Department of Trade
Accidents Investigation Branch
Kingsgate House
66-74 Victoria Street
London SW1E 6SJ

8 February 1979

The Rt Honourable John Smith MP
Secretary of State for Trade

Sir

I have the honour to submit the report by Mr G C Wilkinson, an Inspector of Accidents, on the circumstances of the accident to a Boeing 707 321C G-BEBP which occurred near Lusaka International Airport, Zambia, on 14 May 1977.

I have the honour to be
Sir
Your obedient Servant

W H Tench
Chief Inspector of Accidents

Accidents Investigation Branch
Aircraft Accident Report No. 9/78
(EW/A267)

Operator:		Dan-Air Services Ltd
Aircraft:	*Type:*	Boeing 707
	Model:	321C
	Nationality:	United Kingdom
	Registration:	G-BEBP
Place of Accident:		Near Lusaka International Airport, Zambia.
		Latitude 15° 19′ 00″ S.
		Longitude 28° 23′ 20″ E.
Date and Time:		14 May 1977 at 0933 hrs
		All times on this report are GMT.

Synopsis

The accident was notified by Dan-Air Services Ltd to the Accidents Investigation Branch (AIB) at 1145 hrs on 14 May 1977. A United Kingdom (UK) Accredited Representative and a team of advisers participated in the initial stages of the investigations in Lusaka, after which the Zambian authorities elected to delegate the whole of the inquiry to the UK in accordance with paragraph 5.1 of Annex 13 to the ICAO convention.

The accident occurred during a manually flown approach, in daylight and in good weather. Shortly after the selection of landing flap the right horizontal stabilizer and elevator became detached causing the aircraft to pitch rapidly nose down and dive into the ground about two nautical miles short of the runway landing threshold.

The report concludes that the accident was caused by a loss of pitch control following the in flight separation of the right hand horizontal stabilizer and elevator as a result of a combination of metal fatigue and inadequate failsafe design in the rear spar structure. Shortcomings in design assessment, certification and inspection procedures were contributory factors.

1. Factual Information

1.1 History of the flight

The aircraft was engaged on a non-scheduled international cargo flight on behalf of International Aviation Services for Zambian Airlines. It was carrying a through load of palletised freight from London Heathrow to Lusaka International Airport, with intermediate stops at Athens and Nairobi, where there was a crew change. The flight from London to Nairobi was without incident and only minor aircraft unserviceabilities were recorded en route.

The aircraft took off from Nairobi for Lusaka at 0717 hrs with a fresh crew on board comprising a commander, co-pilot, two flight engineers (one under training) and a loadmaster. In addition there was one passenger on board, a ground service engineer whose duty was to supervise ground handling during transit stops.

The flight proceeded normally and apparently without incident at Flight Level (FL)310. At 0907:35 hrs the co-pilot contacted Lusaka Approach on 121.3 MHz and at 0911:38 hrs G-BEBP was cleared to descend to FL 110 at the LW Non-Directional Beacon (NDB). At 0923 hrs the co-pilot reported that the aircraft was levelling at FL 110 at a Distance Measuring Equipment (DME) range of 37 nautical miles from Lusaka. G-BEBP was then cleared by Lusaka Approach to FL 70 in 1000 foot steps following about five minutes behind another aircraft also bound for Lusaka International Airport.

At 0928:53 hrs the co-pilot reported that the airfield was in sight. Lusaka then cleared the aircraft to descend to an altitude of 6,000 feet (2,221 feet above touchdown elevation) on the aerodrome QNH of 1,021 millibars. A minute later at 0929:55 hrs the co-pilot reported that G-BEBP was turning downwind with the preceding aircraft in sight ahead. The Lusaka Approach controller then gave the aircraft a clearance to make a visual approach to join on left base leg for runway 10 as number two on the circuit, and to report leaving 6,000 feet. Shortly afterwards the co-pilot reported that the aircraft had left 6,000 feet and he was instructed to change to the Lusaka Tower frequency of 118.1 MHz. At 0932:02 hrs the co-pilot contacted the Tower Controller and reported that G-BEBP was turning on base leg with an aircraft in sight on the runway. The Tower Controller then cleared G-BEBP to finals. The co-pilot replied 'roger'; this was the last transmission received from G-BEBP.

A readout of the Cockpit Voice Recorder (CVR) indicated that 50° flap was selected at 0932:53 hrs and that the landing checks were completed by 0933:11 hrs. Six seconds later at 0933:17 hrs a loud "break-up" noise was recorded with the record terminating 5 seconds later at 0933:22 hrs.

Eye witnesses on the ground observed G-BEBP established on what appeared to be a normal approach to runway 10 at Lusaka International Airport. They saw a large portion of aircraft structure separate in flight. The aircraft then pitched rapidly nose down and dived vertically into the ground from a height of about 800 feet and caught fire.

The accident was observed from the airfield; the fire and rescue services responded rapidly and were quickly at the scene of the accident. When the fire was under control it became apparent that the degree of damage to the cockpit structure was such that no one could have survived the impact forces. In fact, all six occupants were killed. However there were, fortuitously, no casualties to persons on the ground.

The complete righthand horizontal stabilizer and elevator assembly was found some 200 metres back along the flight path indicative of having become detached in flight prior to the final nose down pitch manoeuvre.

The accident occurred in daylight, in good weather conditions at approximately 0933 hrs at a point on the extended centre line of runway 10 at Lusaka International Airport 3,660 metres from the runway threshold at an elevation of 3,822 feet above mean sea level (amsl) ie 43 feet above runway touchdown elevation.

1.2 Injuries to persons

Injuries	*Crew*	*Passengers*	*Others*
Fatal	5	1	–
Non-fatal	–	–	–
Minor/None	–	–	–

1.3 Damage to aircraft

The aircraft was destroyed as a result of ground impact and fire damage.

1.4 Other damage

The aircraft crashed in an area of scrubland causing no other damage.

1.5 Personnel information

Flight crew.

1.5.1 Commander: Male age 34 years

Licence: Airline Transport Pilot's licence valid until 30 January 1978. A class 1 medical certificate was last issued on 23 March 1977 with no restrictions. His instrument rating was last renewed on 1 April 1977.

Aircraft ratings: Boeing 707/720., Boeing 727 and Comet 4.

Mandatory Checks: Initial (and last) Certificate of Test on Boeing 707 was dated 1 April 1977. Last route check was dated 25 April 1977 and the last safety equipment and procedures check was dated 4 February 1977.

Flying experience:
Total on all types: 6,782 hours
Total in command all types: 3,953 hours
Total on Boeing 707 (all in command): 115 hours
Total in last 28 days (all Boeing 707) : 73 hours.

Duty time: The commander had a rest period of 30 hours 30 minutes at Nairobi before coming on duty for the subject flight at 0640 hrs. At the time of the accident he had been on duty for 3 hours 3 minutes.

1.5.2 Co-pilot: Male age 57 years

Licence: Airline Transport Pilot's Licence valid until 24 July 1986. A class 1 medical certificate was issued on 17 December 1976 with a restriction that spectacles correcting for near vision were to be worn. His instrument rating was last renewed on 16 February 1977.

	Aircraft ratings: Boeing 707/720, Comet 4, Britannia and PA 30.
Mandatory checks:	Last certificate of Test on Boeing 707 was dated 4 September 1976. Last safety equipment and procedures check was dated 24 November 1976.
Flying experience:	Total on all types: 13,745 hours Total as co-pilot all types: 5,575 hours Total on Boeing 707 (all co-pilot): 2,482 hours Total in last 28 days (all Boeing 707): 58 hours.
Duty time:	The co-pilot had a rest period of 30 hours 30 minutes at Nairobi before coming on duty for the subject flight at 0630 hrs. At the time of the accident he had been on duty for 3 hours 3 minutes.

1.5.3 Supervisory Flight Engineer: Male aged 38 years

Licence:	Flight Engineers licence valid until 26 April 1978. Last medical certificate was issued on 20 April 1977 with no restrictions. Flight Engineer aircraft ratings: Boeing 707 and Comet 4.
Mandatory checks:	Last certificate of Test on Boeing 707 was dated 10 February 1977. Last route check was dated 26 September 1976 and the last safety equipment and procedures check was dated 24 September 1976.
Flying experience:	Total on all types: 6,103 hours Total on Boeing 707: 848 hours Total in last 28 days (all Boeing 707): 67 hours.
Duty time:	At the time of accident he had been on duty for 3 hours 3 minutes, after a rest period of 30 hours 30 minutes at Nairobi.

1.6 Aircraft Information – Boeing 707-321C G-BEBP

1.6.1 General Information

(i)	Manufacturer	The Boeing Company, Seattle, USA
(ii)	Date of Manufacture	1963
(iii)	Constructors Number	18579
(iv)	Registered Owner	Dan-Air Services Ltd registered 14 October 1976
(v)	Certificate of Airworthiness	United Kingdom Transport Category (Passenger) first issued 14 October 1976 valid until 13 October 1977.
(vi)	Certificate of Maintenance	21 April 1977 valid for 500 flying hours.
(vii)	Total airframe hours	47,621
(viii)	Total landings	16,723

(ix)	Hours since issue of UK C of A	1,649
(x)	Landings since issue of UK C of A	438
(xi)	Hours since last C check on 21 February 1977	662
(xii)	Landings since C check	176
(xiii)	Hours since last B check on 21 April 1977	175
(xiv)	Landings since last B check	50
(xv)	Estimated weight at time of accident	111,030 kg
(xvi)	Maximum permitted landing weight	112,039 kg
(xvii)	Estimated C of G at time of accident	19.95% MAC
(xviii)	Permitted C of G range at accident weight	19% – 35% MAC
(xix)	Estimated fuel remaining at time of accident	13,880 kg Jet A1

1.6.2 *Airframe history*

G-BEBP was the first aircraft off the 707-300C series convertible passenger/freighter production line.

Since manufacture it had been operated in the passenger carrying role registered as N765PA. After it was withdrawn from service in March 1976 it was put into storage in Florida. In June 1976 the aircraft was flown to the UK where it went through a modification and overhaul programme at the Dan-Air engineering facility prior to the issue of a USA Export C of A which was the basis for the issue of a United Kingdom C of A in the Transport category (Passenger) on 14 October 1976.

