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AIRCRAFT ACCIDENT REPORT #H90002
TRANSPORT CANADA BEECHCRAFT 65-90
REGISTRATION C-FCAS
SHERRINGTON, QUEBEC
MAY 1, 1979

AVIATION
SAFETY BUREAU

AVIATION SAFETY
INVESTIGATION

BUREAU DE LA SÉCURITÉ
AÉRONAUTIQUE

ENQUÊTE SUR LA SÉCURITÉ
DE L'AVIATION

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TP 1090

"This accident was investigated to provide guidance toward the prevention of a recurrence. The content of this report is confined to relevant circumstances and is published for accident prevention purposes only."

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SYNOPSIS

On 1 May 1979 a Transport Canada King Air 65-90 was calibrating a radar antenna located at Dorval airport. While flying a 20 mile orbit at 1700', the right wing separated from the aircraft. The aircraft crashed in open farm land and was destroyed. Both occupants were killed.

This accident was investigated and this report was prepared by the Aviation Safety Investigation Division, Aviation Safety Bureau, Transport Canada.

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Honourable the Minister of Transport

This Accident Investigation and this Report have been audited by the Aircraft Accident Review Board. The Board has considered all information available including that from involved parties. The Board agrees with the contents of this Report for release of public information.

It will be noted however that although the essential facts pertaining to the immediate cause of the accident have been identified and evaluated, there are some fundamental issues of metallurgy that have not been fully resolved.

The definite resolution of these issues requires many more months of work in research laboratories, and it is the opinion of ASI and of the Board that the public interest is best served by the issuance of the present report, pending the availability of additional basic scientific information.

Aircraft Accident Review Board
Approved Nov 20, 1980

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1. FACTUAL INFORMATION

1.1 History of Flight

C-FCAS, a Beech 90, was utilized for calibration of navigational and approach aids. C-FCAS took off from Dorval Airport (Montreal) with a crew of two at 1341Z* on the 1 May 1979 to calibrate a radar antenna.

1.1.1 At 8 NM from the airport, the aircraft established contact with the radar calibration team on 121.4 MHz. The calibration team was located in the Montreal Control Centre in the Terminal Building. The first calibration manoeuvre involved flying a 20 nautical mile orbit of the antenna at 1700' AGL. The crew advised radar that they would fly a counter clockwise orbit starting at the 280° radial. They subsequently reported that they were established in the orbit and the aircraft was observed by the calibration team to be covering 5° of orbit every 2 or 3 sweeps of the radar antenna.

1.1.2 As the aircraft crossed the 165° radial the radar operator observed target quality deteriorating followed by a complete loss of return on the next scan. After 2 more scans, 3 cycles of an ELT signal were heard at 1405Z but these were not immediately associated with C-FCAS. (A radar technician observed that it was unusual for a "hole" to suddenly appear in the radar coverage where none had existed before.)

* All times are "Z" (local + 4 hours)

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1.1.3 The radar operator made several unsuccessful attempts to contact CAS on 121.4 MHz. The flight check supervisor initiated a communications search but the aircraft could not be raised. The supervisor then requested sector control to send an aircraft to overfly CAS's last known position, (165° radial @ 20 N.M.), and advised the Quebec Provincial Police and other agencies. The searching aircraft reported smoke and heavily damaged wreckage in the vicinity of CAS's last known radar position. Shortly thereafter the Quebec Provincial Police reported an aircraft crash in the area of Sherrington, Quebec, located close to the 165° radial. In-flight separation of a wing had been observed by an eye-witness on the ground. Subsequent investigation verified the wreckage was that of C-FCAS. Later examination at the crash site revealed apparent failure of the right lower outboard wing attachment fitting which caused separation of the wing adjacent to the right engine nacelle. The section of wing was found some 2100' from the main wreckage.

1.1.4 It is estimated that impact with the ground took place about 12 seconds after wing separation.

1.2 Injuries to Persons

The crew of two, the pilot and co-pilot, were killed in the crash.

1.3 Damage to Aircraft

The aircraft was destroyed.

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1.4 Other Damage

1.4.1 There were deep gouges in the ground and surface fuel burns from spilled fuel in the area.

1.4.2 Access to the wreckage, which was approximately 3/4 mile from a road, was through farm fields adjacent to the crash area.

1.5 Crew Information

1.5.1 The pilot-in-command who was in the right hand seat, was 46 years of age, possessed an ATR pilot licence, and had acquired more than 13,000 flying hours on a variety of single and multi-engine aircraft. At the time of the accident his licence was endorsed for Douglas DC3, DC4, DC8, Constellation and Viscount. His medical examinations did not indicate any medical condition that could have impaired his performance as a pilot.

1.5.2 The co-pilot who occupied the left seat, was 39 years of age, possessed an ATR pilot licence and had acquired approximately 4400 hours on single and multi-engine aircraft and helicopters. His medical examinations did not indicate any medical condition that could have impaired his performance as a pilot except that he was required to have glasses available.

1.6 Aircraft Information

1.6.1 (a) Type	King Air 65-90
Manufacturer	Beechcraft
Serial Number	LJ23
Owner	Transport Canada
Certificate of Airworthiness	Valid (See Text)
Certificate of Registration	Valid
Total Airframe Hours	5466.00
Power Plants	2 P&W PT6A-20

(b) The aircraft, based in Montreal, had no reported existing defects in the journey log that could be related to the accident. The maintenance documentation indicated that an inspection as outlined in AD 70-25-04 and service instruction 0394-018 had been carried out at 4907 hours when the aircraft was at the Transport Canada main base in Ottawa for maintenance. However, examination of the aircraft technical log after the accident revealed that no further inspection of the lower forward "bathtub" wing fittings had been done.

(c) The fuel used was aviation type JP-4.

(d) Calculated gross weight before taxiing was 9406 lbs with a fuel load of 2432 lbs. The aircraft was within allowable C of G limits and 56 lbs. over allowable ramp weight.

1.7

Meteorological Information

Actual conditions issued by the Quebec Forecast Office for 1300Z to 1500Z indicated scattered to broken cumulus clouds, and a few towering cumulus clouds within 150 miles of the Montreal area. Widely scattered rain showers were also evident in the same region. A radar report at 1345Z indicated heavy cumulus clouds at 120 nautical miles southeast and south south-west of Montreal with light rain showers and tops to 15000 feet.

The approximate time of the accident was 1405Z. Hourly weather observations for 1400Z from 3 airports in the area recorded scattered cloud at 2500' and 3000', broken cloud at 5000' and 9000', visibilities of 12 and 15 miles, temperature 10°C and winds from 230° to 260° of 7 and 8 knots.

1.8

Aids to Navigation

All navigational aids were operating normally.

1.9

Communication

The aircraft was communicating with the Dorval radar calibration team on 121.4 MHz. There was no recorded or apparent difficulty in communication between the flight crew, tower or calibration team.

1.10 Aerodrome Conditions

Not applicable.

1.11 Flight Recorders

Recorders were not installed or required.

1.12 Wreckage and Impact Information

1.12.1 The aircraft right wing separation occurred while the aircraft was flying on an easterly heading. The impact angle was estimated at 35°. An intense fire erupted on impact and much of the wreckage was consumed by fire. In addition to the right outboard wing assembly which landed 2100' to the west of the accident site, the major parts that escaped fire damage were a portion of the right wing center section, the right undercarriage, and the entrance door. The upper and lower wing attachment fittings (bathtub fittings) for both the main and rear spar attachments were on the center section structure together with the portions of the fittings which had fractured from the right wing.

1.12.2 The fracture faces of the attachment fittings had not received any significant fire or post separation impact damage. Examination of the fracture faces showed that the upper front and both the upper and lower rear spar fittings had fractured from overload. The lower front spar fitting fracture surface appeared to be characteristic of a progressive type of cracking.

1.12.3 The positions of the primary flight control surfaces were not determinable. The rudder trim tab was deflected approximately 1" to the left. The elevator trim tabs were in the full up position.

1.12.4 The flap actuator indicated that the left flap was fully retracted at impact. The right flap actuator had partly separated and flap hinge rivets had sheared, resulting in a post-impact configuration of the right flap being slightly down. Trapped instrument readings indicated that both engines were developing substantial power at the time of impact. The aircraft struck the ground inverted. The left engine had penetrated some 3' into the clay type soil, while the right engine had buried itself to a depth of 2½'. Post impact fire had consumed the bulk of the compressor inlet cases, the accessory gearboxes and accessory components on both engines.

1.12.5 The ELT was destroyed by fire.

1.13 Medical and Pathological Information

1.13.1 The Pilot-in-Command The pattern of injury showed that this pilot was sitting on the right side of the cockpit and upon impact was thrown forward and to the left, over the throttle quadrant. His seatbelt had been fastened. The injury pattern also indicated that the ailerons were fully hard over to the left (right wing up). Massive injuries to the head were consistent with an impact of at least 120 g. No alcohol and only 1.6% saturation of carbon monoxide was found in his tissues.

1.13.2 The post-mortem tissue lactate profile indicated an acute stress reaction time of 8 to 9 seconds.

1.13.3 The Co-pilot This pilot was occupying the left seat in the cockpit. The pattern of injury indicated that he was holding the control wheel hard over to the left (right wing up) while pushing on the right rudder pedal. His seatbelt had been fastened. The injuries also indicated that the impact forces came from an anterior-posterior direction, slightly from the left with an impact of at least 110 G. No alcohol or carbon monoxide were found in his tissues.

1.13.4 Post-mortem tissue lactate profile indicated an acute stress reaction of approximately 7 seconds.

1.14 Fire

1.14.1 An eyewitness reported that fire occurred on impact. A burn pattern on the terrain illustrated that fuel was sprayed in various directions, principally in the direction of the final flight path. The source of ignition is not known and is presumed to be either a hot engine or electrical.

1.14.2 The fire smouldered for over 24 hours.

1.14.3 Firefighting equipment was not used due to the inaccessibility of the site from the road.

1.15 Survival Aspects

The accident was not survivable.

