a lower wing fitting at less than 25 percent of its predicted life during fatigue tests. DeHavilland established that this premature fatigue failure was caused by corrosion and fretting of the fitting.

The stress analysis submitted to the Federal Aviation Administration by Strato regarding the CPD-2004 fitting noted that the service life of the fitting was predicated upon maintenance of the DeHavilland tolerances. However, the engineering drawing which was prepared, checked and released by Strato and subsequently approved by the FAA as part of the STC data Specified a tolerance which could result in 0.0022 inches greater diametrical clearance than that specified in the fatigue analysis.

The fatigue life of the CPD-2004 fitting was also predicated, in part, upon the use of a material with a higher allowable ultimate tensile strength than that used for the original fitting. Accordingly, the design drawing specified that the fitting was to be constructed from 4130 alloy steel heat treated to a tensile strength from 180,000 to 200,000 p.s.i. The drawing did not, however, specify the process by which this heat treat was to be accomplished. According to Military Handbook 5A, which was used in the design of this modification, a part fabricated from 4130 alloy steel with the size and geometry of this fitting could not be consistently hardened throughout the section thickness to attain he specified tensile strength; tables in the handbook indicate that 4340 alloy steel would be preferred in order to attain the desired strength level.

Because of its interest in the types of aircraft currently in use in air taxi operations, the Board not only reviewed the modification of this aircraft but also the process by which the aircraft was certificated. Supplemental type certification is used when changes to the existing type certificate are not considered significant enough to require a new type certificate (TC); the STC is considered an amendment to the original TC.

The applicant for an STC must show that the altered product meets applicable airworthiness requirements. However, the responsibility for assurance that the modification of the aircraft meets the standards of the Federal Aviation Regulations rests with the FAA and is accomplished by FAA Engineering and Manufacturing/Aircraft Engineering personnel in the regional offices.

In actual practice, most of the review of an STC program is accomplished by employees of the applicant who act as representatives of the FAA, and who are titled Designated Engineering Representatives (DER's). DER's are appointed at the convenience of the FAA; they are guided by the same requirements, instructions and procedures as FAA employees; and the amount of review of their work is dependent, in part, upon the confidence the FAA regional personnel have in their capability.

The duties and responsibilities delegated to a DER are outlined in FAA Handbook 8110.4, "Type Certification." That handbook notes that a DER has the authority either to approve specific data (subject to spot review by the FAA) or to recommend that FAA approve the data. The handbook also notes that, in approving data, the DER must completely satisfy himself that all pertinent FAR requirements are complied with. He must accept the responsibility for approving the technical data as complying at least with the prescribed minimum airworthiness standards. However, the Chief Engineer of Strato Engineering Company, who functioned as a DER in the structures and flight test areas, testified that in one case his signature on technical data merely indicated that he had reviewed the data and that he thought it was a proper document. In arriving at this conclusion, he approved the general approach used in the calculations, but he did not check the numerical accuracy. He felt that actual approval of the data was the responsibility of the FAA. He also noted that, although he initialed drawing CPD-2004 as a DER, he did not check it for material strength allowables.

Another DER on this project testified that, with the exception of Handbook 8110.4, he had not been provided guidance regarding his duties and responsibilities as a DER.

In addition to its responsibility for design adequacy, the FAA has a responsibility to assure that the modified aircraft conforms to the design drawings. The conformity inspections of the aircraft were performed by FAA Manufacturing Inspectors from the local district office. The inspector who performed the majority of these inspections said that these inspections were done on a sampling basis. He also said that he had no instructions from the regional engineering personnel as to what he should inspect or check.

Although the discrepancy in the material selection/heat treatment criteria remained undetected, the Board noted that the manufacturing inspector rejected the fitting on the basis of its strength. This part was rejected because a hardness test on another part from the same heat treat lot was not within its hardness specifications and the entire lot was rejected. The inspector did not, however, followup to assure compliance with his request for a subsequent inspection to determine that this part was properly heat treated. Although the procedures used for the ultimate acceptance of this part were never determined, the fitting was subsequently installed, and the aircraft was certificated.

ANALYSIS AND CONCLUSIONS

The 35-foot separation between the distinct craters formed by the right wing and by the fuselage shows that the wing separated from the fuselage before ground impact. Other facts support this finding. The generally more severe destruction of the right side of the fuselage indicated that the fuselage impacted on its right side. The impact gouges on the fracture surface of the CPD-2004 fitting and the general outboard and aft crushing of the wing both suggest that this wing first impacted on its forward root end. Damage of this nature to both the fuselage and the wing would not have been possible if the wing had remained in place on the fuselage until ground impact. The proximity of the two impact craters and the fact that the witnesses saw nothing separate from the aircraft suggest that detachment of the wing occurred abruptly at a relatively low altitude, probably just prior to ground impact.