During service on the US register the aircraft had been maintained in accordance with an FAA approved schedule and, subsequent to its transfer onto the British register, it had been maintained to a UK CAA approved schedule. The records indicate that the aircraft had not been involved in any previous accident and there is no record of it having been involved in any incidents which might have affected the aircraft's structure. It has been established that both the horizontal stabilizers on the aircraft at the time of the accident were those fitted at the time of manufacture. Both left and right horizontal stabilizers were removed and reinstalled by Dan-Air to provide access to the stabilizer centre section and for minor refurbishment.

1.6.3 *Serviceability*

Consideration was given to reports that the aircraft pitch trim was unusual in its response on the previous flight. No evidence was found that could be related to these reports, which referred to an unusually sensitive stabilizer trim brake. Such behaviour could only be related to the stabilizer structural failure had there been stabilizer torsional deflections large enough to affect significantly the aircraft's flight characteristics. It is considered that such gross torsional deflections would have produced total failure at that time and the reported behaviour is not therefore considered relevant to the accident.

1.7 Meteorological information

The weather recorded at Lusaka International Airport at 0930 hrs was:

Wind:	280° 5 knots
Visibility:	30 kilometres – slight haze
Cloud:	3/8 cumulus base 600 feet
Temperature:	+28.6° C
QNH:	1,020 millibars
QFE:	888 millibars
Light:	Bright daylight

There was no record of any atmospheric turbulence.

1.8 Aids to navigation

Not relevant to this accident.

1.9 Communications

All contacts on RTF between the aircraft and ground stations were completely normal and there were no indications that the crew were aware of any problems affecting the safety of the aircraft. The last contact with the aircraft was when the co-pilot acknowledged clearance to finals with one word 'roger' approximately 1 minute before the accident.

1.10 Aerodrome information

Lusaka International Airport is located in open country to the North East of Lusaka. The single paved runway 10/28 is 3,962 metres long, 46 metres wide and is equipped with VASI and approach and runway lighting. The actual heading of runway 10 is 104° (M) and the landing threshold elevation is 3,779 feet amsl.

1.11 Flight Recorders

1.11.1 *Flight data recorder*

The aircraft was equipped with a Sunstrand engraved foil flight data recorder, type FB-542, mounted in the left hand side of the tailcone aft of the rear pressure bulkhead. The recorder was severely damaged by impact forces and the foil was torn completely in two over the scriber roller, with a small section torn from the central area. Examination of the foil record for the accident flight revealed that data for all the recorded parameters were present. The quality of the recording was generally good with the exception of the time base marks on one side of the foil being unintelligible, probably due to a worn scriber.

The following list of eight parameters were recorded against elapsed time with a nominal foil speed of 30.5 cm/hour (11.81 secs/mm):

(i) Indicated airspeed

(ii) Pressure altitude

(iii) Magnetic heading

(iv) Normal acceleration

(v) Pitch attitude

(vi) Roll attitude

(vii) Flap position

(viii) Engine pressure ratio

The most recent calibration checks had been carried out on the recorder on 5 October 1976. These data were used in refining the recorder readout. Damage to the recorder precluded any post accident calibration checks. The recorder was returned to the UK where a good quality readout was made and a series of plots obtained (see appendix B) related to the last 5 minutes of flight.

In order to assess whether a large static jump in the fatigue fracture (discussed later) was related to a high load manoeuvre or gust, a review was made covering some 1,800 flying hours, for which recorder foils were available, to determine if a normal acceleration level exceeding an increment of 1g had been achieved at any time. The scanning of the foils was carried out by the recorder manufacturer. The results of this review indicated that this g threshold was penetrated only once during a gust encounter in the cruise regime on 28.1.77 when the maximum values recorded were +1.9g and –0.2g.

1.11.2 *Cockpit voice recorder*

The aircraft was equipped with a Fairchild type A100 cockpit voice recorder (CVR) mounted in the rear fuselage ahead of the rear pressure bulkhead and using plastic tape as a recording medium. It operated at a nominal tape speed of 4.76 cm/second. The track allocation of the installation was as follows:

Track	Source
1	Commander's MIC/TELS audio
2	Hot microphones (Both pilots and the flight engineer)
3	Cockpit area microphone
4	Co-pilot's MIC/TELS audio

The recorder was extensively damaged by impact forces but on examination the tape was found to be undamaged. The CVR was returned to the UK where the record was transcribed and refined.

The two MIC/TELS tracks were found to be of good quality. The cockpit area microphone had recorded correctly but the high background and atmospheric noise reduced the clarity of low level speech; the situation was aggravated by the presence of five crew members on the flight deck. The hot microphone track provided a clear record for the co-pilot's position but not for any of the other crew members. In addition RTF traffic had been recorded on this track.

Transcription of the CVR record indicated that the crew were not aware of any handling problems nor was there any evidence of abnormal trim operation. The only crew comment related to aircraft or system unserviceability concerned the weather radar. In particular there was no comment by the crew concerning turbulence or adverse weather conditions other than a reference to haze obscuring the airfield as the aircraft approached FL70 in the descent. The descent, approach and landing check lists were carried out normally and the CVR record indicated that the commander was handling the aircraft with the co-pilot carrying out the RTF procedures.

At 0933:10 hrs between 25 and 26 seconds after the selection of 50° flap there was a loud noise lasting 0.5 seconds recorded, readily identified as an airborne structural failure. A further 0.5 seconds later a shout from a crew member suggested that the aircraft had started a violent manoeuvre. Ground impact and cessation of the CVR record occurred between 4.8 and 4.9 seconds after the noise of the structural failure.

A schematic flight profile of the last 31 minutes of flight derived from the transcript of the CVR record is at appendix B.

1.12 Examination of the Wreckage

1.12.1 Accident site

The wreckage of G-BEBP was confined to two relatively small and separate areas (see figure 1). The first area contained the complete right horizontal stabilizer together with some small fragments of structure from the rear fuselage adjacent to the empennage and from the inboard end of the right horizontal stabilizer.

The second area of wreckage was on the extended centreline of runway 10 at an elevation of 3,822 feet and some 200 metres down track from the forward edge of the first area and contained the whole of the remainder of the aircraft. This area had been subjected to intense post-impact fire which consumed a large part of the centre fuselage and affected the majority of the remaining wreckage. There was no evidence of fire prior to ground impact.

Examination of the detached stabilizer revealed evidence of a fatigue failure of the top chord of the rear spar at a point approximately 36 cm outboard (see figure 3) of the root attachment pin. The rear spar centre chord, lower chord and the front spar root attachments had failed in overload (see figure 4) due to stabilizer downwards bending. There was evidence of a pre-existing fracture of the rear spar upper web between the top chord (adjacent to the fracture) and the centre chord and in certain sections of the closure rib and associated structure.

Examination of the remaining wreckage indicated that the aircraft had struck the ground in a 50° dive laterally level. Distortion of the engine cowlings and compressor casings indicated that at the time of impact the aircraft pitch attitude was approximately − 100° relative to the horizontal. (See figure 2). The distribution of the main wreckage was consistent with the impact conditions described. There was sufficient evidence to establish that, with the exception of the right horizontal (separated) stabilizer and associated structure, the aircraft was complete at impact although it is likely that the number four engine support structure was beginning to break up just prior to impact.

In view of the facts already established, further examination of the wreckage on site was restricted to that necessary to establish the configuration, power and trim settings, and to establish the pre and post impact sequence in sufficient detail to be satisfied that there were no anomalies.

The aircraft struck the ground with 50° trailing edge flap and leading edge flaps fully extended, with the landing gear down. Engine power could not be accurately assessed in the field but the damage to each unit indicated a low to moderate power setting. It was later established that the spoilers were retracted at impact.

The stabilizer trim screw jack and associated cable drum were recovered from adjacent, but separate, areas of wreckage. Both units were found to be set at positions consistent with a stabilizer setting of 6¼ units aircraft nose up.

It was not possible to establish rudder and aileron trim settings although the cockpit rudder trim indicator was found at an approximately neutral setting. However, the impact attitude tended to rule out any significant directional or roll trim problems.

All structure which became separated in the air, together with the left horizontal stabilizer, stabilizer centre section, stabilizer jack screw and trim drum, and the power level consol were transported to UK for more detailed investigation.

1.12.2 *General examination*

The detailed investigation of the wreckage was confined primarily to the stabilizer and rear fuselage structure to establish (i) the reason for and age of the fatigue failure and (ii) to determine why the failsafe structure in the rear spar had failed to carry the flight loads once the top chord had fractured as a result of fatigue.

1.12.2.1 Metallurgical examination

(i) Top chord fracture

The primary fracture surfaces in the top chord were examined independently by metallurgists at the Royal Aircraft Establishment, Farnborough, and at the Boeing Company. The outboard fracture face (see figure 5) was in good condition with, generally, little corrosion or bruising. There was some degree of bruising however on the lower edge of the top chord and on the aft flange which indicated that the fracture had occurred some significant time before the failure of the spar structure as a whole.

The origin of the top chord fatigue crack was found to be sited at the upper edge of the 11th fastener hole in the forward flange of the chord. This region sustained a small amount of damage during impact with the ground which prevented the exact nature of the origin from being determined. A number of minor fatigue cracks were also observed in this region of the hole together with corrosion pits and layer corrosion. However none of these small fatigue cracks, which would have originated at about the same time as the failure crack and then stopped as the failure crack reached sufficient proportions to relieve the stresses in that area, were found to be associated with the corrosion. It is therefore probable that the failure crack was not associated with corrosion and that the corrosion occurred after the crack initiation period.

The primary major fatigue crack (see figures 6 and 7) was found to have propagated rearwards across the flange and into the main cross section of the chord. A secondary major fatigue crack originating in the forward side of the 11th fastener hole propagated across the flange in the same plane as the principal crack until it had reached the forward edge of the flange.

It was found that the primary major fatigue crack growth rate increased rapidly over the first 2mm of crack, after which the increase in growth rate was more gradual until, at about 7mm crack length, the growth rate settled at a reasonably constant value of approximately 125 flights per mm. This rate continued until the crack reached a point when it had progressed across approximately 60% of the exposed top surface of the chord. At this point the crack front made a static tensile jump of approximately 21mm (at its widest point) in one flight after which the growth rate settled back to its original value of

125 flights/mm. This large static jump was confined to the internal crack front only; the external crack length did not increase significantly at this time, (see figure 6). During the next 200 or so flights the growth rate remained basically stable but there were a number of smaller tensile jumps ahead. When the crack had progressed right across the top of the exposed chord surface and had reached approximately 60% of the depth of the main cross section there was another large jump and after a short period during which the crack progressed in the fatigue mode, there was a final tensile jump and the chord failed completely. The total number of flights between the initiation of the fatigue crack and the final failure of the upper chord is estimated as being in the order of 7,200 flights with some 3,500 flights of propagation occurring across the exposed top surface of the rear chord.