1.16 Tests and Research

1.16.1 Failed Wing Attachment Fitting

Field investigation indicated two of the four outboard wing attachment fittings failed in overload mode and a bolt for a third fitting also failed in overload. The forward lower outboard "bathtub" wing attachment fitting (P/N 50-110057), however, showed evidence of a massive precrack of a progressive nature (see Photos 1 through 3). The magnitude of the observed precrack in the right lower forward outboard wing attachment (bathtub) fitting at the time of the accident was such that sudden total failure of the remaining uncracked cross-section could occur under operational loading typical of the service in which the aircraft was engaged (i.e. there was no need for the application of a massive gust load or severe pilot-induced load to cause separation). The remaining three attachment assemblies were not considered to be (nor were they required to be) capable of carrying all flight loads within the manoeuvre envelope if the subject fitting suffered total failure. It follows that failure of the bathtub fitting would lead to separation of the outboard wing section.

1.16.1.1 The subject fitting was submitted for laboratory examination by the Department of Transport's Aviation Safety Engineering Facility (Ref. 1). Because of the discovery of an unusual material deficiency which appeared to have had a very significant influence on the fitting's failure, additional laboratory examination was carried out by Battelle Columbus Laboratories for the F.A.A. (Ref. 2) and by the National Research Council of Canada at the request of the Aircraft Accident Review Board.

1.16.1.2 The laboratory evaluation initially concentrated on scanning electron microscopic (SEM) evaluation of the inboard fractured remnant shown in Photo 4 to determine the mode and mechanism of cracking. SEM evaluation indicated there were three fairly distinct zones of differing fractographic appearance. In zone "a" of Photo 5, crack growth initiated at a zone of intergranular separation at the crack origin "0", and grew thereafter in a mixed mode of intergranular cracking and transgranular fatigue cracking as illustrated by areas "i" and areas "f" respectively in Photo 6. In zone "b" of Photo 4 the cracking was virtually all intergranular in nature (similar to areas "i" of Photo 6). In zone "c" of Photo 5 the cracking was virtually all characteristic of transgranular fatigue cracking (similar to areas "f" of Photo 6). Photo 7 shows a characteristic area of zone "c" cracking in which it can be seen that the variable crack arrest line spacing was typical of operational loading induced fatigue crack growth. All of the intergranular areas of fracture were characterised by an irregular dispersion of roughly hemispherical holes on the grain boundary surfaces (see areas "i" of Photo 6).

1.16.1.3 Photo 8 shows multiple path intergranular cracking was still occurring even when the crack had grown to the extent of breaking through the outer surface of the "bathtub" (having initiated at the inner surface). Photo 9 shows crack initiation occurred at a location where machining had exposed "end grain" structure (i.e. the predominant forging grain flow was essentially perpendicular to the free surface). Photo 10 shows a typical metallurgical section through the forging in the vicinity of failure. Voids, of essentially spherical shape and of up to 0.001 inch diameter, were found distributed throughout the section along the grain boundaries. These voids were consistent (in size, shape and distribution) with the hemispherical holes observed on the intergranular areas of the fracture surfaces. Further metallurgical and material evaluation indicated the failed fitting was manufactured from the specified 2014 aluminum alloy, had been heat treated to the specified -T6 temper, and appeared to meet all general material property and dimensional requirements.

1.16.2 Discussion of "bathtub" fittings and applied loading

1.16.2.1 Manufacturer's information on wing attachment fittings*

The subject fitting (P/N 50-110057) was used in production of Queen Airs up to serial number LD-270 & King Airs up to serial number LJ-67. P/N 50-11057 may be termed the "Generation 1" fitting. Queen

* From evidence obtained at Coroner's Inquest on 7 and 8 October 1980.

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and King Airs do not have a certification basis that specifies fatigue evaluation of the load-bearing structure; however tests to establish fatigue life of wing and wing carry-through structure were carried out by Beechcraft for the structurally similar Beech Model 99. The Beech Model 99 fatigue test was performed using wing attachment fittings to P/N 50-110097; (-1 and -2) which may be termed the "Generation 2" fitting. Significant changes were an increase in "bathtub" section from 1.44 square inches (50-110057) to 1.88 square inches (50-110097), an increase in total length from 11.84 inches (50-110057) to 14.58" (50-110097), and an increase in thickness clamped by the bolt from 0.75 inch (50-110057) to 0.85 inch (50-110097). The stronger Generation 2 fittings were installed on all Beech Model 99's from Serial number U-1, Queen Airs from serial number LD-271, and King Airs from serial number LJ-68. This corresponds to aircraft produced after August-September 1965. In the Beech Model 99 fatigue test the left wing attachment fitting reportedly failed at 14,200 "flight hours" (8,500 "test hours") while the right wing fitting was reportedly cracked approximately 0.5 inch at that time and did not fail until after 20,000 "flight hours". These Generation 2 fittings cracked in the same plane and at the same approximate origin location as the failed fitting from C-FCAS. A "Generation 3" fitting was introduced in King Airs LJ-482 and on a Model 99's U-129 and on in September-October 1969. These are P/N 50-110097 (-3 & -4) and are identical dimensionally to Generation 2. However Generation 2 forgings were received in the -T6 temper and then finish machined; whereas Generation 3 forgings are received in the -T4 temper, are finish machined and then artificially aged to the

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-T6 temper by Beech. The change in manufacturing procedure was reported by Beech to be intended to minimize or eliminate the potential for stress corrosion cracking (see Appendix C, Figure 3 and associated narrative).

1.16.2.2 Flight Loads on DOT Navaid Calibration Aircraft

There being no generally accepted flight load spectrum that could be used for a fatigue life assessment of a King Air 65-90 when used on calibration missions, an effort was made to obtain such data by instrumenting a small number of aircraft with counting accelerometers. A total of 579 hours of accelerometer data were collected from five aircraft, and the data were analysed for each of the various missions flown by the aircraft. The results of the analysis are reported in LTR ST 1199. The statistical basis for a firm conclusion regarding the absolute rates of fatigue damage accumulation is somewhat limited, but in comparative terms the ILS role appears to be more damaging than enroute flying by up to an order of magnitude. The conclusions are attached as Appendix D to this report.

1.16.2.3 Aircraft Utilization

A review of the aircraft journey log for the period, 23 Feb 76 to 27 Dec 78, indicates a total of 1087:35 hours air time. A breakdown of the flying hours attests to the following aircraft utilization:

Calibration	799:25
Ferry below 5000'	19:55
Ferry 5000' to 10000'	65:35
Ferry above 10000'	109:45
Training below 5000'	69:20
Training above 5000'	23:35

The aircraft was rarely used for transport so this type of mission was listed under ferry time. The bulk of ferry hours is composed of flying time spent in transit to and from navigation and approach facilities that were to be calibrated. If the ferry and training hours carried out below 5000' where the aircraft was exposed to more severe turbulence, were included with time spent on calibration, the usage at low altitude would have been over 80% of actual hours flown.

1.16.3 Discussion of Inspection Requirements and Evidence of Compliance with Requirements

1.16.3.1 In accordance with AD 70-25-04 the failed fitting in C-FCAS was due for initial inspection at 5000 hours, using dye penetrant methods as described in SI 0394-018 (see Appendices B and C). The AD also requires further inspections, but the text used may lead to other interpretations, one of which would require such further inspection at 500 hour intervals, while another would indicate that further dye penetrant inspection be carried out only if wing skin cracks were found at 100 hour inspection intervals. From witness and documentary

evidence it was indicated that an inspection, using dye penetrant, was performed at 4907 hours when the aircraft was available for other extensive maintenance. Dye penetrant inspection was not done at 5407 hours according to witness and documentary evidence. The consensus among technical and engineering staff is that the crack would probably have been detected if a dye penetrant inspection of the failed fitting had been accomplished at 5407 hours.

1.16.3.2 There is well documented and independently cross-referenced evidence that dye penetrant inspection per AD 70-25-04 was performed at 4,907 hours, some 19 months and 559 flying hours prior to the accident, and that the results were negative. The inspector who carried out the inspection is reputed to be a careful, reliable and painstaking technician who invariably masked off areas to be penetrant tested, and subsequent to inspection invariably cleaned off the inspected areas with both solvent and detergent cleaners. The available evidence supported the inspection at 4,907 hours having been properly carried out, and that the negative results at that time reflect an absence of detectable cracking at that time.

1.16.3.3 A review was made of the maintenance procedures of a number of other operators of Beech King Air Aircraft to determine their interpretation of the applicable Airworthiness Directive. This study revealed a number of different interpretations by Maintenance Supervisors and AME's (see also para 2.1.4) of the intent of the originators of the Airworthiness Directive.

1.16.4 Overview of Problems with Fittings Similar to the Failed C-FCAS Fitting

1.16.4.1 Review of Beech and F.A.A. Activities with Respect to Fitting Problems (Pre- and Post-Accident)

In November 1970, Beech issued Service Instructions 0393-018 and 0394-018, (Queen Airs and King Airs respectively) in which (in part) they drew attention to the necessity of inspecting fittings to P/N 50-110057 for evidence of fatigue or stress corrosion cracking. (note that these S.I.'s also cover inspection of many other parts of the wing structure.) The Beech Model 99 fatigue test program had been initiated in early 1969 and ran for about 2 years. The fatigue crack indication location delineated in Figure 2 of these S.I.'s matches the fatigue crack locations in the stronger P/N 50-110097 fittings of the Beech Model 99 fatigue test airframe. (Note, Transport Canada has been unable to determine how the stress corrosion crack indication locations delineated in Figure 3 of these SI's were established). The FAA made the SI's mandatory for both aircraft series which then shared the P/N 50-11057 fittings (The Generation 1 fittings). All cracking detected to date in these fittings has occurred in Generation 1 fittings.