Such an occurrence would result in the aircraft rolling to the right, and this is consistent with the damage observed on the fuselage.

The orientation in which the ends of the wing center section Loom fitting were found wedged into the right front spar upper fittings indicated that this wing first rotated upward approximately 80° before it finally separated.

The events leading up to the separation of the right wing can only Le postulated. However, the Board concluded that N4922V began a descent from its cruise altitude, and that both propellers were feathered either prior to, or during this descent.

No physical evidence of any condition which would warrant the inflight shutdown of both engines was observed in the examination of the engines, the engine accessories, the propellers, the propeller governors, the engine fuel system, or the engine attach structure. However, the evidence indicates that both propellers were feathered by the crew, probably after an indication of some serious emergency situation which apparently did not involve the engines. Furthermore, it appears to the Board that the rapid descent may have been initiated by the crew in an attempt to cope with that emergency situation.

During the investigation only one discrepancy was found which could have triggered such a response by the crew. This was the fatigue failure of the right wing lower main joint fitting.

Because of the preexisting fatigue damage, the load-carrying capability of the wing joint had been reduced considerably. Thus, an encounter with turbulence such as that encountered by the private pilot in the Phoenix area could have precipitated the failure of the severely weakened aft side of the fitting. The remaining section of the fitting may have begun to deform at that time, without complete failure occurring. This deformation is indicated by the separation of the wing fairing which normally covers the aft spar fitting. The fairing was found 1,160 feet northeast of the fuselage crater, suggesting that the panel separated before the wing and that it drifted downwind to that location during its descent.

Thus, in summary, it appears that the wing failure was progressive in nature. The aft side of the CPD-2004 fitting failed at cruise altitude: the aircraft then descended rapidly to a low altitude where the remaining wing support structures failed, permitting the wing fiist to deflect upward, and then to separate completely from the fuselage an instant before ground impact.

It is the opinion of the Board that the cause of the wing separation must be attributed to the preexisting fatigue crack in the right-hand CPD-2004 fitting. The initiating source for this fatigue was a small pit formed by fretting between the wing attach bolt and the wall of the attach bolt hole. The fretting was, in turn, likely caused by localized high bearing stresses at the upper aft and lower forward walls of the bolt hole.

After if had examined the cause of the wing separation, the Board then directed its efforts toward determining the underlying factors which permitted this fatigue failure. The results of that phase of the investigation led back to the modification of the aircraft from a standard Dove to the Von Carstedt Model 104-7AXC.

In reviewing the design of this modification the Board noted two errors which affected the fatigue life and load-carrying capability of the CPD-2004 fitting. One of these was, the failure to transfer information regarding dimensional tolerances from the design data to the engineering drawing from which the parts were manufactured. This omission seems particularly significant to the Board in view of the known premature failure of the DeHavilland fatigue test specimen, and that company's finding that the failure was related to bolt clearances. Although deformation of the failed fitting in the accident aircraft precluded the determination of the actual diameter of the hole, the hole tolerance callout on the engineering drawing was considerably larger than that specified in the fatigue data. Excessive clearances could have caused high bearing stresses at the hole wall. The Board, therefore, concludes that this increase in clearance may have contributed to the initiation of the fracture.

The other error was the selection of an alloy steel (4130) that did not harden uniformly in the various sections of the fitting when the part was heat treated. This resulted in a fitting which had a lower average tensile strength than the value used in the stress analysis. The Board believes that this lower strength may also have contributed to the premature failure of the fitting.

In addition to the influence of the design errors on the cause of this accident, other facets of the certification program must be considered significant. For example, both of the errors discussed might have been detected if the DER's had properly reviewed the design data and engineering drawings which they, in effect approved by affixing their signatures or initials thereto. However, the Board noted that the DER's involved were not fully aware of the responsibilities associated with that position. Also, the erroneous heat treatment callout on the design drawing might well have been detected by the Manufacturing Inspector if he had followed up on his rejection of the entire lot in which the fitting was heat treated.

Thus, the factors which permitted certification of this aircraft seem to derive from the general nature of the implementation of the STC program. In theory the system may work well, but, as implemented in this case, it allowed this problem to develop. In retrospect, it is quite clear that adequate communication among all parties concerned, and increased surveillance by the FAA of the STC process and of the parties implementing this program, might have prevented this accident.

PROBABLE CAUSE

The National Transportation Safety Board determines that the probable cause of this accident was the inflight failure and subsequent separation of the right wing. This failure was the result of a fatigue fracture in the lower main root joint fitting which propagated from an area of corrosion and fretting damage which, in turn, was caused by design deficiencies. These deficiencies remained undetected because surveillance of the supplemental type certification process and the modification programs was not adequate to assure compliance with design and inspection requirements.