(ii) Remaining rear spar structure

The failure of the centre and lower chords and the lower web resulted from overload in stabilizer downwards bending following the fatigue of the top chord and subsequent static failure of the upper web. The fracture path through the rear spar structure is shown in figure 11a.

The upper web static tear fracture was confirmed as having occurred at a significant time interval before the total spar failure. Pre-existing fractures (relative to the final stabilizer failure) of certain pieces of closure rib and associated structure in the vicinity of the upper rear spar were also confirmed. Three small areas of fatigue were also found: one in each of the pair of centre chord top flange fastener holes just inboard of the top web fracture line, and another in the closure rib attachment lug on the forward side of the rear spar top chord.

A fourth zone of fatigue was found in the upper edge of the closure rib approximately 5 cm forward of the rear spar. This zone contained a pair of fatigue cracks on opposite sides of the cross section (measuring approximately 6 mm x 1 mm and 2 mm x ½ mm) which were produced by plate bending in the upper edge of the rib. There were also a number of similar but much smaller cracks in the same general area.

The stresses which gave rise to all of these secondary fatigue cracks would not have been present in the intact structure, and an attempt was made to determine the number of flights during which the cracks were propagating in order to establish the minimum time interval between the failure of the top chord and total spar failure. It was not possible to establish the number of flights with any degree of certainty because the random loads encountered during flight could be expected to be significant in terms of the growth rate of the secondary fatigue cracks. However, it is considered that there were probably up to 100 flights between top chord failure and stabilizer separation.

Test samples were cut from each element of the rear spar and subjected to a series of tests to check the material specification. In each case the material was found to be up to specification in all respects.

1.12.2.2 Inboard top chord forward flange and fasteners

Of the right hand stabilizer top chord forward flange fasteners inboard of the crack, only numbers 1 and 2 were recovered. The number 2 fastener was found to be a lower strength aluminium "cherry lock" fastener different to that originally fitted. The numbers 2, 3 and 4 fasteners from the corresponding area of the left stabilizer were also replacements of the same type as number 2 fastener on the right hand side.

There was evidence of fretting between the skin and forward flange on both horizontal stabilizers. The fretting on the right side was extremely heavy and is considered to have occurred, mainly, after top chord fracture.

Fastener head distortion which was consistent with high load transfer in a generally spanwise sense was found on fasteners 1 to 10 of the left stabilizer. During the post-accident fleet inspection a Boeing 707 series 436 aircraft was found to have a crack also at the eleventh fastener on the right hand horizontal stabilizer top chord forward flange. The fasteners inboard of the crack on this stabilizer were distorted in an identical manner to the distortion found on the left stabilizer of G-BEBP and fretting damage to the chord and top skin was also found, again identical to the damage found on the left stabilizer of G-BEBP.

The left stabilizer top chord of G-BEBP was examined for cracks using visual and eddy current techniques, but none were found.

Examination of the fastener holes in the top skin of the left stabilizer revealed a pattern of distortion that reflected the previously observed fastener distortion and fretting between the skin and flange. The axis of hole elongation was found to be variable. Generally the axes were spanwise, but the inboard group were at an angle to the chord spanwise axis. The holes in the corresponding skin from the right stabilizer were also found to have been distorted but with the elongation in two distinct directions; one set of axes being a mirror image of those on the left stabilizer skin and the other set being spanwise. The precisely spanwise elongation is considered to have occurred when the fasteners sheared following top chord failure and the other elongation pattern to have occurred as a result of a similar loading mechanism to that which produced the corresponding damage on the left stabilizer.

The skins from the 436 series cracked stabilizer were found to have fastener hole distortion similar to the left stabilizer of G-BEBP.

1.12.2.3 Detailed examination of empennage

The left and right horizontal stabilizers, centre section, stabilizer screw jack and rear cable drum assembly were examined in the UK. It was found that all the debris from the airborne separation wreckage zone came from the rear fuselage structure in the vicinity of the left stabilizer pivot and rear spar cut out, and from the right stabilizer closure rib and inboard leading edge structure.

Examination of the structure from the vicinity of the stabilizer mountings revealed a consistent pattern of distortion indicating that the left stabilizer had been wrenched tip downwards and had partially broken away from the centre section mountings in the rear fuselage. The stabilizer trim screw jack was bent and fractured in a manner which indicated that the damage was caused by a left stabilizer tip downward twisting of the centre section relative to the rear fuselage. The downwards movement of the left stabilizer rear spar resulted in a small section of rear fuselage skin and supporting structure adjacent to the rear spar cut-out (see figure 8) being broken free of the surrounding structure. Prior to its becoming detached, contact between the lower edge of the left stabilizer and this piece of skin had produced witness marks, which are discussed in later paragraphs.

Examination of the elevator control circuit revealed no evidence of pre-breakup failure or malfunction. It was established that at the time of right stabilizer separation the righthand elevator push-pull tube had been at a position equivalent to the application of maximum up elevator deflection. This was entirely consistent with the tension failure of the push-pull tube caused by the separation of the right stabilizer from the fuselage.

The stabilizer trim setting was established as 6¼ units aircraft nose up from the trim drum and trim screw jack. This figure was identical wtih that obtained from the preliminary examination conducted in the field. Witness marks on the rear fuselage skin adjacent to the stabilizer rear spar cut out, made by the edge of the lower skin of the left stabilizer, confirmed this trim setting. (See figure 9). This piece of fuselage skin was a part of the structure which became separated during the right stabilizer breakup sequence and therefore reflects the left stabilizer position at, and immediately after, right stabilizer separation.

A second witness mark on the same piece of skin indicated that at some time between stabilizer separation and final impact the stabilizer trim angle had moved to a position displaced + 25° from its initial value, established previously as being 6¼ trim units aircraft nose up. This is well outside the normal trim range and is consistent with a failure of the trim screw jack during the breakup sequence.

1.13 Medical and pathological information

Autopsy examinations were carried out on all six occupants of G-BEBP. No signs of any pre-existing disease or of a possible medical cause for the accident were found. However, the severity of the injuries, which were entirely consistent with an aircraft accident, precluded formal autopsy procedures and therefore no histological or toxicological examinations were possible. None of the bodies showed any sign of burning.

1.14 Fire

The accident was observed by the Lusaka International Airport station fire officer who immediately alerted the airport fire service. The accident site was some 3 kilometres distant from the fire station via an aerodrome perimeter crash gate. On arrival at the site an area of approximately 100 metres x 60 metres was burning fiercely. One large foam tender and one medium foam tender were deployed and using both foam monitors and side lines had the fire under control by 0956 hrs, approximately 13 minutes after arriving at the accident site. There was no evidence of fire prior to ground impact.

Approximately 1,060 litres of protein foam compound and 35,961 litres of water were used to extinguish the fire.

Lusaka City Fire Brigade and Zambian Air Force appliances also turned out in support of the Airport Fire Service.

1.15 Survival aspects

The magnitude of the impact forces were such that the accident was non-survivable.

1.16 Tests and research

1.16.1 Stabilizer static load tests

As a part of the investigation a full scale static load test programme was conducted by the Boeing Company. The purpose of the tests was to investigate the stress distribution in the stabilizer structure generally with particular emphasis on the area around the inner end of the rear spar and to examine the failsafe behaviour of the structure subsequent to a top chord failure.

The test specimen consisted of a left stabilizer and centre section from a 707-436 aircraft which was withdrawn from service after 56,227 hours and 20,052 landings, together with a right stabilizer from a different 707-436 aircraft which had experienced 54,086 hours and 19,991 landings before being withdrawn from service. The components were checked for structural integrity, build and modification standard before being installed in a static test rig simulating an aircraft installation. The entire assembly was secured to the test frame by means of the two aft stabilizer pivot points and the stabilizer trim screw jack pivot point. The stabilizer was set at the zero degree neutral position for all the tests. (See figure 10).

Test loads were applied to the structure at five spanwise stations on each stabilizer. At each station the loads, which were generated by hydraulic jacks operating through a system of linkages, were applied externally via load pads at the front and rear spars.

Both the stabilizers and the centre section were comprehensively equipped with foil strain gauges. In addition, vertical structure deflections were measured along the front and rear spars at ten spanwise locations by means of electrical deflection indicators.

The test loads were calculated to approximate the design values for (in order of priority) bending moment, torsion and vertical shear at the fatigue crack station. A total of nine tests were conducted, using theoretically derived loads representing the four conditions listed below.

Condition 1. Level flight, 1g with 50° flap (simulating the Lusaka approach).

Condition 2. Level flight, 1g with 50° flap (simulating abrupt full up elevator during the Lusaka approach).

Condition 3. Positive manoeuvre.

Condition 4. Negative gust.

Tests 1-4, which were intended to provide stress distribution data only, were completed without incident.

In test number 5 the rear spar upper attachment pin was removed. A load was then applied, incrementally, to 67% of load Condition 4. There was no damage to the structure. This was similar to the principal failsafe condition analysed for the purposes of certification.

Test number 6 was designed to simulate fastener distress in the inboard fasteners. The attachment pin was replaced and the first ten fasteners at the inboard end of the rear spar top chord forward flange of the left stabilizer were removed for one test. A load was applied incrementally to 100% of load Condition 1. There was no damage to the structure.

In test number 7 the ten fasteners were replaced and the rear spar top chord of the right stabilizer was cut to simulate the fatigue failure found in G-BEBP. A load was then applied to the test specimen incrementally to 100% of load Condition 1. At 87% load all skin to chord fasteners on the forward flange inboard of the cut failed and at 100% load the top web fractured from a point just inboard of the cut. Post-test examination of the structure revealed that the top web fracture extended down to within 6 mm of its lower edge and that there was a tear 38 mm long in the stabilizer top skin aft of the rear spar.

In test number 8 no repairs were made to the damaged right stabilizer specimen. Loads were applied up to 100% of load Condition 2. Examination of the structure after the test revealed that the top web fracture had extended to the full depth of the web and that the length of the tear in the skin aft of the rear spar had increased to 100 mm.