1.16.4.2 Prior to the accident to C-FCAS, inspections of other aircraft by their operators had detected cracking in four Queen Air fittings in the location of the fracture in C-FCAS. (Three other Queen Air fittings had also been found cracked in other locations ascribed by

the Service Instructions to stress corrosion cracking.) The 4 Queen Air fittings were reportedly examined by Beech, who attributed the cracking to "stress corrosion initiation" followed by growth via a "fatigue/corrosion fatigue" mechanism. The above successful detection of such cracking at an adequately early stage of growth served to reassure Beech and the FAA that the Service Instructions and Airworthiness Directives were working effectively to prevent total failures of these fittings.

1.16.4.3 Immediately after the accident to C-FCAS, Transport Canada (by means of Canadian Airworthiness Directive CF-79-06, effective 7 May 1979) sought to ensure no repeat of the C-FCAS accident by requiring an immediate dye penetrant inspection of all fittings similar to the one that failed in C-FCAS in all similarly equipped Beech aircraft in Canada. Later, the FAA, subsequent to provision of detailed information by Transport Canada, released AD 79-16-01 on 27 July 1979 requiring inspection (on or before 7 September 1979) of 567 Beech Queen and King Airs containing fittings similar to the one that failed in C-FCAS. Canadian AD CF-79-13 further required compliance with AD 79-16-01 by Canadian operators of any of the referenced 567 aircraft.

1.16.5 Review of Lives of Similar Defective Fittings Revealed by Inspections

1.16.5.1 The FAA reported, in a letter to Transport Canada dated 20 December 1979, that they had at that time received 153 replies (of 567 aircraft) to AD 79-16-01. Two fittings in King Air aircraft were

reported, as a result of this inspection, to be cracked in essentially the same location as the crack in C-FCAS. The FAA noted -"These reports show a need to reassess inspection schedules in our Airworthiness Directives (AD's) 70-25-01 and 70-25-04"- . The letter included the following data:

AIRCRAFT		CRACK DISCOVERY	
MODEL	SERIAL NO.	HOURS IN SERVICE	YEAR
<u>QUEEN AIRS</u> 3000 hours initial AD inspection for Queen Airs			
65-A80	LD-197	5142	1977
65-A80	LD-201	2912	1977
65-A80	LD-220	4108	1974
65-A80	LD-226	3641	1971
<u>KING AIRS</u> 5000 hours initial AD inspection for King Airs			
65-90	LJ-23 (C-FCAS)	5466	1979
+65-90	LJ-14	2600	1979
+65-90	LJ-13	4300	1979

+ Inspection per AD 79-16-01 issued after accident of C-FCAS

The FAA verbally indicated that aircraft LD-197, LD-201, LD-220, LD-226, LJ-13, LJ-14 and LJ-23 were all manufactured during a single Beech production year. Of the above batch of fittings, all three King Air fittings are known to exhibit spherical voids along the grain boundaries as noted above.

1.16.5.2

As a result of the discovery of the cracks in LJ-13 and LJ-14 fittings the FAA arranged for an independent evaluation of the cracked LJ-13 fitting and the remains of the LJ-23 fittings by Battelle Memorial Institute, Columbus, Ohio. This evaluation was intended to evaluate the mechanism of failure and also to attempt to evaluate crack growth rates. Battelle's resulting Report (Ref. 2) was evaluated both by Transport Canada and by the National Research Council of Canada. It is concluded that all parties perceive the same physical evidence, and are in general agreement with a view advanced by Battelle that analytical procedures intended to quantify either the precise mechanism of failure or the crack propagation rate are of doubtful value in the present state of knowledge. A primary cause for this is the fact that the influence of the spherical voids found on the grain boundaries in the failed and cracked fittings on the material's fatigue and crack propagation resistance is unknown. An extensive literature search (Ref. 3) has failed to reveal any useful information in this key area. The Battelle evaluation of crack propagation rates, using best available data for unflawed material, has generated times to failure of anywhere from in excess of 100,000 hours to less than 250 hours depending upon the assumptions.

Of the possible theories that have been conceived to explain the fracture in even a qualitative sense, the most probable theory is based on the belief that the existence of the voids is to be regarded as a deficiency capable of adversely affecting the life of the fitting. In the light of further research, this conviction may have to be revised, but until new research data become available, and for

the purpose of this accident investigation, the voids will be regarded as a material deficiency in the metallurgical circumstances of the subject fitting.

2.0 ANALYSIS

2.1 In this analysis, from the viewpoint of safety action and despite the areas of uncertainty in determination of the precise mechanism and speed of failure, certain areas of concern are immediately apparent:-

- (i) Laboratory analysis has identified a deficiency in the manufacturing process which contributed to the fitting failure.
- (ii) Witness and documentary evidence indicate that the fitting was inspected at 4,907 hours and that no crack was detected at that time.
- (iii) The fitting was not re-inspected at 5,407 hours. It is probable that a detectable crack existed at this time, and its detection would have prevented the accident.
- (iv) It was found that the Airworthiness Directive 70-25-04 was interpreted in a variety of ways by maintenance supervisors and AMEs.

2.1.1 Laboratory analysis established that the failed fitting from LJ-23, and the similarly cracked fitting from LJ-13, exhibited a manufacturing deficiency. This deficiency consisted of a broad distribution of tiny spheroidal voids located on grain boundaries. It was not possible to positively determine how these voids were generated, but it is believed that they are a result of the formation of gas (probably hydrogen).

A detailed literature search (Ref. 3) has failed to positively establish either the gas bubble formation mechanism or the influence that the bubble's presence would have on crack propagation in such flawed material. It is, however, shown by the physical evidence that the progressive crack propagation path was apparently strongly influenced by the presence of these bubbles. It should be noted that no conventional quality control procedure (other than 100% destructive inspection) could have been expected to detect this manufacturing deficiency.

2.1.2. The weight of evidence in favour of a properly conducted dye penetrant inspection having been carried out on the subject failed fitting in C-FCAS at 4,907 hours is substantial. The lack of any reported crack at that time indicated an absence of a detectable crack. Beech and the FAA consider the specified inspection procedure should be adequate to detect any cracks of 0.2 to 0.3 inch or larger. Via other general test programs, it has been established that penetrant inspection of the type specified here can give a 50% probability of successful detection with 95% confidence for 0.5 inch cracks in similar components (Ref 4). Laboratory examination failed to provide, and after 19 months would not be expected to provide, any positive confirmation of dye penetrant inspection at 4,907 hours.

2.1.3. Witness and documentary evidence, and laboratory examination, all confirmed that the failed fitting in C-FCAS was not inspected by dye penetrant means at 5,407 hours. Although, as noted earlier, no realistic rate of crack propagation could be established; at 5,407

hours, only 59 hours and a month before the accident, it is considered that a detectable crack most probably existed in the fitting; and its detection would have prevented the accident occurring.

2.1.4 The format of AD 70-25-04 is such that if Para. B inspection reveals a positive indication, then Para. C must be accomplished immediately; but if Para. B inspection is negative no further action is required. The Montreal Region Transport Canada maintenance staff interpretation of the AD was that if the Para. A 100 hours inspection for skin cracking proved negative then no further action was required. They hence considered dye penetrant inspection of the fitting per Para. B was not required at 5,407 hours, given the absence of skin cracking. Post-accident investigation established that there was indeed no skin cracking present around the failed fitting. Review of several other maintenance operations determined that there were in existence several differing interpretations of the AD.

3.0 FINDINGS

3.1 The right hand lower forward outboard wing attachment fitting of C-FCAS had developed a progressive crack of sufficient magnitude to cause in-flight fracture, and consequent immediate wing separation outboard of the right hand engine nacelle, under normal flight loads.

3.2 The laboratory analysis revealed the progressive cracking was complex and of mixed modes in the initiation and early growth stages. The latter stages appeared to be essentially all due to fatigue cracking, driven by cyclic stresses generated in flight. The failed fitting exhibited a manufacturing deficiency in the form of gas bubbles located on the grain boundaries. Much of the early cracking appeared to be strongly influenced by the presence of these bubbles.

3.3 It was not possible to precisely evaluate the mode or rate of crack propagation since insufficient basic data exists to determine the adverse influence of the gas bubbles on the performance of such flawed material.

3.4 The fitting met all relevant specifications. It is not considered any typical quality control procedure would have detected the gas bubble deficiency. The available evidence suggests that only some fittings in a relatively small production batch of P/N 50-110057 fittings are defective in this manner, and that cracking similar to that in C-FCAS only occurs in this batch of fittings.

- 3.5 The evidence indicates that the fitting was inspected in accordance with Airworthiness Directive 70-25-04 and Service Instruction 0394-018 at 4,907 hours, and that no crack was detected at that time.
- 3.6 Since the accident the FAA and the manufacturer have affirmed that the dye penetrant inspection should have been made at 5407 hours on the basis of a 500 hour inspection interval. This inspection was not made - if it had been made it is most probable that a crack would have been detected.
- 3.7 There is a lack of clarity in AD 70-25-04 that resulted in the DOT and other maintenance personnel interpreting the AD as meaning that a 500 hour dye penetrant inspection was not mandatory if wing skin cracks were not found at the regular 100 hour skin inspections, which were carried out.

