Accidents Investigation Branch

Department of Transport

Report on the accident to Boeing Vertol (BV) 234 LR G-BISO in the East Shetland Basin of the North Sea on 2 May 1984

LONDON
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List of Aircraft Accidents Reports issued by AIB in 1986/87

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Department of Transport
Accidents Investigation Branch
Royal Aircraft Establishment
Farnborough
Hants GU14 6TD

7 October 1987

The Rt Honourable Paul Channon
Secretary of State for Transport

Sir,

I have the honour to submit the report by Mr D F King, an Inspector of Accidents, on the circumstances of the accident to a Boeing Vertol (BV)234 LR G–BISO which occurred in the East Shetland Basin of the North Sea on 2 May 1984.

I have the honour to be
Sir
Your obedient Servant

D A COOPER
Chief Inspector of Accidents

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Accidents Investigation Branch

Aircraft Accident Report No. 5/87
(EW/C869)

Registered owner and operator:		British Airways Helicopters (BAH) Ltd
Aircraft:	*Type:*	Boeing Vertol (BV) 234 LR
	Nationality:	British
	Registration:	G–BISO
Place of Accident:		In the East Shetland Basin of the North Sea, 8 miles west north west of the Cormorant Alpha Rig Latitude 61° 10′ N Longitude 00° 52′ E
Date and time:		2 May 1984 at 1241 hrs All times in this report are UTC

Synopsis

The accident was notified to the Accidents Investigation Branch (AIB) by the operator and an investigation began the same day.

The aircraft was engaged on a flight from the Polycastle Rig in the Magnus Field to Aberdeen carrying a full load of 44 passengers, one cabin attendant and two flight deck crew. Shortly after establishing in the cruise at 120 knots (kt) a violent disturbance was experienced. There then followed a series of disturbances, with changes in aircraft attitude, normal acceleration ('g') and rotor speed, associated with fluctuations in the No 2 flight control hydraulic boost system pressure.

After implementing the 'Emergency Check List Drill, Flight Boost Hydraulic Pressure Low' and attempting to regain control of the aircraft by changes in speed and height, the crew elected to ditch, fearing all control might be lost. A successful ditching was achieved 8 miles west north west of the Cormorant Alpha Rig. With the aircraft on a north easterly heading, into wind and swell, an attempt was made to water taxi towards the North Cormorant Rig. However, when the aircraft was found to be taking on water and sinking, an evacuation of the passengers commenced. After all the passengers and the cabin attendant were clear, the engines and rotors were stopped, following which the two pilots evacuated the aircraft. Shortly afterwards the aircraft capsized. It remained floating inverted until its recovery. All crew and passengers were rescued by other helicopters or surface vessels.

The report concludes that the accident was caused by two separate flying control system malfunctions that produced intermittent loss of collective control. Lack of floatation integrity led the aircraft to capsize after ditching.

1. Factual Information

1.1 History of the flight

The aircraft was allocated to operate a service from Aberdeen to the Magnus Field in the East Shetland Basin of the North Sea and return. The outbound flight was uneventful and the aircraft landed on the Polycastle Rig, adjacent to the Magnus Rig, at 1210 hrs. The passengers were disembarked and additional fuel was taken on to bring the fuel load up to 8000 lb. The refuelling was supervised by the cabin attendant and at the completion of this she prepared the cabin for the return flight, having made an external inspection of the aircraft during which nothing untoward was evident. A full load of 44 passengers was then embarked and briefed by the cabin attendant.

The aircraft took off at 1220 hrs and climbed to 2500 feet. The weather was good with little cloud, 30 kilometres visibility and no turbulence. The aircraft was established in the cruise flying smoothly at 120 kt with the Automatic Flight Control System (AFCS) and the Barometric Height Hold engaged, with the Tactical Air Navigation System coupled to the Navigation mode. The Longitudinal Cyclic Trim (LCT) and Differential Air-Speed Hold (DASH) cockpit indicators showed the actuators were in their normal operating positions. There were no malfunctions indicated in the cockpit.

Soon after 1230 hrs a violent disturbance was felt. The aircraft pitched slightly nose down with a distinct sensation of negative 'g', immediately followed by a harsh increase in positive 'g' and violent upward movement of the whole aircraft with the nose rising and the sound of decreasing rotor speed. This was accompanied by the illumination of an unidentified annunciator caption and a master caution light. The handling pilot reduced collective pitch and controlled aircraft pitch attitude instinctively. The rotor speed then rose rapidly and after reaching higher than the required value settled down at the cruise setting of 98%, following which everything appeared to be normal for a short time. There then followed, over a period of approximately five minutes, a series of similar disturbances during which it was possible for the crew to correlate the violent vertical acceleration and rotor speed variations with fluctuations in the No 2 flight control hydraulic boost system pressure indications. The pressure was seen to drop to approximately 300 pounds/square inch gauge (psig) accompanied by the associated annunciator panel captions, and then recover to the normal 3000 psig. Rotor speed was observed as low as 91% and on other occasions at over 108%.

The first officer, who had been the handling pilot throughout the flight, uncoupled the flight director immediately after the first disturbance and put the aircraft into a descent, reducing speed to approximately 80 kt. The commander, in accordance with Emergency Check List Drill No 43 'Flight Boost Hydraulic Pressure Low', selected the flight control system (FCS) boost switch to Boost 1, thereby switching off No 2 system. At the moment he did this there was another violent disturbance so he immediately selected both systems back on and control of the aircraft was regained. He then switched off No 2 AFCS and confirmed that the relevant channel warning lights were indicating off. The aircraft did not respond in an unusual way and the No 2 AFCS remained off for the remainder of the flight. However, the disturbances continued and the commander, with reference to the Emergency Check List, completed Drill No 43 again. When he selected the FCS boost switch to Boost 1 there was another violent disturbance. As previously, he immediately selected both systems on, restoring control of the aircraft, and the FCS boost switch was not operated for the remainder of the flight.