The final test, number 9, was conducted without making any repairs to the stabilizer. A load was applied incrementally to 80% of load case 4. Whilst holding this load the right stabilizer rear spar, front spar and closure rib fractured.

The character of the spar failure produced on test was strikingly similar to that evident in the wreckage of G-BEBP. (See figures 11a and 11b).

1.16.2 *Measurement of horizontal stabilizer flight loads*

In order to check the accuracy of existing stabilizer flight load data, which had been based on wind tunnel tests and on extrapolation of flight data obtained from earlier models of the 707 and from AWACS aircraft, the Boeing company conducted a flight test programme on a suitably instrumented 707-300 series aircraft during which horizontal stabilizer flight loads were recorded throughout the normal flight envelope. In general the load values obtained (see figure 12) approximated quite closely to the predicted values. It can be seen that the maximum (normal operational) horizontal stabilizer down loads were experienced with the aircraft in the landing configuration with 50° wing flap, leading edge flap extended and the landing gear down. In the normal landing configuration the flight tests indicated that the horizontal stabilizer bending moment during a simulation of the Lusaka approach was 75% of the value which caused the static test specimen to fail. Analysis shows that application of up elevator could increase this figure to about 120% of the test failure load.

It was found that during a normal landing roll, with spoilers deployed and using reverse thrust, the horizontal stabilizers were subjected to oscillating loads, at a rate of approximately 5 cycles/second, which peaked at a value of 80% of the maximum load on a typical flight. These oscillating loads, which were found to be caused by speedbrake deployment, were not accounted for during the initial fatigue analysis and explain the higher than expected crack growth rate on G-BEBP.

1.17 Additional information

1.17.1 Boeing 707 type history

The Boeing 707 family of aircraft began with the KC 135 military transport and 100 series airliner. A small number of 100 series derivatives known as the 200 series were also produced. The horizontal stabilizers on each of these variants were identical and had rear spars of 2 chord layout (see figure 13a). The failsafe capability of these stabilizers was demonstrated for the purpose of certification by dynamically failing a special top chord attachment pin during a ground test programme so allowing the front spar structure to carry the bending loads and transfer them to the centre section. The manufacturers also carried out a stabilizer fatigue test to investigate the sensitivity of the structure to fatigue damage. The 100 series was subsequently developed into the 720 series which retained the original 100 series type of stabilizer.

During fatigue testing on the 100 series stabilizers (which is of a different design to the 300 series) a fatigue crack was produced in the rear spar top chord at a point slightly outboard, relative to the closure rib, of the crack on G-BEBP. The crack started on the chord aft flange as a result of loads being transmitted into the chord by the trailing edge structure and is not related to the crack in G-BEBP which occurred in the chord forward flange as a result of load transfer from the torsion box top skin.

When the 707-300/400 series was developed the stabilizer assembly was extensively redesigned. The stabilizer span was enlarged to increase tail volume, and to cater for the resulting increase in load, the failsafe capability of the structure was upgraded by the provision of a third chord and terminal fitting which was located at mid-spar depth on the rear spar neutral axis. (See figure 13b). No fatigue tests were carried out on the redesigned stabilizer structure.

During the 300 series development flying programme the aircraft was found to have unsatisfactory elevator response characteristics. The cause of the problem was identified as a shortfall in stabilizer torsional stiffness. The torsional rigidity of the stabilizer structure was restored to an acceptable level by doubling the stabilizer light alloy bottom skin over the inboard region of the torsion box and by changing the material of the corresponding top skin to stainless steel, with a suitable adjustment of skin thickness. (See figure 14).

1.17.2 Airworthiness requirements

1.17.2.1 Certification history

All Boeing 707 variants were originally certificated by the United States Civil Aeronautics Administration, later the Federal Aviation Agency (FAA). Acceptance of the series onto the British register was based on the Air Registration Board (ARB), later Civil Aviation Authority (CAA), validation of the FAA certification. The first of the 707 series to be accepted on the British register was the model 436 on 26 May 1960. The 707-400 series is essentially a 300 series aircraft fitted with Rolls-Royce engines. The first model 300C, a 336C, was accepted onto the British register on 7 January 1966 after an assessment had been made of the differences notified by the manufacturers between the 436 and 336 models. The original USA certification dates for the 400 and 300C series were 12 February 1960 and 30 April 1963 respectively.

The relevant British Civil Airworthiness Requirements (BCAR) related to the problems of fatigue at the time of validation by the ARB were BCAR Section D Chapter D3-1 (16 March 1959) paragraph 5 and its associated appendix material. The relevant US Civil Air Regulations (CAR) were CAR 4B:270 and the related appendix H.

Both sets of national regulations contained safe fatigue life or failsafe design options. The Boeing 707 was designed so as to comply with the requirements of the failsafe option. Neither the US nor the British airworthiness regulations specifically required fatigue testing. In both cases the manufacturer was permitted to demonstrate compliance "by analysis and/or tests". Also, for the safe fatigue life case, it was acceptable that the service history of aircraft of similar design, taking into account differences in operating conditions and procedures, be used as a basis for fatigue life assessment.

The British evaluation of the 707 series 436 for UK type certification was based on an assessment of the certification work already carried out as a part of the US type certification programme for the 300 series aircraft. The procedure adopted by the UK CAA (then ARB) was to accept, subject to review, those areas of US certification work which were equivalent to the then current UK requirements. In those areas where the UK regulations were more stringent, or where the type 436 differed from the previously US certificated 300 series, a re-assessment was made. The outcome of this evaluation in relation to the fatigue integrity of the structure was that the aircraft was awarded a British type certificate on the basis of an FAA certificate of compliance with the failsafe option of CAR 4B:270 together with an ARB specified special condition. This condition required that a cracked structure should withstand limit flight loads, generally somewhat higher than the loads specified in CAR 4B:270, and the submission of an inspection programme designed to detect cracks before they reached dangerous proportions. The 707-321C series aircraft was certificated in the UK following the evaluation of those areas, notified by the manufacturer, which differed from the 436 series aircraft.

1.17.2.2 Certification testing

The Boeing 707-100 series aircraft underwent fatigue and failsafe test programmes prior to Civil Aeronautics Agency/Federal Aviation Agency (CAA/FAA) certification. These tests included tests specific to the horizontal stabilizer structure.

As a result of the fatigue test programme, cracks were found in the horizontal stabilizer rear spar upper chord rear flange after 240,000 simulated flights.

Failsafe tests were conducted using special spar top chord attachment pins which were designed so as to fail dynamically at the test load. These tests demonstrated that the load was transferred to the front spar assembly and that the structural integrity of the horizontal stabilizer was maintained.

No further structural testing was carried out nor was it required by the then existing or present US airworthiness requirements, when the redesigned horizontal stabilizer was introduced for the 300 series aircraft. The airworthiness criteria were met by calculations which were deemed to show that the static and failsafe strengths of the 300 series horizontal stabilizer were adequate for the designed purpose.

It should be noted that the existing UK regulations pertaining to the horizontal stabilizers required that "parts which may be critical from fatigue aspects shall be subjected to such analysis and load tests as to demonstrate either a safe fatigue life or that such parts of the primary structure exhibited the characteristics of a failsafe structure". No such tests were called for by the UK ARB who based their acceptance of the aircraft type on the US certification which had taken into account the 707-100 series tests together with analysis of the 707-300 series design.

1.17.3 *Maintenance schedules*

Whilst G-BEBP was operating on the US register it was maintained to a maintenance schedule approved by the US CAA/FAA. When the aircraft operated on the British register it was maintained to a maintenance schedule approved by the UK CAA.

The manufacturers publish a "Maintenance and Planning Data" document (BMPD) for the 707 series aircraft, which is intended to provide guidance for operators and their respective airworthiness authorities in drawing up each individual operator's maintenance schedule. In practice the BMPD usually forms the basis of the initial "approved maintenance schedule", but it should be emphasised that the information it contains is for guidance only. The document is divided into volumes, with the first volume covering the normal maintenance A, B and C type checks, and the second volume covering the structural inspections designed to highlight critical areas in the structure.

The regular inspection items in the BMPD which cover the stabilizer rear spar outboard on the closure rib are listed under " 'C' check type items" and call for a visual inspection (to a standard defined in the document and reproduced in appendix A) of the stabilizer "exterior surfaces" and the "rear spar and hinge fittings". The 'C' check inspection does not require access panels to be opened. The opening of access panels is a requirement in the inspection called up in volume II which requires a more searching inspection of the rear spar structure. This requires that the rear spar structure is inspected internally through access panels in the trailing edge structure and via the leading edge panels. The operator is advised to "Inspect spar and rib chords, webs and stiffeners for cracks or loose fasteners". The recommended period for this inspection is continually being modified in the light of service experience, but at the time of the accident to G-BEBP the recommendation was that a quarter of the fleet be inspected every 21,000 hrs.

The UK CAA approved Dan Air maintenance schedule applicable to G-BEBP was, generally, similar to the recommendation given in the BMPD. However, there were significant detail changes. The "exterior surfaces" check listed in the BMPD appeared as "External Skin – CHK" (with a CAA defined meaning of "CHK", ie check, listed in the front of the schedule and reproduced in this report, appendix A). In addition to the instruction to "CHK" there was the note " – close visual inspection check all exterior plates closed". In the case of the BMPD's entry "Rear spar and hinge fittings", there was no corresponding direct reference to the rear spar contained in the Dan-Air schedule, although the "hinge fittings" were covered as a part of the elevator checks. The corresponding work card used by Dan-Air, contained no direct reference to the rear spar. It did however give the instruction to inspect, amongst other items, "all visible structure" which, if carried out, would cover the rear spar to the same level as the exterior surfaces entry in the BMPD. It also listed five key points which covered cracks in the elevator closure ribs, hinges, skin doublers at the fin attachments, balance panel covers and condition of the trailing edge honeycombe structure.

The BMPD's structural inspection was reproduced without alteration in the Dan-Air Schedule, but the period between inspections was reduced to quarter fleet every 14,000 hours.