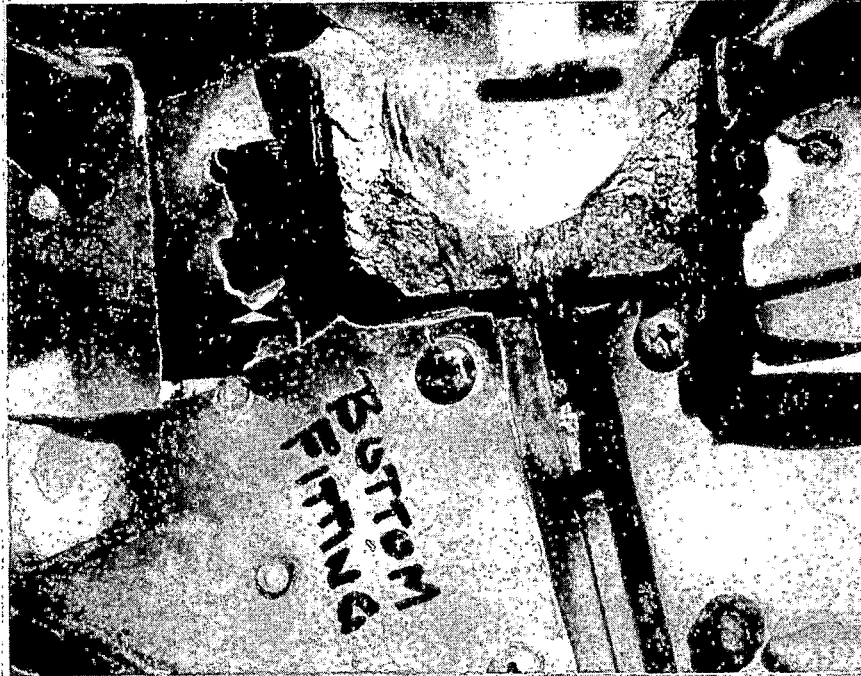


PHOTO 1 - IN-SITU VIEW OF THE OUTBOARD REMNANT OF THE FAILED LOWER FORWARD OUTBOARD WING ATTACHMENT FITTING STILL IN THE SEPARATED WING PANEL.



PHOTO 2 - CLOSE-UP OF PHOTO 1. THE DARKER AREA OF THE FRACTURE FACE DEFINES THE EXTENT OF THE PRECRACKED ZONE.

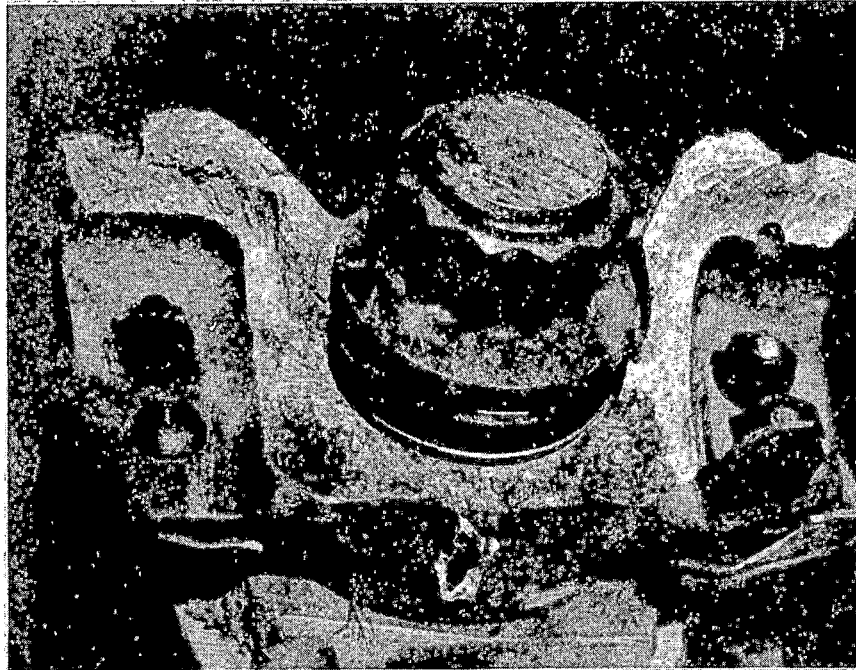


PHOTO 3 - CLOSE-UP OF THE INBOARD REMNANT OF THE FAILED FITTING, SHOWING ALSO THE INTACT BOLT, NUT & WASHER PACK.

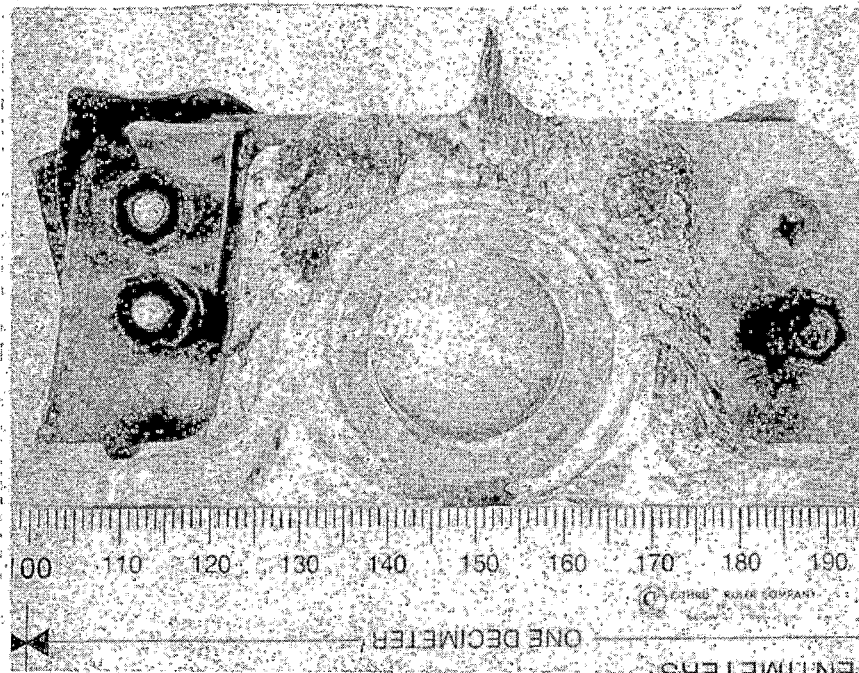


PHOTO 4 - THE FAILED FITTING REMNANT SHOWN IN PHOTO 3, REMOVED AND CLEANED FOR SUBSEQUENT LABORATORY EXAMINATION.

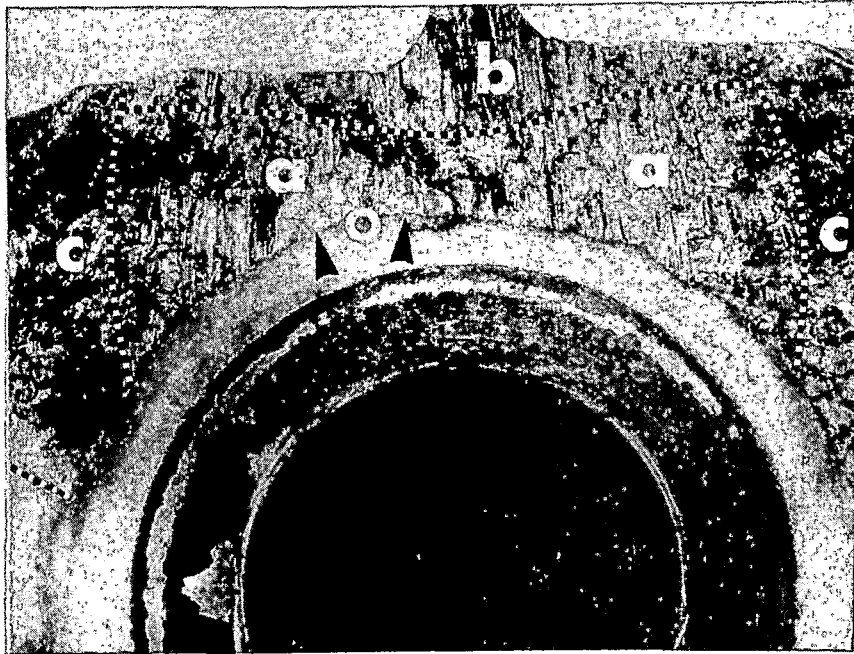


PHOTO 5 - AN OPTICAL CLOSE-UP OF PHOTO 4. THE PRECRACK ORIGIN IS AT "0". ZONES "a", "b", & "c" REPRESENT INTERGRANULAR PLUS FATIGUE, INTERGRANULAR, & FATIGUE CRACK GROWTH RESPECTIVELY.

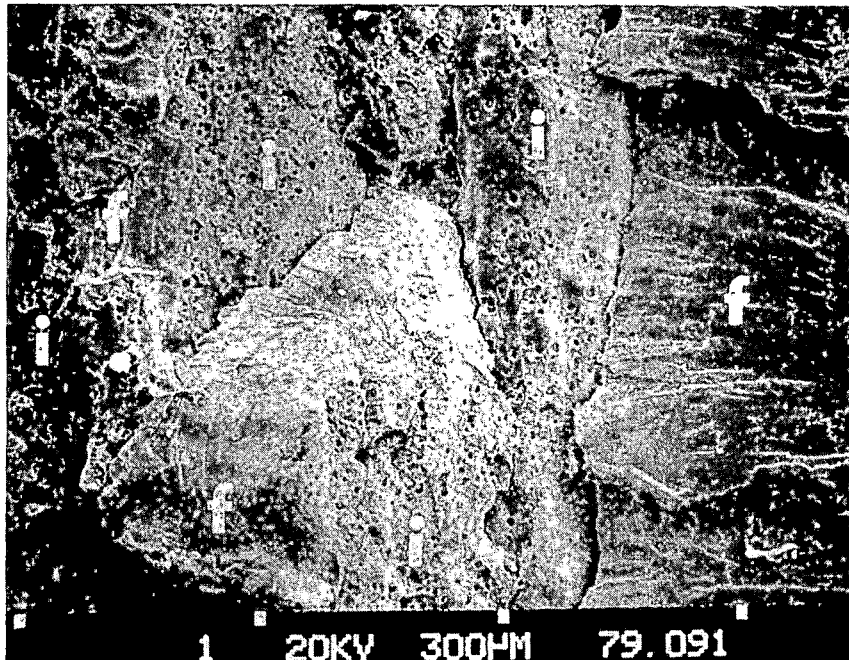


PHOTO 6 - S.E.M. IMAGE OF ZONE "a"(PHOTO 5) A FEW mm. FROM "0". AREAS "f" & "i" ARE FATIGUE & INTERGRANULAR CRACKING RESPECTIVELY. CRACK GROWTH IS RIGHT TO LEFT. x1,00 approx.