RECOMMENDATIONS

1. On May 12, 1971, the National Transportation Safety Board submitted the following recommendations to the Administrator of the Federal Aviation Administration:

(a) Conduct a one-time metallurgical inspection on an expedited basis by approved methods of all lower main wing spar root fittings PIN CPD-2004 on all DeHavilland Model DH-104 "Dove" airplanes that have been modified under STC No. SA1747WE.

(b) Review the adequacy of Airworthiness Directive 70-15-6 and revise as necessary to assure adequate service limits on this fitting.

In his reply dated May 17, 1971, the Administrator stated that the FAA agreed with the Board recommendations and that corrective action would be taken pending completion of an engineering evaluation. He also noted that the airworthiness certificate of these aircraft had been suspended on May 11, 1971.

On October 22, 1971, the FAA issued an addendum to STC SA1747WE. This addendum provided for the installation of a steel reinforcing strap on the lower front spar cap, and the replacement of the upper wing fittings with identical parts fabricated from 4340 steel. The addendum stated that AD 70-15-6 is not applicable to aircraft modified in accordance with that STC.

2. As a result of data developed later in this investigation, the Board has concluded that the problems associated with the certification of this aircraft indicate a need for possible remedial action to assure the airworthiness of aircraft modified under such programs. Accordingly, the Board recommends that the Federal Aviation Administration reevaluate its STC program to ensure continuity in quality control in the supplemental type certification process.

BY THE NATIONAL TRANSPORTATION SAFETY BOARD:

/s/ JOHN H. REED
Chairman

/s/ OSCAR M. LAUREL
Member

/s/ FRANCIS H. McADAMS
Member

/s/ LOUIS M. THAYER
Member

/s/ ISABEL A. BURGESS
Member

June 1, 1972

INVESTIGATION AND HEARING

1. Investigation

The Board received notification of the accident at 1437 m.s.t. on May 6, 1971. Board investigators were dispatched to the scene from the Los Angeles, California, Field Office and from Board headquarters at Washington, D. C. Working groups were established for Operations, Records, Structures, and Powerplants. Interested parties were the Federal Aviation Administration, Apache Airlines, Inc., and AiResearch Manufacturing Company. The on-scene phase of the investigation was completed on May 14, 1971.

2. Hearing

A public hearing was held in Scottsdale, Arizona, on July 21 and 22, 1971. Parties to the investigation were the Federal Aviation Administration and Apache Airlines, Inc.

3. Preliminary Reports

An interim report of investigation summarizing the facts disclosed during the field phase of the investigation was published on June 6, 1971.

CREW INFORMATION

Captain Ted N. Huntington, aged 31, possessed Commercial Pilot Certificate No. 15554058 dated August 20, 1966, with airplane single- and multiengine land and instrument ratings. He had a first-class medical certificate dated July 6, 1970, with the limitation that he wear corrective glasses while exercising the privileges of his pilot certificate. Captain Huntington was current in the DeHavilland Dove and was qualified for the operation involved. His total flight time prior to the accident was about *6,000* hours, 2,500 of which were as pilot-in-command in the DeHavilland Dove. He had flown 9 hours in the last 24 hours.

First Officer Donald B. Nelson, aged 30, possessed Commercial Pilot Certificate No. 1688806 dated August 21, 1968, with airplane single- and multiengine land and instrument ratings. He had a first-class medical certificate dated July 23, 1970, with no limitations. First Officer Nelson was current in the DeHavilland Dove and was qualified for the operation involved. His total flight time prior to the accident was about 3,500 hours, 2,000 of which were in the DeHavilland Dove aircraft.

AIRCRAFT INFORMATION

The DeHavilland Dove 104 was originally a 12-place aircraft powered by two reciprocating engines. The fuel cells were placed in each wing between the engines and the fuselage. The maximum *gross* weight was 8,950 pounds.

The Carstedt conversion DeHavilland Dove 104-7AXC, called CJ-600, has two AiResearch jet engines, driving two Hartzell propellers. The fuel cells have been relocated outboard of engines. The maximum gross weight has been increased to 10,500 pounds. The fuselage was lengthened both fore and aft of the wing, the passenger capacity was increased from 12 to 18, and the gross weight was increased to 10,500 pounds.

The Von Carstedt conversion of the DeHavilland Dove was engineered by Strato Engineering Co., Glendale, California, for C & W Aviation a company owned by Mr. Carstedt which accomplished the conversion. The Western Region FAA Engineering Division of Flight Standards monitored and approved the engineering which resulted in the issuance of the Supplemental Type Certificate (STC). The certificate was issued to Mr. Carstedt and was held by him from 1968 to June 1, 1971, at which time the STC was returned to the Western Region FAA Engineering Office.