Other Maintenance Schedules

The schedules (applicable at the time of the accident) used by a number of other UK operators were examined and compared to the Dan-Air schedule. The schedules used by two of the operators were not of UK origin (one originating in the USA and one in Europe) but had been approved by the UK CAA for use as an interim measure until the operators concerned had developed their own schedules. In each case there was no direct reference to the rear spar. The wording of the external inspection item varied from "horizontal stabilizer external skin – CHK" as used in the Dan-Air schedule to "horizontal stabilizer complete – CHK". The two schedules of foreign origin were found to call for a check on the "External skin panels" (European schedule) and an "External thorough inspection of structure" on "Horizontal tail surfaces" (USA schedule).

1.17.4 Stabilizer cracks 707-300 series

A review of the 707 fleet worldwide in June 1977 showed that 521 aircraft were then operating fitted with the 300 series horizontal stabilizer. A survey of post-accident inspections of these aircraft revealed that 38 of these aircraft (ie 7% of fleet) were found to have horizontal stabilizer rear spar cracks of varying sizes. Four of these required spar replacement.

2. Analysis

2.1 It is quite clear that the direct cause of the accident was the loss of pitch control following an in-flight separation of the right horizontal stabilizer and elevator together with associated damage to the stabilizer incidence control. The investigation was therefore directed towards determining how such a major structural failure and loss of control could occur in a well established aircraft type. The main areas covered in the investigation were design, metallurgy, airworthiness certification and maintenance.

2.2 **Sequence of events**

The evidence from the accident site clearly indicated that at the time of ground impact the aircraft was in a 50° dive, but that the nose down pitch attitude was approximately 10° beyond the vertical. The aircraft was in the normal approach configuration with the landing gear and flaps fully extended and the engines delivering low to moderate power. All spoilers were fully retracted.

The extreme negative incidence at impact, considered in conjunction with the distribution of the wreckage points to an extremely rapid nose down pitch change following the loss of the stabilizer. Some of this pitch change can be explained by the sudden loss of half the stabilizer total down-load. However, calculations based on the known pitch inertia, the estimated pitch change between stabilizer departure and aircraft impact and the associated time interval derived from the CVR, suggest that the loss of the stabilizer alone could not have produced the required pitch acceleration without the application of full aircraft nose down trim on the remaining (left) stabilizer.

Examination of the wreckage produced evidence from three independent sources of a stabilizer trim setting of 6¼ units "aircraft nose up" at the point of stabilizer separation ruling out any suggestion that a trim runaway had occurred.

Evidence was also found which indicated that, following the loss of the right stabilizer, the sudden application of asymmetric loads to the stabilizer centre section and rear fuselage were of a magnitude sufficient to break out partially the centre section mountings and allow the left stabilizer to move downwards, twisting the centre section relative to the fuselage (see figure 15). This damaged the fuselage structure adjacent to and below the left stabilizer rear spar. Several sections of structure from this area, including one with witness marks produced by the left stabilizer were found on the ground in the airborne separation debris zone. It follows, therefore, that these stabilizer position witness marks were made during the airborne break-up sequence. The stabilizer jack screw was found to have been bent and fractured at the base. The direction of bend was consistent with a tip downwards movement of the left stabilizer, which is corroborated by the other evidence. It is considered probable that this bending and subsequent fracture occurred in the air. Once the jack screw had been fractured, the stabilizer would be free to rotate in pitch. There is evidence that during the separation of the right stabilizer the right elevator push/pull tube was torn from the elevator quadrant, which is situated in the rear fuselage, resulting in the elevator circuit being forced to the full up elevator position. In addition to increasing the down load on the remaining stabilizer, this would tend to shift the stabilizer centre of pressure (CP) aft and if the jack screw were broken, servo the whole left stabilizer into a high stabilizer leading edge up attitude (ie aircraft nose down).

Further examination of the section of structure containing the stabilizer position witness marks revealed evidence of a second witness mark made after the first, but still during the airborne break-up sequence, corresponding to a stabilizer incidence setting displaced 25° stabilizer nose-up from the pre-crash trim position. This position is well outside the normal range of movement of the stabilizer trim and tends to confirm that the trim screw jack fractured during the break-up sequence. The full aircraft nose down trim change, which the initial pitch rate calculation indicated was a necessary factor in explaining the violent pitch rate, is thus supported as having occurred.

In summary the accident sequence is seen as follows:

(i) Shortly after the selection of 50° flap, with the aircraft on a normal approach the right horizontal stabilizer and elevator became separated from the airframe structure.

(ii) The instantaneous loss of a significant stabilizer download caused an immediate nose down pitching moment associated with a large asymmetric load on the stabilizer centre section to fuselage attachment points.

(iii) In the break-up, the elevator linkage was pulled to the full up elevator position, further increasing the asymmetric load on the stabilizer centre section mountings.

(iv) The asymmetric load was sufficient to wrench the left stabilizer downwards twisting the centre section in the fuselage, fracturing the stabilizer jack screw and allowing the whole stabilizer to rotate in pitch to a large stabilizer nose up angle under the influence of the fully up elevator, which would tend to act as a very large servo tab. It is also likely that crew reaction to the aircraft nose down pitch change would be to hold the stick fully back.

(v) A very rapid pitch down resulted and the aircraft struck the ground in a 50° dive but with the pitch attitude at approximately −100°, some 500m beyond the point of stabilizer separation.

2.3 Design

The original Boeing 707-300 series stabilizer differed from the 100 series design by having increased span and a re-designed rear spar of three chord construction. The rear spar was re-designed because the failsafe capability of the original structure with a top chord failure would not have been adequate to cope with the increased loads acting on the larger stabilizer. It was during the initial 300 series design phase that the assessment of fatigue sensitivity and failsafe capability was made for the purposes of certification.

Fatigue tests on the earlier 100 series stabilizers had produced a crack in the top chord of the rear spar after a period representing some 240,000 flight cycles. The crack was caused by loads which were being fed into the chord by the trailing edge structure at a point where there was a change in chord geometry. There were no indications of problems arising out of loads from the torsion box. The new 300 series spar chords were continuous extrusions with integral terminal fittings and had no abrupt changes in section. It was therefore reasonable to conclude that, because of the similarity of the 100 and 300 series structures in the undamaged state, these spar chords would have an improved fatigue life over the original 100 series chords. The manufacturer appears to have taken this view and considered the rear spar safe in terms of fatigue in a normal service environment. However, the design was certificated on the basis that it was failsafe, not as a result of fatigue tests.

During the initial flight test programme a lack of stabilizer torsional stiffness became apparent. This shortcoming was cured by stiffening the top and bottom inner torsion box skins, which in the case of the top skin was achieved by a change in material from light alloy to stainless steel. This modification was made after the basic stress analysis work had been carried out. Because the stabilizer was certificated on static strength failsafe capability restressing was limited to that necessary to ensure that the static strength was not reduced by the modification.

It was known that the greater stiffness of the stainless steel skin would result in higher skin loadings, and hence higher fastener loads in the steel "hi-shear" fasteners towards the root end of the rear spar top chord. These higher fastener loads would also increase the bearing stress in the chord forward flange. However, given the existing chord flange design, there was little that could be done to improve this situation because the use of larger diameter fasteners to reduce the bearing stresses would have reduced the edge margin to an unacceptable level. (Boeing's current 1978 fatigue design practice is to use larger edge margins than were used on the 300 series.) However, it was considered that the design was adequate in this area, given the general acceptance at that time of its failsafe capability. It was not realised that the skin modification, whilst improving the static strength, would significantly reduce the fatigue strength. This was the first of a chain of events which culminated in the accident to G-BEBP.

In the original 100 series design, which was also certificated on the basis of failsafe capability, the ability of the stabilizer structure to carry the flight loads following failure of the top chord was demonstrated on test by dynamically failing a special top chord terminal pin at the full test load, thus allowing the loads to be carried by the front spar and the rear spar bottom chord. This was considered to be the "worst case" which would cater for a failure at any station on the rear spar top chord including lug, pin and centre section lug failures. This was a reasonable assumption to make in the case of a two chord rear spar beam because effective removal of the top chord would remove all rear spar bending stiffness, for all practical purposes, and the lower chord would serve to transfer shear and end loads only with the front spar and top and bottom skins taking over in providing the principal bending stiffness.

An almost identical approach was adopted in the case of the 300 series stabilizer with a top lug failure being considered the most critical, and service experience has since shown the lug areas to be potential failure points. The designers appear to have been conscious of the need to cater for a failure at this point, (ie inboard of the closure rib). This concern is reflected in the design of the shear plates (see figure 13b) between the top and centre chords just inboard of the closure rib, which are far stiffer than the normal transfer of shear load to the bottom chord terminal fitting dictates. The excess load capacity in these shear plates was clearly intended to transfer the loads from the top chord to the centre chord terminal fitting in the event of a lug type failure. Care was also taken to ensure that the centre chord and centre chord terminal pin would not be damaged by fatigue during normal service. This was achieved by locating the centre chord on the rear spar neutral axis and by making the terminal pin a loose fit compared to the top and bottom chord pins, thus ensuring that the centre chord assembly was not loaded under normal conditions.

It is considered that the design of these shear plates and the centre chord assembly is evidence of a responsible approach on the part of the manufacturers in attempting to cover, with additional margins of safety, the failure case which they considered to be the most critical, or the most likely to occur. However the apparent lack of attention given to potential top chord failure cases outboard of the terminal fittings strongly suggests that the earlier work on the 100 series design influenced thinking on the 300 series design.

Whilst it might be considered reasonable to view the 707-100 and 300 series horizontal stabilizer structures as being broadly similar, this line of thought is only appropriate when the structures are completely undamaged. Subsequent to a top chord failure, the 300 series stabilizer structure behaves in a fundamentally different manner to that of the 100 series stabilizer.

In the case of the 100 series stabilizer, the rear spar carried no significant bending loads following a top chord failure. In the 300 series design, for a similar top chord failure, the rear spar was intended to carry the bulk of the bending loads so that for top chord fractures at various spanwise locations, different stress behaviour patterns in the adjacent spar structure could be expected as the loads transferred around the fracture. The local stress behaviour in the spar is especially critical, as events have shown, in the inboard region of the spar. It is clear that the behaviour of the structure with a fracture just outboard of the large shear plates will be more critical than a fracture on the inboard side. The significance of the local stress field in the vicinity of a top chord fracture does not appear to have been appreciated at the design stage. Instead, the failsafe capability was calculated on the basis of the strength of the centre chord, lower web and bottom chord acting in isolation inboard of the fracture.