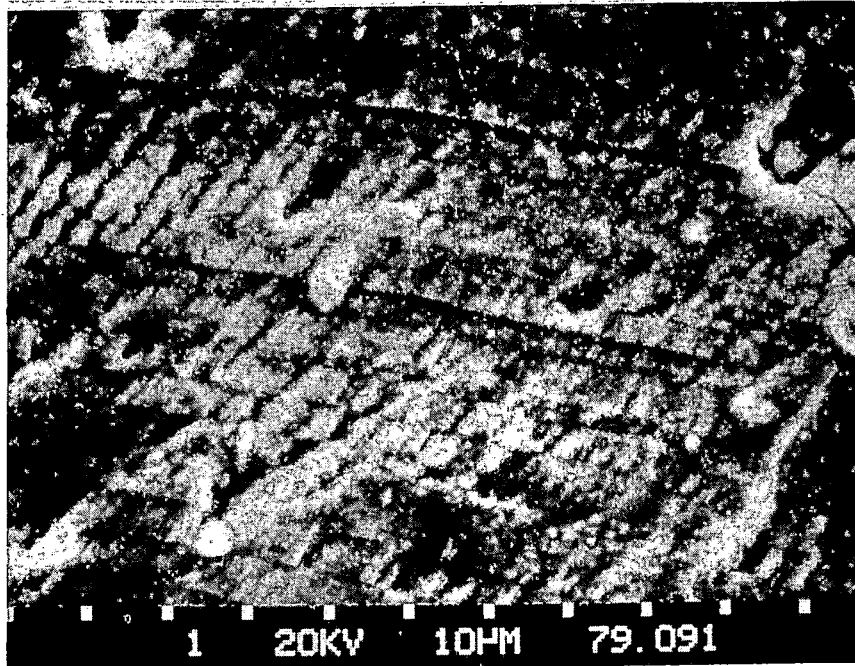


PHOTO 7 - S.E.M. IMAGE OF ZONE "c"(PHOTO 5) SHOWING TYPICAL RANDOM SPACING OF THE FATIGUE CRACK ARREST LINES. x1,000 approx.

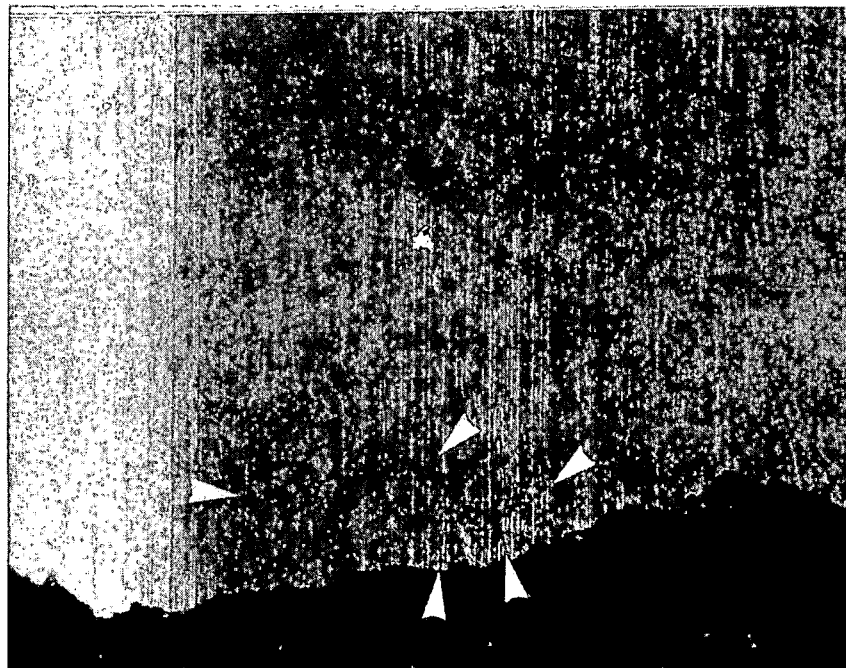


PHOTO 8 - SECONDARY INTERGRANULAR CRACKING (ARROWED) ON THE OUTER SURFACE OF THE FAILED FITTING CLOSE TO THE FRACTURE FACE.

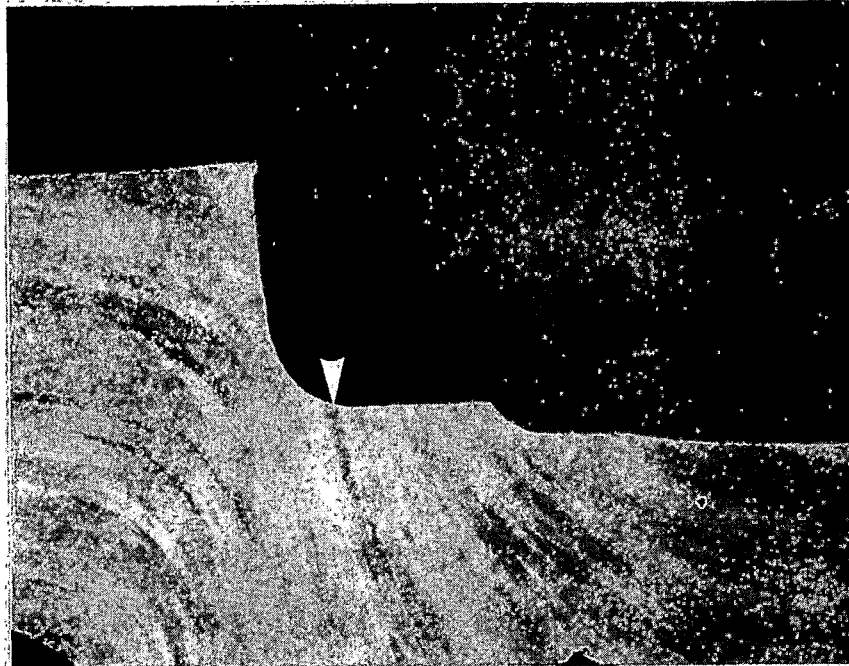


PHOTO 9 - AN ETCHED MACROSECTION CUT THROUGH THE UNBROKEN LEFT WING LOWER OUTBOARD ATTACHMENT FITTING SHOWING THE INTERNAL GRAIN FLOW. NOTE THAT THE GRAIN FLOW IS NEARLY PERPENDICULAR TO THE MACHINED NUT RECESS SURFACE AT THE ARROWED LOCATION. THIS CORRESPONDS TO THE LOCATION OF THE PRECRACK ORIGIN IN THE FAILED RIGHT WING FITTING.

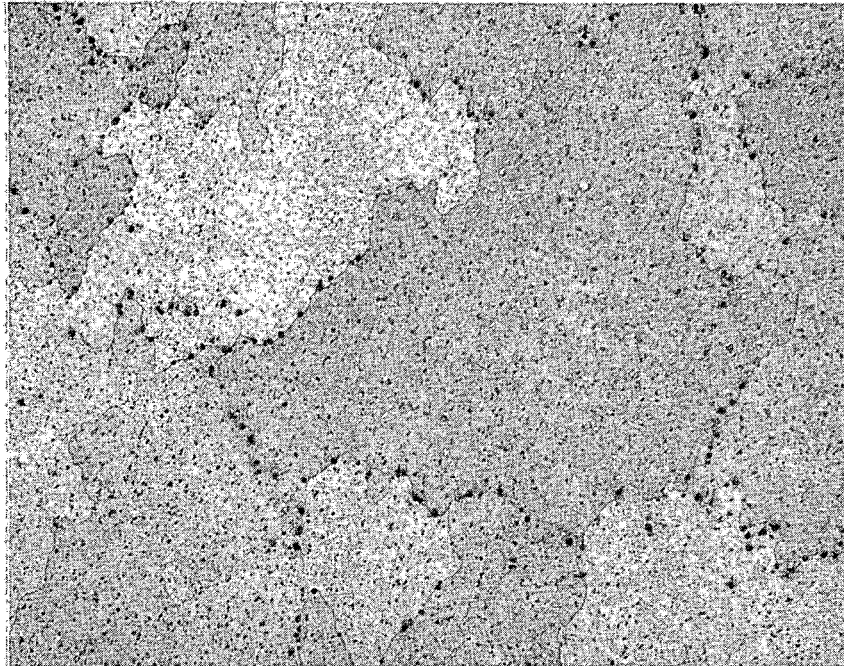


PHOTO 10 - AN ETCHED MICROSECTION, CUT FROM THE FAILED FITTING CLOSE TO AND PARALLEL TO THE FRACTURE FACE, AND PERPENDICULAR TO THE WING SKIN, SHOWING THE DISTRIBUTION OF THE SPHEROIDAL VOIDS ALONG THE GRAIN BOUNDARIES. (N.B. COMPARE SIZE AND DISTRIBUTION OF THESE VOIDS WITH THE SIZE AND DISTRIBUTION OF THE HEMISPHERICAL HOLES IN THE INTERGRANULAR FRACTURE FACE ZONES "i" OF PHOTO 6). X50, KELLERS ETCH.

Appendix B

70-25-04 BEECH: Amdt. 39-1121 as amended by Amendment 39-1332. Applies to Model 65-90 (Serial Numbers LJ-1 through LJ-67) airplanes with 5,000 or more hours' time in service.

Compliance: Required as indicated.

To detect any cracking of certain wing center section and other wing panel front spar carry through structural components, within the next 100 hours' time in service after December 5, 1970, unless already accomplished, accomplish the following:

A) Visually inspect the lower wing skin area adjacent to each outer wing panel front spar attachment fitting for cracks in accordance with Beechcraft Service Instruction No. 0394-018 or later revision approved by the Chief, Engineering and Manufacturing Branch, FAA, Central Region, and thereafter repeat the inspection at intervals not to exceed 100 hours' time in service.

(1) If wing panel skin cracks are found at the most outboard screw hole as noted in Figure 1 of Beechcraft Service Instruction No. 0394-018, the wing structure must be inspected in accordance with Paragraphs B and D of this AD prior to returning the aircraft to service. In addition, if a skin crack is found at the most outboard screw hole, as indicated in said Figure 1, the Paragraph B inspections must be performed at intervals of not more than 300 hours, and the Paragraph D inspections must be performed at intervals of not more than 500 hours.

B) Inspect by visual and dye penetrant methods, the right and left lower forward outboard wing attachment fittings for cracks in accordance with Beechcraft Service Instruction No. 0394-018 or later revision approved by the Chief, Engineering and Manufacturing Branch, FAA, Central Region, and thereafter repeat the inspection at intervals not to exceed 500 hours' time in service, except as noted in Paragraph A(1).

C) If fatigue cracks are found in either wing attach fitting during the inspections required by Paragraph B, prior to further flight -

(1) Both right and left outer wing panel lower forward spar caps including the wing attachment fittings and the skin panels adjacent to the outer panel wing attachment fittings must be replaced with new parts in accordance with the procedures, limitations and reinspection intervals specified in Beechcraft Service Instruction No. 0394-018, or later revision approved by the FAA.

(2) After one replacement of the assemblies listed in Paragraph C(1), replace the wing center section lower forward spar cap and fittings with new parts in accordance with Beechcraft Service Instruction No. 0394-018, or later revision approved by the Chief, Engineering and Manufacturing Branch, FAA, Central Region.

D) Inspect the structural components set forth below and in Beechcraft Service Instruction No. 0394-018, or later revision approved by the Chief, Engineering and Manufacturing Branch, FAA, Central Region, by visual, eddy current and dye penetrant methods, as specified in the Service Instructions, and thereafter repeat the inspection at intervals not to exceed 1,000 hours' time in service, except as noted in Paragraphs (A)1 and D(8). The structural components to be inspected are as follows:

(1) Right and left lower forward wing attachment fitting of the center section main spar.

(2) Lower forward wing fitting-to-spar attachment area and the edges of the forward and aft flanges on the lower forward spar cap in the center section (outboard of each main gear wheel well).

(3) Lower forward spar cap in each main gear wheel well.

(4) Lower surface of the lower forward spar cap in the nacelle inboard of each main gear wheel well.

(5) Lower surface of the lower forward spar cap between each nacelle and the fuselage.

(6) The four universal rivet holes in the forward flange of the lower forward spar cap inboard of each nacelle in the area of the wing root rib.

(7) The lower forward spar cap within the fuselage.

(8) The centerline skin splice in the area between the forward and aft center section spars, and the fuselage formers along the centerline between the forward and aft center section spars. If fuselage former cracks are found in this area, they must be repaired in accordance with said Service Instruction prior to return to service and the 1,000 hour inspection interval must be reduced to not more than 500 hours.

E) If cracks are found during the inspections required by Paragraph D, except cracks located in the areas specified in Paragraph D(8), prior to further flight, replace the wing center section lower forward spar cap, both the right and left outer wing panel lower forward spar caps including the wing attachment fittings and the skin panels adjacent to the outer panel wing attachment fittings with new production parts in accordance with Beechcraft Service Instruction No. 0394-018, or later revision approved by the Chief, Engineering and Manufacturing Branch, FAA, Central Region.

F) Aircraft logbook entries must be made and notification in writing must be sent to Chief, Engineering and Manufacturing Branch, FAA, Central Region, of the location and length of any cracks found during inspections required by this AD and also the total time in service of the component at the time the crack was discovered. Malfunction or Defect Report, FAA Form 8330-2, may be used for this purpose. (Report approved by the Bureau of the Budget under BC No. 04-R0174.)

G) Replacement of parts required by Paragraphs C and E will permit the owner/operator to establish new initial inspection times in compliance with Paragraphs A and B.

H) Equivalent methods of compliance with this AD must be approved by the Chief, Engineering and Manufacturing Branch, FAA, Central Region.

NOTE: The eddy current inspections required by this AD should be performed by certificated personnel trained and qualified in the operation of eddy current equipment. The replacement of critical parts such as the spar caps and wing attach fittings required by this AD should be performed by certificated personnel or facilities properly equipped to perform such repairs.

Amendment 39-1121 became effective December 5, 1970.

This amendment 39-1332 becomes effective November 16, 1971.

Beechcraft SERVICE INSTRUCTIONS

90

No. 0394-018
ATA Code 53-10

Kit No. 90-4063 S

SUBJECT: AIRFRAME, INSPECTION OF BASIC STRUCTURAL COMPONENTS

AIRCRAFT AFFECTED: BEECHCRAFT King Air 65-90, serials LJ-1 through LJ-67.

REASON FOR INSPECTION: To inspect certain basic structural components as required by recent FAA requirements. The criteria herein are based on laboratory testing by Beech Aircraft Corporation. In-service aircraft have not exhibited any fatigue cracks in spar components.

ACCOMPLISHMENT: At the initial times and subsequent intervals specified in a forthcoming FAA Airworthiness Directive or subsequent amendments dealing with the same subject.

DESCRIPTION: On the BEECHCRAFT King Airs, serials noted above, certain basic structural components should be inspected as follows:

WING SKIN CRACK INSPECTION

The skin area adjacent to each lower forward outerpanel wing attachment fitting should be visually inspected for cracks (see figure 1). Cracks in this area are only considered to be indicators. Their repair has no structural significance, and would not change the inspection intervals. The location and length of any cracks in these areas and the total aircraft hours at the time the cracks are detected should be recorded in the aircraft log book and the information forwarded to Parts and Service Operations, Beech Aircraft Corporation.

NOTE

It is not necessary to remove paint and/or primer which has cured for at least 30 days from areas to be inspected by dye penetrant procedures unless a crack is indicated or if the finish absorbs the penetrant so that bleed out prevents satisfactory inspection. If either condition exists, the paint and/or primer in these areas should be removed and the area reinspected using normal dye penetrant inspection procedures. Use fine emery tape or other dry abrasives to remove the paint and/or primer so that a possible crack will not be filled. Areas which still appear questionable should be examined visually under magnification up to 50 diameters or as required to determine the nature of the indication.

WING ATTACHMENT FITTING INSPECTION

1. Inspect the RH and LH lower forward inboard and outboard wing attachment

No BECP

1170

THE OPERATION, CARE AND MAINTENANCE OF AN AIRPLANE IS THE OWNER'S RESPONSIBILITY. A COMPETENT, LICENSED, BEECH AIRCRAFT CORPORATION ISSUES SERVICE INSTRUCTIONS RECOMMENDING MODIFICATIONS AND OPERATIONAL PROCEDURES TO ENSURE THE SAFETY OF THE MAXIMUM UTILITY AND SAFETY FROM HIS AIRPLANE. THESE INSTRUCTIONS ARE AVAILABLE TO BEECHCRAFT OWNERS THROUGH THEIR DISTRIBUTORS OR THROUGH THE BEECHCRAFT SALES OFFICES. WHEN INSTRUCTIONS ARE ISSUED, IMMEDIATE RECORDING OF THE PUBLICATION SHOULD BE MADE IN THE APPROPRIATE LOG BOOK.

Service Instructions No. 0394-018

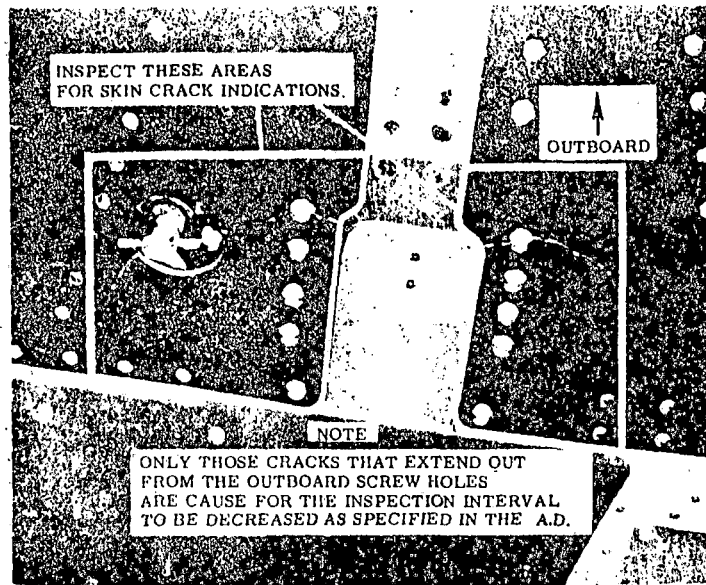


Figure 1. Skin Cracks

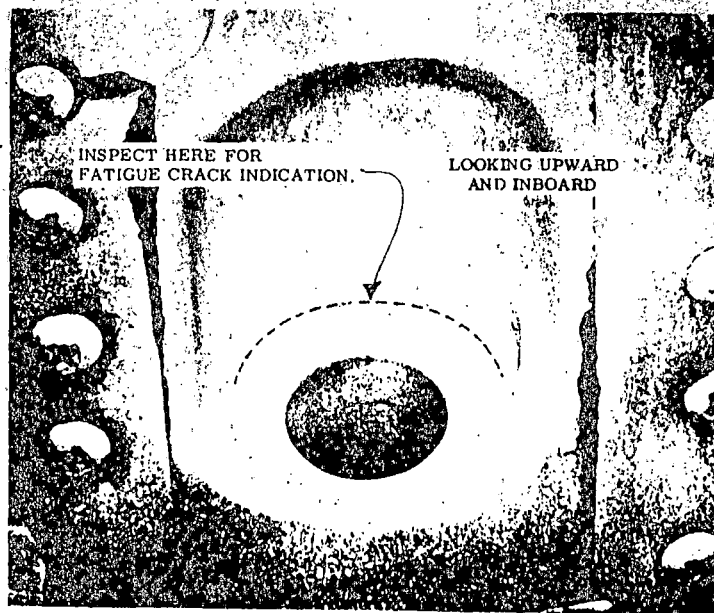


Figure 2. Wing Fitting Fatigue Crack

Service Instructions No. 0394-018

fittings using dye penetrant procedures as outlined in FAA Manual AC43.13-1, "Aircraft Inspection and Repair".