The failure to appreciate the influence which the top chord and upper web inboard of the fracture has on the local stress distribution was the principal factor in bringing about the final spar failure which resulted in the accident to G-BEBP.

2.3.1 *Failsafe design philosophy*

A failsafe design is one in which there are one or more redundant structural elements which are capable, in the event of a failure of one of the primary load members, of carrying the flight loads. However, a failsafe design is only failsafe whilst the degree of redundancy is sufficient to cater for a failure, ie a singly redundant structure (as in this case) is only failsafe whilst the primary structure is intact. Once this has failed, the principle of safe-life obtains and it becomes necessary to find the failure in the primary structure before the failsafe members themselves can become weakened by fatigue, corrosion or any other mechanism. Because the strength reserves in the failsafe mode are usually well below those of the intact structure, this means that in practice, the failure must be found and appropriate action taken within a short time compared with the normal life of the structure. Obviously the degree of urgency which attaches to a given failure depends on the design of that particular structure and its stress and corrosion environment as well as the type of operations being flown. But it remains a fact that in order to maintain the safety of a failsafe structure, there must be an inspection programme forming an integral part of the total design to ensure that a failure in any part of the primary structure is identified well before any erosion of the strength of the redundant failsafe structure can occur.

The concept of failsafe structures (as understood in 1978) depends on the following inter-related factors, all necessary for the safe implementation of the philosophy:

(a) Adequate design to cater for the basic operating and load environment in a manner which:

- (i) Meets the basic structural strength requirements for the intact structure.
- (ii) Predicts and minimises the likelihood of significant damage or failure in the primary structure caused by fatigue, corrosion, accidental damage, or for any other reason.
- (iii) Provides adequate access to (and identification of) primary structure so that regular inspection for failure or damage can be accomplished easily.
- (iv) Can, in the event of a failure of any part of the primary structure, sustain the flight loads with adequate safety margins for a period long enough to enable the failure to be detected during routine inspection.
- (v) Utilizes suitable materials for each element in the design.
- (vi) Defines those areas of structure which fall into the category of primary structure, and gives adequate guidance as to the type of inspection which should be carried out and maximum periods between such inspections.

(b) Quality control – to ensure that each aircraft is built and serviced with spares which meet the design specification.

(c) Adequate maintenance – the implementation of a suitable maintenance schedule which incorporates in a suitable form the "failsafe design" inspection package produced by the manufacturer.

(d) Feed-back, to the manufacturer and Airworthiness Authorities, of information about significant failures and recurring failures – so that any shortcomings in the design can be detected and remedial action taken.

In the case of the 300 series stabilizer only the basic structural strength requirements of the intact structure and the quality control requirements appear to have been completely satisfied.

2.4 Horizontal stabilizer failure

The total failure of the right horizontal stabilizer structure was the result of two physical mechanisms. Firstly, the failure, as a result of long term fatigue damage, of the rear spar top chord and secondly, the inability of the redundant failsafe structure to carry the flight loads for a period long enough to enable the fatigue crack to be detected during routine inspection using the then current inspection procedures.

2.4.1 *Fatigue failure of the top chord*

Metallurgical examination established that the fatigue crack originated in the upper edge of the eleventh skin fastener hole in the top chord forward flange and that the crack had been growing over a period of some 7,200 flights before chord failure occurred.

A superficial consideration of the structure does not give any indication of the reason why the crack should have occurred at precisely that location. The spar chord itself is a smoothly tapering section with no stress concentration inducing irregularities or discontinuities in the load input points, such as bolted fittings. However, the discovery, as part of the post accident fleet inspection, of a 707-436 aircraft with a large crack also originating from the 11th fastener indicates that the crack in G-BEBP was not an isolated case and that some mechanism was operating which was causing cracks in the top chord adjacent to the closure rib on the outboard side. This was reinforced by the discovery of another extensively cracked top chord on a 707-300 series aircraft, this time originating from the 5th fastener.

Although metallurgical examination indicated that the origin of the crack on G-BEBP lay in the upper edge of the 11th fastener hole, impact damage prevented the nature of the origin itself from being determined. Other small cracks were also found to be present in the same region and there was evidence of corrosion in the form of small pits and layer corrosion. None of the small cracks, which would have formed at about the same time as the major crack, were associated with corrosion. This indicates that the corrosion probably occurred after crack initiation and suggests that a high local stress is likely to have been the sole cause of the major fatigue crack. This is also compatible with the small variation in the location of cracks found on other aircraft, where corrosion might be expected to introduce a more random distribution.

It is known that the stress distribution in the skins of a torsion box structure will be modified in the vicinity of the closure rib, which, because of its lack of bending stiffness, cannot react the bending (spanwise) stress in the skins. The only way which these stresses can be reacted is by diffusing out into the spar chords and thence to the chord root fittings, with the result that there will be a roughly triangular dead area of unloaded skin between the spars adjacent to the closure rib. The diffusion of the skin stresses out into the inboard corners of the torsion box covers (see figure 16) will result in the principal stress direction adjacent to the inboard skin-to-chord fasteners being at a significant angle to the chord spanwise axis. It follows that in order to react the spanwise stress component in the skin, the magnitude of these fastener loads, acting at an angle to the top chord, must be greater than the fastener loads further outboard which are more closely aligned with the chord spanwise axis.

The spanwise load component on the innermost fasteners will be increased further as a result of the normal load transfer characteristics of the type of joint used between the stabilizer skins and the spar chords. Theoretically, the end fastener will be the highest loaded in this type of joint. In addition to these two mechanisms the introduction of the stainless steel skin will also lead to a higher overall load transfer into the spar chord because of its greater stiffness, which will result in its carrying more load than the alloy skin it replaced. On deeper consideration therefore, a high load environment in the inboard fasteners is not altogether unexpected and the environment which produced the crack in G-BEBP can to some extent be explained. It should be noted that the manufacturer anticipated higher unit fastener loads in this area as the design utilizes steel "hi-shear" fasteners in place of the light alloy fasteners used further outboard.

The fact that on two different aircraft (G-BEBP and a series 436) fatigue cracks were found emanating from the same fastener hole (No. 11) cannot be lightly dismissed as merely a coincidence. The reason for the precise location of the crack on G-BEBP can be seen from a study of the top skin to rear spar fasteners. Some fastener heads on the left (ie intact) stabilizer from G-BEBP were distorted in a manner which indicated high load transfer in a generally spanwise sense. This distortion was most extensive on the inboard fasteners and the distortion gradually reduced until, by about the 10th fastener outboard, there was no detectable distortion. It was also noted that the Nos. 2, 3 and 4 fasteners had been replaced by light alloy "cherry lock" type fasteners which were not of the correct standard and lower in strength than those supplied by Boeing. No record could be found which indicated the reason why these fasteners were changed and in the absence of information to the contrary, it is considered probable that the original fasteners were distorted or loosened to the extent that they required replacement, and the maintenance organisation carrying out the work did not realize that the fasteners were located in a critical area requiring specific fasteners. In any event they did not comply with the general requirements of the Boeing Structural Repair Manual which, in Sections 51-2-1 and 51-2-3 require either "hi shear" fasteners or fasteners of equivalent or greater strength to be used.

Examination of the top skin fastener holes from both the left and right stabilizers from G-BEBP reveals a mirror image pattern of hole distortion, although there is a second pattern of distortion on the right skin which occurred as the fasteners sheared following chord fracture. This similarity in the damage pattern, (see figure 18), indicates that the general stress behaviour in the left stabilizer was identical to that in the right stabilizer prior to top chord failure. Further evidence of a common stress behaviour was noted in the top skin fastener hole elongation pattern from the 436 series cracked stabilizer, which was very similar to that on G-BEBP. The skin hole damage patterns clearly indicate the skin stress diffusion into the root end of the chord and this evidence agrees well with the theoretical diffusion pattern discussed earlier and shown in figure 16. Although the direction and degree of elongation of the holes are not a direct measure of the direction and magnitude of the fastener loads, because the structure stiffness changes as the distortion progresses, they nevertheless illustrate very clearly the general stress behaviour pattern and confirm that this was the high load inducing mechanism which gave rise to the fatigue crack. The damage pattern also indicates why the fatigue did not occur at the most highly loaded inner fasteners.

It can be seen that the greatest hole distortion occurred at those fasteners at the inboard end where the axis of elongation (ie direction of stress) is at the greatest angle to the spar spanwise axis. Further outboard the degree of elongation of the holes and the angle of the hole axis is reduced until at about the 10th or 11th hole, there is negligible distortion and the stress direction is parallel with the chord spanwise axis. It would appear therefore that the crack occurred at the 11th fastener in this case rather than further inboard, because the unit fastener loads inboard of that location were sufficiently high to distort significantly the fastener and the skin, thus relieving the load before any damage could be done, but at the 11th fastener the load direction was spanwise and the magnitude of the fastener load was therefore lower. It appears that this load was just below that which would damage the fastener or skin, but was high enough and maintained for long enough that the fastener induced stress in the flange hole, when combined with the basic stresses due to bending in the spar, were sufficient to initiate and propagate a crack.

Further confirmation of the general stress behaviour, was obtained from the post-accident structural test programme. In figure 17 the principal stresses in the skin adjacent to the rear spar at the inboard end are shown for the "full up elevator deflection" case. The similarity between this figure and the photograph of the skin hole distortion (see figure 18) and the idealised stress pattern (figure 16) can clearly be seen. The tests also demonstrated that the general behaviour of the stresses is not significantly influenced by the application of up elevator (ie increased torsion load) and that the magnitude of the stress is also relatively insensitive to torsion. (The direction of the stresses for the zero elevator deflection load case (not shown) were the same but the magnitudes of the stresses were approximately 30% lower.) Figure 19 further illustrates the rapid load transfer from skin to chord close to the inboard end of the chord.

A comparison of the top and bottom skin stresses in figure 17 shows that the effects of diffusion in the top skin are far more pronounced than those in the bottom skin in the same area. It appears that the diffusion effects in the stainless steel top skin were much more localized than the designers anticipated, judging by their use of "hi-shear" fasteners out as far as 60 cm from the closure rib. It also seems likely that the stress level in this area of the top skin was significantly higher than anticipated. It has not been possible to establish positively why this was so. The exaggerated stress diffusion behaviour in the stainless steel top skin compared with that in the light alloy bottom skin in an otherwise broadly symmetrical structure and load field, can partly be explained by the inherent instability of the bottom skin when acting under compressive loads, but this is not considered to be the principal factor. It is more likely that the change in the top skin material from light alloy to stainless steel significantly modified the stiffness distribution within the structure allowing the top skin to carry a greater load than was intended. It was the diffusion of these loads into the top chord which created the high stresses which ultimately caused the fatigue failure in G-BEBP.