Remove the wing bolt cover plate. Remove the lower forward wing bolt and inspect the entire surface of the bathtub depression in each fitting using dye penetrant procedures. Check carefully for crack indications in the bolt counterbore areas shown in Figure 2.

A fatigue crack in either of the outboard lower forward wing attachment fittings is cause to replace both of the lower outboard fittings and lower outer panel spar caps. When fittings and spar caps are replaced, the skin panels adjacent to the lower forward outer wing attachment fitting must also be replaced with new production skin panels of the same part number. Refer to the note on page 6 for information pertaining to cracks in the inboard attach (center section) fittings.

NOTE

Cracks which may be found running in a spanwise direction on the lower face of the fittings boss (see Figure 3) are stress corrosion cracks, not related to fatigue, and are cause for replacement of the affected fitting only.

2. Reinstall the lower forward wing bolts. Refer to the Shop Manual Wing Installation Illustration for wing bolt installation procedures.
3. Reinstall the wing bolt covers and all access plates.

CENTER SECTION SPAR INSPECTION

CAUTION

All Huck bolt and rivet removal operations required to perform the inspections in the following steps should be accomplished very carefully to prevent damage to the spar cap.

1. Inspect the lower forward wing fitting-to-spar attachment and the edges of the forward and aft flanges on the lower forward spar cap in the center section, outboard of each main gear wheel well]. Access for the inspection is gained through the opening on top of the center section, adjacent to the forward spar and through the inverter cooling inlet, if installed, on the bottom of the center section. If the inlet is not installed, cut a new access hole per the kit drawing. Remove the aft inboard-most Huck bolt from the fitting-to-spar attachment area per the kit drawing.

NOTE

The Huck bolt may be removed by cutting the aluminum collar with a suitable tool and carefully driving the bolt out. If a chisel is used to split the Huck bolt collar, a modified AN960-816L washer (see the kit drawing) may be slipped over the collar to protect the adjacent structure. The chisel should be shaped and used as shown on the kit drawing. After removing the collar the Huck bolt should be inspected for damage and any damage that may exist should be smoothed before driving out the Huck bolt. This will prevent scratches within the hole which could cause a false crack indication.

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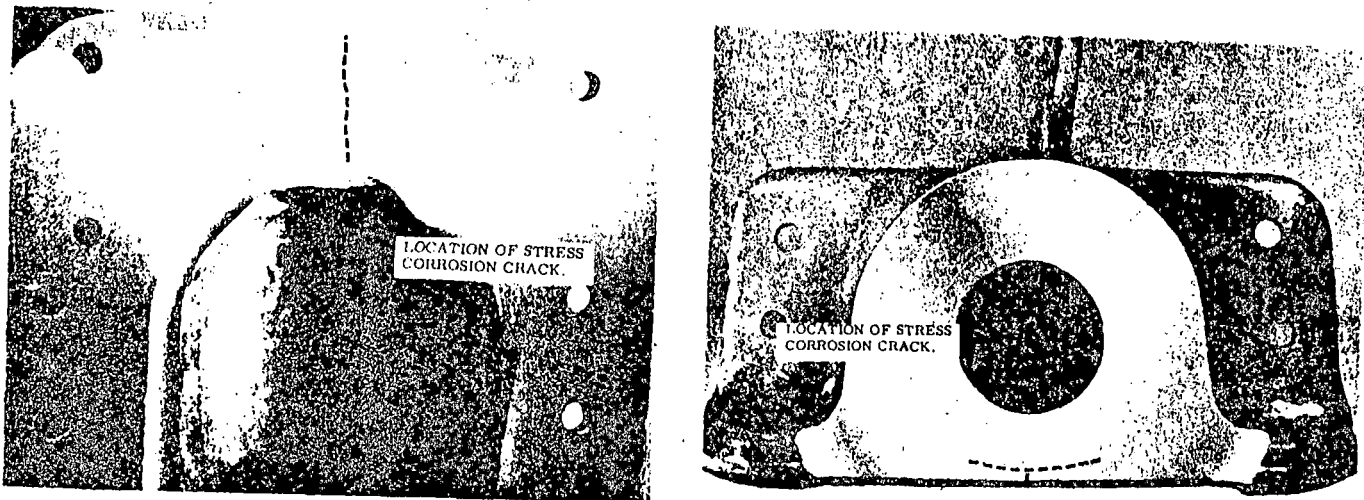


Figure 3. Wing Fitting Stress Corrosion Cracks

DYE PENETRANT INSPECT
SURFACE IN THESE AREAS

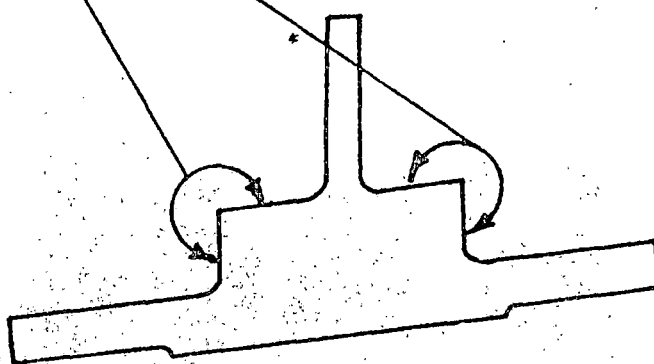


Figure 4. Spar Cap Cross Section

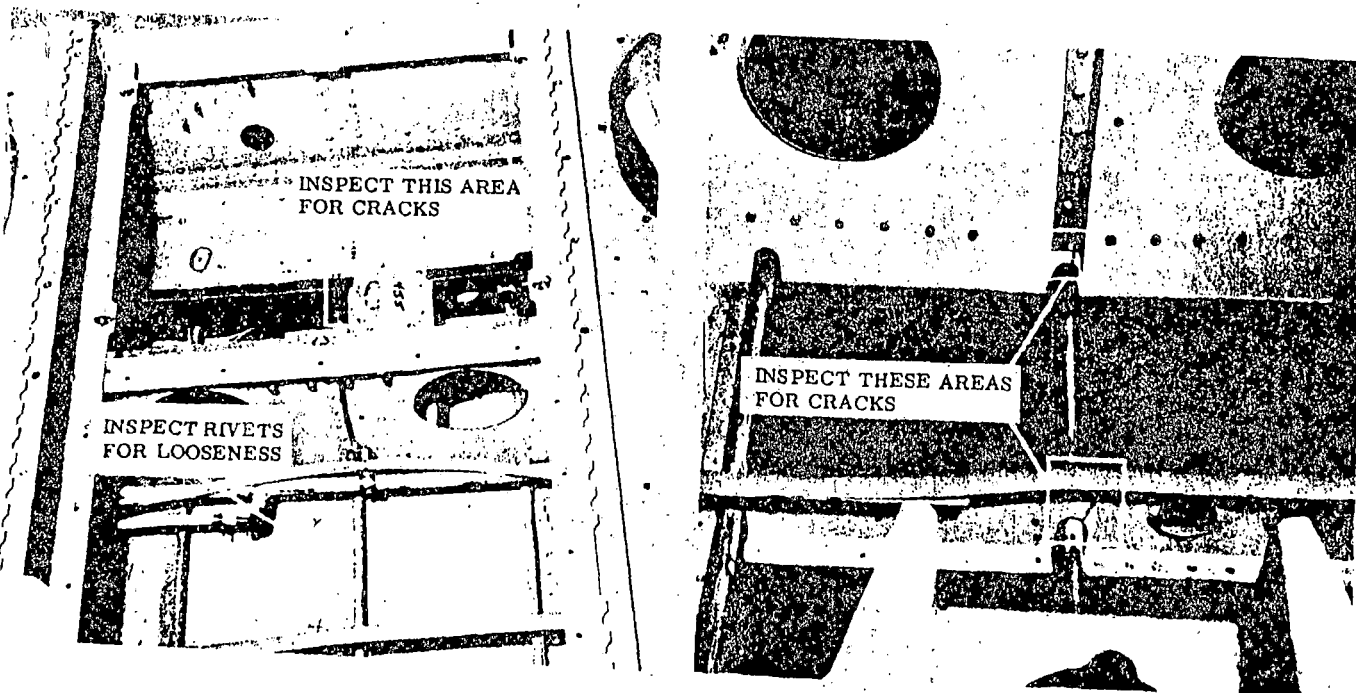


Figure 5. Cracks in Lower Fuselage

Service Instructions No. 0394-018

Use a Magnatest ED520, (P/N of Magnaflux Corporation), or equivalent eddy current tester with a 3/16 inch diameter eddy current probe (inserted in the hole where the Huck bolt was removed) to inspect the wing fitting-to-spar attachment area. Insert the probe into the hole in 1/16 inch increments. At each stop, rotate the probe 360°. (This will apply for all eddy current inspections called for in these service instructions.) Only personnel qualified to operate an eddy current tester should accomplish any of the eddy current inspections. A list of known available operators is given at the end of these service instructions. Particular attention should be placed on the interior of the spar cap near the surface that joins the wing fitting. Use dye penetrant procedures as outlined in AC43.13-1 to inspect the edges of the forward and aft flanges on the lower spar cap in this area.

2. Inspect the lower forward spar cap in each main gear wheel well. Carefully remove the two Huck bolts from the aft flange of the spar cap where the drag leg support is attached to the spar. Remove the 3/16 inch rivet from the aft flange on the spar cap at B.L. 88 (see the kit drawing). Inspect the area adjacent to the holes with an eddy current tester, (reference step 1.) Remove the three bolts immediately forward of the two Huck bolts that were removed and use eddy current and dye penetrant procedures as outlined in AC43.13-1 to inspect the spar cap in this area.