2.4.2 *Failure of the failsafe structure*

The 707-300/400 series aircraft were certificated on the basis that following a fatigue failure or obvious partial failure of a single principal structural element, catastrophic failure or excessive structural distortion was not probable. Events have shown that the intent of this requirement was not met in the case of G-BEBP.

The failsafe component in the rear spar consists of an additional spar chord and root fitting located at mid-spar depth. This chord extends outboard approximately 2.5m from the terminal pin. The chord was located on the neutral axis of the spar so that during normal service it would carry no load and would therefore be undamaged by fatigue. This philosophy of preserving the reserve strength capability of the chord was further extended by the use of a terminal pin of slightly smaller effective diameter than the top and bottom chord pins so that, again during normal service, the pin and centre chord remained unloaded. The only obvious penalty of this design arrangement is that, subsequent to a failure of the top chord and upper web, the bending stiffness of the remaining spar structure, ie centre chord, lower web and bottom chord is reduced to 25% of the original value for the intact assembly.

On the basis of calculations made for the purpose of type certification it was considered that the strength reserves in the rear spar subsequent to a top chord fracture of the type found on G-BEBP were adequate. Calculations made during the course of the investigation, based on the assumption that the spar was effectively reduced to the centre chord, lower web and bottom chord also indicated that the strength of the degraded structure should have been adequate to tolerate the loads estimated to have been acting on the horizontal stabilizers of G-BEBP immediately prior to the break-up.

However, in the accident to G-BEBP the failsafe structure clearly had not behaved in the way which had been predicted and it is necessary to re-examine the problem. The unexpected structural behaviour could be explained:

(i) If the actual flight loads encountered were significantly different from the assumed loads used in the analysis.

(ii) If the materials used in the construction of the stabilizer rear spar were below specification.

(iii) If the standard of manufacture was below specification.

(iv) If the presence of unpredictable factors such as sabotage, military action, pre-existing damage (other than the top chord crack and associated damage) or a violent manoeuvre.

(v) If the behaviour of the damaged structure under load was significantly more complex than the behaviour model used in the analysis.

Flight load measurements, made during flights in which the Lusaka approach of G-BEBP was simulated, indicated that the loads were slightly higher than estimated, but not significantly so in terms of the basic residual strength calculations. Materials specification tests on the components of the right stabilizer spar have shown that the materials were up to specification in all respects. Careful examination of the structure from G-BEBP has revealed no evidence of military action, sabotage, pre-existing damage or non-standard manufacture with the exception of the replaced top chord fasteners, which would not have had a significant bearing on the failsafe strength. Witness statements, wreckage distribution, flight recorder readout and CVR replay all indicated that the aircraft was on a normal final approach involving no high load inducing manoeuvres. Although there was no accurate record of the degree of turbulence present during the approach to Lusaka, the balance of evidence from several sources suggests that turbulence was not a factor in this accident. It follows therefore that the most probable reason for the failure to predict adequately the events which occurred to G-BEBP was that the actual structural behaviour was more complex than the simplified behaviour model used in the initial analysis.

During the post accident static load tests it was found that, following the cutting of the top chord to simulate the fatigue fracture on G-BEBP, the spar fractured completely when the bending moment, (the critical load component), was approximately 35% greater than the best estimate of the steady state normal approach stabilizer bending moment. Application of up elevator during the Lusaka approach could have produced bending moments up to 25% greater than the test failure load.

It is clear therefore that the test specimen failed at a load which closely approximated the load estimated to have been acting in G-BEBP at the time of stabilizer failure, allowing for the presence of additional closure rib damage and centre chord fatigue and the likelihood of some elevator deflection which could not be reproduced on test. This together with the strikingly similar fracture characteristics (see figures 11a and 11b) indicate that the failure occurred as a result of an inherent inability of the failsafe structure to carry the flight loads subsequent to the fracture of the rear spar top chord, and was not due to some factor specific to G-BEBP. Consequently it is necessary to re-examine the behaviour of the rear spar with a fracture just outboard of the closure rib.

The strain survey data for the structure in the "cut" condition demonstrates that the top chord and web inboard of the cut is far from being ineffective, as had been initially assumed. The test data clearly indicated that approximately 30% of the total bending loads were being carried by the top chord root fitting. The stress plots in figure 21 show that the stresses in the top chord outboard of the crack diffuse through the web and into the centre chord in the manner initially predicted but then, instead of transferring directly via the centre chord to the centre chord terminal fitting, a significant proportion of the stresses diffuse back up via the upper web inboard of the crack into the top chord and thence to the top chord root fitting. This stress distribution produces an extremely large stress component on the centre chord top flange, which acts normal to the spanwise axis of the chord. Because the chord is an extruded section, its ability to resist stresses acting in this direction is lower than its strength in the axial direction. The cross-grain stress component due to the diffusion back up to the top chord was sufficient to pull the top flange away from the centre chord, starting a crack running inboard along the base of the flange. This in itself would not be catastrophic, but the change in cross section of the flange and chord just outboard of the terminal fittings together with the changes in the stress field as the crack progressed appear to have caused the crack to turn downwards across the main cross section of the centre chord. With the centre chord fractured, the remaining structure could not sustain the applied loads and total failure occurred. With the advantage of hindsight the reasons for this behaviour are apparent.

The load path from the centre chord (at the fracture station) inboard to the centre section depends on the relative stiffnesses of the centre chord and the top chord/web combination inboard of the fracture. The top chord/web combination inboard of the crack can be considered as a short and deep cantilever of considerable stiffness. The inherent stiffness of this cantilever is further increased by the large bolted shear plates which joint the top chord to the centre chord just inboard of the closure rib. The centre chord is attached to the bottom chord by a similar arrangement and the net result is that the inboard ends of the spar chords form a large "U" fitting between the top and bottom chord terminal fittings, providing a high degree of "built-in" stiffness. The top shear plates also shorten the effective length of the cantilever further increasing its stiffness in bending. In contrast, the inherent axial stiffness of the centre chord inboard of the fracture is significantly reduced by the terminal pin, which is a loose fit relative to the top and bottom chord terminal pins.

Had the stiffness of the top chord/web been insignificant compared with the centre chord stiffness then the structure should have behaved as the designers intended. As it was, the extremely high bending stiffness of the top chord/web together with the reduced axial stiffness of the centre chord inboard of the fracture resulted in load transferring round the web fracture back up into the top chord and thence to the top chord terminal fitting. It was this load transfer up into the top web which resulted in the centre chord flange failure, and the consequent centre chord and total spar failure.

2.5 Airworthiness requirements

At the time of airworthiness certification of the various Boeing 707 aircraft variants the national airworthiness requirements of both the USA and the UK were broadly similar, in that two basic options were available to aircraft manufacturers, safe fatigue life or failsafe. Boeing decided to design the 707 series against the failsafe strength criteria of the USA airworthiness requirements, vide CAR 4b.270(b). This regulation reads, in part:

> "**Failsafe strength.** It shall be shown by analysis and/or tests that catastrophic failure or excessive structural deformation, which could adversely affect the flight characteristics of the airplane, are not probable after fatigue failure or obvious partial failure of a single principal structural element."

Whilst the 707-100 series aircraft together with its military counterpart the KC 135 underwent extensive structural testing prior to certification and introduction into service the 707-300 series aircraft designers relied, quite reasonably as it appeared at the time, solely on the analytical approach in their pre-certification submission of aircraft data to the airworthiness authority.

In the light of this accident it can be argued that when an aircraft design evolves, as every successful airframe design invariably does, there must be a point in the evolutionary process when the manufacturer or the airworthiness authority (or both) should decide that a complete review is necessary because the basic design can have changed sufficiently for doubts to arise as to the validity of extrapolation of data from the earlier model.

Persuasive arguments are regularly put forward both by aircraft manufacturers and the airlines that a modern aircraft designed to failsafe principals should not be arbitrarily limited to a given service life, as the feedback of service experience from the fleet together with thorough inspection procedures would isolate problem areas before they became in any way critical. The circumstances of this accident have reduced the weight of these arguments for two reasons, firstly the emergence of a long term problem which had been totally unforseen, and secondly the fact that it had remained undetected until brought to light during the investigation. It is true that the widespread (in terms of the fleet) fastener distress associated with the fatigue problem could, in an ideal world, have highlighted the problem in ample time to prevent the accident, but this would have required an appreciation by the maintenance organisations concerned that this was a structurally significant area and that it was necessary for them to pass the information to the manufacturer, who would then have to diagnose correctly the cause of the problem and devise a solution. In this way the fatigue problem, but not the failsafe structure problem, could have been corrected. In reality however, fastener distress is a common problem which is not usually critical, indeed the fasteners on the top chord rear flange in the same region are a known "problem" area with the fastener heads pulling through the trailing edge skin, but this is not primary structure and is not critical. Without detailed knowledge of the structural behaviour of the aircraft as a whole, only those failures which are obviously significant will be reported.

The requirement for an inspection programme to detect a fatigue failure in good time was accepted in principal by the certificating authorities of both the United States and the United Kingdom. CAR 4b.270(b) does not specifically mention a need for an inspection to detect cracks, but in the related appendix H it recommends amongst other things "establishing the components which are to be made failsafe" and "establishing inspection programs aimed at detection of fatigue damage". It also advises "arrangement of design details to permit easy detection of failure in all critical structural elements before the failures can become dangerous or result in appreciable strength loss". The UK BCAR D31 paragraph 5 dealing with fatigue strength states that for those designs which are to be certificated as failsafe, "information should be provided in the Service and Instruction Manual as to the frequency and extent of the repeated inspection of the structure necessary to ensure that any failure of the Primary Structure will be found within a reasonable period". As in the case of CARs, the inspection element in the design is not obligatory but is strongly recommended. However, in the case of the Boeing 707 300/400 series aircraft, one of the special conditions for UK certification was the submission of an inspection schedule to detect cracks in the primary structure.

Clearly, in the case of the subject accident, compliance with the provisions of both CAR's and BCAR's was not sufficient to prevent the accident taking place.

2.6 Maintenance

2.6.1 Manufacturers recommended inspection schedule

Recommended inspection procedures published by the manufacturers for the 707 300/400 series (the BMPD document) fall into two categories. The first specifies, in general terms, the areas of structure and types of inspection which should be considered for the various category of regular inspection, referred to as A, B, and C checks. Recommended periods between the various categories of inspection were given in the front of the document with the 'C' check, which was the only check found to contain inspections which cover the rear spar top chord, being recommended at 1,800 hr intervals. The definition of the term 'check' was also listed in the front of the document and was stated to mean a thorough examination of an item for general condition with special emphasis in a large number of areas, including cracks. (See appendix A).

Of those Horizontal Stabiliser items listed as " 'C' check type items", only two entries covered the rear spar top chord (see appendix A) one referring to "Exterior surfaces" and the other to the "rear spar and hinge fittings". Both these items were tasked with the word "check", as previously defined, but the time allocation to cover these items together with a "check" on the "Brush and Seal" and "Drain holes" was given as 24 mins per stabilizer, which tends to suggest that an external visual inspection was intended rather than, for example, a more detailed eddy-current check. The manufacturer states that these were the checks intended to cover cracks in the top chord, and that the term "Rear spar and hinge fittings" meant **rear spar** and **hinge fittings**, rather than **rear spar fittings** and **hinge fittings**. Although such ambiguity is clearly undesirable, the intended meaning could be deduced from the fact that all those spar fittings not individually covered elsewhere in the document are internal, and therefore not accessible during an external visual inspection, whereas the principal elements in the structure of the rear spar itself (ie top and bottom spar chords) had been specifically designed to enable them to be inspected externally. Leaving aside the problem of the ambiguity, it is considered that the inspection recommended by the manufacturer would be adequate to detect a crack in the top chord provided the crack was reasonably visible.

It is known, from those cracks detected as a result of the post accident fleet inspection (which were relatively small when compared with G-BEBP) that partial cracks on the top chord, although visible to the naked eye when their precise location is known, are for all practical purposes undetectable visually, even with the aid of dye penetrant. It follows therefore that the recommended inspection could not have been expected to detect the crack in the spar chord of G-BEBP unless the 'C' check occurred during the interval between top chord severance and total spar failure, which was not so in this case. However the need to detect the chord crack during its early stages of development was a requirement arising as a direct consequence of the shortcomings in the failsafe design, of which there was no knowledge prior to the accident. If the structure had worked in the expected manner, the strength reserves would have been sufficient to allow the crack to progress right across the top chord, at which point the crack should open up sufficiently to allow detection at the next 'C' check.

If it is accepted that the inspection recommendations were based on the assumption that the failsafe structure could cater for a 100% crack in the top chord with adequate reserves, which was the basis for the structural design and subsequent certification, then the recommended procedures, providing they were correctly interpreted, should have proven adequate. It is considered however that the actual instructions contained in the BMPD document Vol. 1 erred on the side of brevity, and the resulting ambiguity could have significantly reduced their effectiveness.

Vol. 2 of the BMPD document contains the more detailed recommendations for "structural inspections". On this subject the recommendations are more explicit and advise an internal inspection of the structure including rib and spar chords. The inspection is not ideal however because there is no access to the forward side of the spar except via the leading edge, which prevents a normal close visual inspection from being carried out (see figure 22) also the interval between inspections is rather large at the periods currently quoted.

In conclusion, it would seem that the recommended 'C' check inspection, although somewhat ambiguous, was theoretically adequate to meet the inspection needs for detection of top chord cracks as those needs were seen at the design stage. The shortcomings in the failsafe design, which prior to the accident were completely unrecognised, imposed the more critical requirement to detect cracks at an early stage in their growth which the recommended 'C' check was fundamentally unable to bring about. The structural inspection covering the top chord is considered to be less than satisfactory because of the lack of sufficient access to the forward face of the spar. This inadequacy, in the category of inspection specifically intended to highlight problem areas in the structure before they become critical, is demonstrated by the failures in practice to detect the very long term cracks in G-BEBP and those other aircraft which were subsequently found to have significantly cracked horizontal stabilizer top chords.

2.6.2 *Dan-Air inspection schedule*

The inspection schedule used by Dan-Air Engineering was primarily based on the BMPD document and the FAA approved schedule used by the previous owner. The Dan-Air schedule was submitted to the UK CAA and approved by them. There were, however, differences, one of which was the omission of the "Rear spar" element of the "Rear spar and hinge fittings" instruction contained in the BMPD. The "hinge fittings" portion of the BMPD entry is listed under "elevator hinge assy on elevator and tailplane". There is also a difference in emphasis in the change from "exterior surfaces", as quoted in the BMPD, to the "external skin" listed in the Dan-Air schedule. It is considered that the change from "surfaces" to "skin" subdues emphasis on the structure, ie the visible section of the spar chords, and increases the emphasis on the skins, ie a check for aero-dynamic cleanliness rather than a structural check. This emphasis towards the skin is reinforced by the additional note "close visual inspection – check all exterior plates closed". Although the BMPD does in fact duplicate, in so far as the "rear spar" external structure check is, in practice, also done during the "exterior surfaces" check, the addition of the separate "rear spar" check does emphasise, and in fact require a double check on, the spar chords. In the Dan Air document, there is no direct mention of the rear spar and all the emphasis is placed on the skins and access panels. It has not been possible to establish whether the omission of the words "rear spar" was intentional, whether it arose as a result of the ambiguity of the BMPD text or whether it was effectively a clerical omission which occurred as a result of moving the "hinge fittings" check into the elevator section. It should be noted that none of the other schedules examined had included the term "rear spar"; most of the UK schedules being, in effect, identical to the Dan-Air document and the USA schedule (see 1.17.3) was the only one to call for a thorough external inspection of "structure".

The work card issued by Dan-Air referred to "structure" and in practice the inspector would be expected to cover the entire visible surface of the structure, however there was no emphasis on the rear spar. This was largely due to the list of check points, which were all added as a result of service experience of problem areas and were therefore in them-selves correct, but they tended to emphasise these other known problem areas at the expense of a reduction in emphasis on the basic structure. Nevertheless experienced personnel carrying out a visual inspection can be expected as a matter of course, to extend the scope of an inspection if they have reason to believe that cracks exist in the structure being examined.

In respect of the structural inspection, the BMPD recommendation was duplicated exactly except that the period between inspection was reduced from quarter fleet every 21,000 hrs to quarter fleet every 14,000 hrs.

It is concluded that the Dan-Air schedule, and to a large extent those other UK schedules examined, did not place adequate emphasis on the rear spar structure and the top chord in particular, and in this respect were less effective in the area in question than the original manufacturer's recommendations.

It should be noted that the shortcomings in the Dan-Air schedule relative to the BMPD were probably not in themselves direclty relevant to the accident for the reasons discussed in paragraph 2.6.1.

3. Conclusions

(a) Findings

(i) The aircraft had been maintained to an approved maintenance schedule and its documentation was in order.

(ii) The crew were properly licenced and adequately experienced to carry out the flight.

(iii) Pitch control was lost following the in-flight separation of the right hand stabilizer and elevator, which occurred shortly after the extension of 50° flap.

(iv) The stabilizer variable incidence screw jack actuator fractured in the stabilizer separation sequence allowing the left hand stabilizer to travel to the fully nose up position under aerodynamic loads thereby increasing the aircraft rate of pitch, nose down.

(v) The right hand stabilizer rear spar top chord had failed prior to the accident flight as a result of long term fatigue damage. The fatigue crack had existed for about 7,200 flights, of which approximately 6,750 flights were made when the aircraft was on the US register.

(vi) Following the failure of the stabilizer rear spar top chord the structure could not sustain the flight loads imposed upon it long enough to enable the failure to be detected by the then existing inspection schedule. It cannot, therefore be classified as failsafe.

(vii) Insufficient consideration had been given at the design and certification stage to the stress distribution in the horizontal stabilizer spar structure following a top chord failure in the region outboard of the closure rib.

(viii) The replacement of the horizontal stabilizer light alloy top skin by stainless steel significantly altered the stiffness distribution of the structure, creating the high fastener loadings which led, ultimately, to the fatigue failure in the rear spar top chord in G-BEBP.

(ix) Neither the inspections detailed in the approved maintenance schedule nor those recommended by the manufacturer were adequate to detect partial cracks in the horizontal stabiliser rear spar top chord, but would probably have been adequate for the detection of a completely fractured top chord.

(x) The inspections required by the Dan-Air UK CAA approved maintenance schedule in respect of the stabilizer rear spar top chord were less specific than those recommended by the manufacturer.

(xi) No fatigue tests were carried out on the 707-300 series horizontal stabilizer structure prior to USA or UK certification. Neither at the time of certification nor at the time of writing were such repeated load tests required by either US or UK legislation for structures declared to be failsafe.

(xii) A post accident survey of the 707-300 fleet, world-wide, revealed a total of 38 aircraft with fatigue cracks present in the stabilizer rear spar top chord. Of this number four stabilizers required chord replacement.

(xiii) Post accident flight tests revealed that deployment of speed brakes during the landing roll produced an horizontal stabilizer load condition spectrum which was significantly different to that used in the original design.

(*b*) *Cause*

The accident was caused by a loss of pitch control following the in-flight separation of the right hand horizontal stabilizer and elevator as a result of a combination of metal fatigue and inadequate failsafe design in the rear spar structure. Shortcomings in design assessment, certification and inspection procedures were contributory factors.

4. Safety Recommendations

4.1 When an aircraft has been certificated against failsafe criteria, those portions of structure considered significant in the failsafe design should be identified in the approved inspection schedule and their relative importance defined.

4.2 Defects involving the structure identified in recommendation 1 should be reported to the manufacturer and airworthiness authority to enable a "fleet picture" to be developed and appropriate action to be taken.

4.3 When an aircraft design has evolved significantly, a critical assessment should be made by the Certification Authority in association with the manufacturer as to the validity of extrapolation of the original design data when applied to a developed aircraft variant.

4.4 After an aircraft type has been established in service for some years a searching "mid-life" review should be carried out by the manufacturer to the satisfaction of the Certification Authority when all aspects of the aircraft's structural performance should be studied in the light of fleet service experience and current design knowledge.

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