NOTE

Do not run the drill through the spar cap. When the rivets are being removed, drill through the rivet head and carefully drive the rivet out of the spar cap.

3. Inspect the lower forward spar cap in the nacelle inboard of each main gear wheel well.

Access to the lower surface is gained by cutting an access opening in the lower nacelle, just below and aft of the spar cap as shown on the kit drawing. Access to the upper surface of the forward flange is gained by removing the access door in the upper wing skin, inboard of the nacelle. Use dye penetrant procedures as outlined in AC43.13-1 to inspect the forward and aft flanges of the lower spar cap in this area.

4. Inspect the exposed lower surface of the lower forward spar cap between each nacelle and the fuselage for cracks. Use dye penetrant procedures as outlined in AC43.13-1 to accomplish this inspection.

5. Inspect the holes where the four Jo-bolts are installed in the forward flange of the lower forward spar cap inboard of each nacelle in the area of the wing root rib.

Access to the area is gained by removing the large rectangular access panel inboard of the nacelle and by cutting a round access opening inboard of the root rib. (Reference the kit drawing for location of the access opening.) Remove the four Jo-bolts and inspect the holes with a 1/4 inch eddy current probe and the eddy current tester described in step 1. Inspect the upper surface of the forward spar cap flange accessible through the round opening using dye penetrant procedures.

6. Inspect the lower forward spar cap within the fuselage. Access to the spar cap is gained by removing the forward cabin seats and the floorboard panels immediately forward and aft of the forward spar. Use dye penetrant procedures as outlined in AC43.13-1 to inspect the forward shoulder areas (see figure 4) of the

Service Instructions No. 0394-018

spar cap in the immediate vicinity of the aircraft center line and the aft shoulder
from outboard of the second stringer to the left and right of the aircraft
center line to the pressure skin on both sides.

NOTE

A fatigue crack in any section of the lower center section
spar cap or fitting is cause for replacement of the center
section lower spar cap and both outerpanel lower spar caps
including the wing attachment fittings.

- 7. At the aircraft centerline, in the area between the forward and aft center section spars, visually inspect the center line skin splice for loose rivets and cracks and visually inspect the fuselage formers along the center line for cracks (see Figure 5). If loose rivets or cracks are detected, repair using standard procedures as outlined in AC43.13-1 and record the total airframe time and the location and extent of the cracks in the aircraft log book and forward the information to Parts and Service Operations, Beech Aircraft Corporation.
- 8. Reinstall the floorboards, access plates and seats removed in step "6".
- 9. Install the doubler and the access plate on the lower inboard nacelle as shown in the kit drawing (reference step "3").
- 10. Install bolts, washers and nuts in place of the two Huck bolts and the 3/16 inch rivets removed from each wheel well in step "3" as shown in the kit drawing. Torque to 15 to 25 inch-pounds.
- 11. Ream the holes and install screws, washers and nuts (see the kit drawing) in place of the Jo-bolts removed in step "5".
- 12. Install the access cover for the opening cut in step "5". (See the kit drawing.)
- 13. Install a bolt, washer, and nut in place of the Huck bolt removed from the wing fitting-to-spar attachment area in step "1" as shown in the kit drawing. Torque to 15 to 25 inch-pounds.
- 14. Install the access cover for the opening cut in step "1". (See the kit drawing.)

Fatigue tests on complete wings are still in progress at Beech Aircraft Corporation, the results of which, combined with information gained from the inspections, performed under these service instructions, will be used in negotiations with the FAA for increased inspection intervals.

Kit 90-4063 S contains the parts and instructions which will be required to accomplish the complete inspection the first time. Subsequent complete inspections or inspections of only the wing attachment fittings will require only wing bolt washers as follows:

Part Number	Description	Quantity
01475-14-43.5	Torque Indicating Washer Assembly (for wing bolt)	2

Service Instructions No. 0394-018

Upon completion of these service instructions, make an appropriate statement in the aircraft log book, specifying the kit identification number and kit serial number (the kit serial number appears on the pink card contained in each kit).

THE FOLLOWING INFORMATION IS FOR PLANNING PURPOSES ONLY.

Estimated man-hours to accomplish first inspection: 36 hours
Suggested number of men to accomplish first inspection: 3 men
Estimated man-hours to accomplish subsequent inspections: 12 hours
Suggested number of men to accomplish subsequent inspection: 2 men
Suggested selling price of kit: To be advised

Since most aircraft operators will not have immediate access to personnel who are qualified in the operation of eddy current equipment, the following list of locations where such inspections may be contracted is presented. Contracts may be expected to involve expenses, travel costs, and an hourly charge for performing the inspection. The aircraft operator should have the various bolts and rivets removed, and access provided at the time of the scheduled inspection.

It is understood that the following organizations are prepared to send equipment and personnel from the Chicago office in support of inspections scheduled through the office closest to the operator's location.

Magnaflux Testing Laboratories, Division of Magnaflux Corporation:

Chicago, Ill.	Phone 312-384-1561
Cincinnati, Ohio	513-771-3292
Cleveland, Ohio	216-271-1900
Decatur, Ill.	404-241-0504
Detroit, Mich.	313-542-4515
Hartford, Conn.	203-522-3253
Houston, Tex.	713-923-9169
Little Ferry, N. J.	201-487-8600
Los Angeles, Calif.	213-685-6001
Philadelphia, Pa.	215-329-4070
Pittsburgh, Pa.	412-923-1722
San Diego, Calif.	714-279-5892

Automation Industries, Inc., CONAM Inspections subsidiary:

Chicago, Ill.	Phone 312-671-1672
Columbus, Ohio	614-491-3000
Gadsden, Ala.	205-547-2633
Gardena, Calif.	213-770-1740
Houston, Tex.	713-774-9657
Kenai, Alaska	907-283-7823
Marrero, La.	504-347-5393
Odessa, Tex.	915-337-5288
Pittsburgh, Pa.	412-731-7370
Portland, Ore.	503-246-8861
Rahway, N. J.	201-381-0050
Richmond, Calif.	415-233-9668
Tulsa, Okla.	800-331-3070

Dealers, distributors, and aircraft operators who wish to have their own personnel qualified for operation of eddy current equipment should contact Parts and Service Operations, Beech Aircraft Corporation, for information regarding the establishment of a training program. The offices listed above may also be contacted with respect to purchase of equipment.

Taken from LTR ST 1189 - Flight Loads on DOT King Air and Gulfstream II Navaid - Calibration Aircraft.

7.0 CONCLUSIONS

1. A total of 579 hours of valid accelerometer data were obtained from the five calibration aircraft operated by DoT. Fatigue-life estimates were made for several of the roles flown. The life-estimates are for a component on the lower wing, outboard of the main landing gear, and include the effects of the ground-air-ground cycle.

2. For all five aircraft, the low-level ILS flying is the harshest for all the roles flown.

3. Compared to the high-level enroute flying done by the Edmonton A100 King Air, the damage rate for the ILS flying of the King Airs is 5.6 to 13 times as harsh, depending on the aircraft. The mean ILS spectrum for all the King Airs is 12 times as severe. These relative damage rates are only approximate however, since there was only one non-zero data point on the negative side of the Enroute/HIGH spectrum, and a slope had to be assumed for the negative spectrum.

4. For the mean ILS spectrum, 50% of the fatigue damage occurs for acceleration peaks below 1.34 g, i.e., for accelerations of less than 0.34 g from level flight. These are relatively small accelerations, and are probably due to a combination of small gusts and the frequent corrections required in order to stay on the glideslope.

5. Low-level (below 2500 ft.) VOR flying by the Edmonton A100 King Air had a damage rate 1.5 times that of high-level enroute flying. Intermediate-level (2500 ft. to 10,000 ft.) VOR flying was 2.3 times as severe as high-level enroute flying.

6. Compared to some loads measured by the Australians on pressurized light twins in normal operations, the ILS flying by the King Airs varied from 2.6 to 6 times as severe. The mean ILS spectrum was 5.3 times as harsh as the Australian spectrum.

7. For the Gulfstream II, the low-level ILS flying had a damage rate 4.8 times higher than for enroute flying.

above 10,000 ft. The VOR flying was 1.6 times as harsh, while training and test flying combined was 2.3 times as severe as the enroute flying.

8. The Gulfstream II data were compared to some loads measured by the Australians on executive jet aircraft. The Gulfstream II ILS spectrum had a damage rate 1.7 times as severe as the Australian spectrum. All other flying by the Gulfstream II was less damaging than the loads for the Australian executive jets.

9. In calculating airframe (not powerplant) inspection intervals and retirement times for the King Air and Gulfstream II aircraft involved in nonaid-calibration flying, it is suggested that flying time spent on such missions would have to be multiplied by the following factors, in order to correct for the increased rate of fatigue damage:

AIRCRAFT	ROLE	MULTIPLICATION FACTOR
A90 and A100 King Air	ILS	12
	VOR	2
Gulfstream II	ILS	5
	VOR	2
	TRAINING	2

10. Since some of the data presented in this report are based on a study of limited duration, and are of unknown statistical validity, a long-term study is needed if the factors in item 9 above are regarded as severely conservative.

11. Consideration should be given to the possibilities of, and the need for, a continuous fatigue load-monitoring program for aircraft used in these and similar non-conventional roles.

12. Comments on the significance of the loads data reported herein should be obtained from the aircraft manufacturers, who, in the absence of adequate fatigue data, should be requested to provide all the quantitative information available, both from analysis and from the service experience of other operators, on a continuing basis.

Appendix E

REFERENCES

1. Transport Canada Report LP 91/79, Amended November 1980.
2. Battelle Columbus Laboratories Report to FAA, 24 July 1980.
3. Technical literature search via Canada Institute for Scientific & Technical Information October 1980.
4. Reliability of nondestructive inspections.
Final report, Dec 1978, SA-ALC/MME 76-6-38-1.
Prepared by the Lockheed-Georgia Co.