The aircraft was by now approximately level at 500 feet and Shetland Radar had been informed of the emergency. The commander took control while the first officer confirmed that the emergency drills had been carried out correctly. The first officer then resumed control of the aircraft. There then followed approximately two minutes during which the aircraft flew smoothly and steadily and all cockpit indications were normal. Viking Approach were informed of the problem and agreed that the aircraft should continue under the control of Shetland Radar. In the absence of any continuing malfunctions the commander decided to divert to Sumburgh. A second series of similar disturbances then occurred during which the first officer felt that the amount of control he had over the aircraft was diminishing, particularly in collective pitch. On one occasion a steady application of collective pitch to arrest a rate of descent of 800 feet per minute had no effect on the rotor speed, torque, or the aircraft, and it was as if the collective lever was disconnected. One disturbance followed another so closely that the first officer described it as a continuous series of recoveries from unusual attitudes.

It became clear that there was little alternative to a precautionary landing on the sea. A gentle touchdown was achieved, in spite of the control difficulties, 8 miles west north west of the Cormorant Alpha Rig, just before 1241 hrs. The wind was 10-15 kt from the north east with a sea swell of 2-3 feet. When on the water the engines and rotors were kept running and the collective lever was lowered, but the aircraft disturbances continued although their severity was greatly reduced. After a short time the No 2 hydraulic system pressure was lost completely and the disturbances ceased.

The first officer turned the aircraft onto a north easterly heading, into the wind and swell and started water taxying slowly towards the North Cormorant Rig, which was visible on the horizon. Waves were breaking onto the nose of the aircraft and occasionally came halfway up the pilot's windscreen. At one stage the first officer turned the aircraft 40° to the right of the wind to see if this would provide better conditions for launching the liferafts from the right side. However, the aircraft started to roll an estimated ± 10° and the blades of the forward rotor could be seen disturbing the water as they passed close to the surface. The aircraft was therefore turned back into wind. At 1257 hrs the crew experienced further difficulty with control and requested a rescue Bell 212 hovering nearby to confirm that there was adequate clearance between the rear rotor and the water. The Bell 212 reported that the rear rotor appeared to have twice the clearance of the forward rotor.

At 1323 hrs the evacuation of passengers commenced when it became evident that the aircraft was shipping water from the aft end and sinking. The occupants reported that flooding was deeper at the rear of the cabin than elsewhere, with the water eventually rising above the level of the seat arms. By 1350 hrs all the passengers and the cabin attendant were clear. The engines and main rotors were stopped and the two pilots were clear of the aircraft by approximately 1401 hrs, leaving the auxiliary power unit (APU) running. The aircraft capsized about two minutes later, 82 minutes after touchdown. It remained floating inverted until it was recovered by a surface vessel later the same day.

1.2 **Injuries to persons**

Injuries	Crew	Passengers	Others
Fatal	—	—	—
Serious	—	—	—
Minor/none	3	44	

1.3 **Damage to aircraft**

The aircraft was not damaged by the emergency landing but sustained some damage during recovery and severe damage from the effects of saltwater immersion.

1.4 **Other damage**

None

1.5. **Personnel information**

(a) Commander:	Male, aged 43 years
Licence:	Airline Transport Pilot's Licence (Helicopters) valid until 29 March 1986
Helicopter type ratings:	Bell 47, Sikorsky S61N, Boeing Vertol Chinook (BV 234)
Instrument rating:	Renewed 20 January 1984
Medical Certificate:	Class 1 issued 29 February 1984
Certificate of Test:	Boeing Vertol Chinook, 20 January 1984
Flying experience:	Total all types : 6025 hours Total Boeing Vertol Chinook : 2231 hours Total flying last 28 days : 53 hours
Duty time:	Off duty 1500 hrs 1 May until 0830 hrs 2 May (17 hours 30 minutes) On duty 0830 hrs 2 May until 1401 hrs 2 May (5 hours 31 minutes)
(b) Co-pilot:	Male, aged 42 years
Licence:	Airline Transport Pilot's Licence (Helicopters) valid until 13 April 1986
Helicopter type ratings:	MBB 105, Sikorsky S61N, Boeing Vertol Chinook (BV 234)
Instrument Rating:	Renewed 19 April 1984
Medical Certificate:	Class 1 issued 24 April 1984

Certificate of Test:	Boeing Vertol Chinook, 19 April 1984
Flying experience:	Total all types : 4630 hours Total Boeing Vertol Chinook : 437 hours Total flying last 28 days : 27 hours
Duty time:	Off duty 1600 hrs 1 May until 0830 hrs 2 May (16 hours 30 minutes) On duty 0830 hrs 2 May until 1401 hrs 2 May (5 hours 31 minutes).

1.6 Aircraft information

1.6.1 Leading particulars

Manufacturer:	Boeing Vertol Company, Philadelphia, USA
Type:	BV 234 LR
Date of Manufacture:	1981
Certificate of Airworthiness:	Transport Category (Passenger) issued 9 November 1983 valid to 8 November 1984
Certificate of Maintenance:	Issued 19 February 1984. Valid until 22 June 1984 or 4519 total airframe hours
Total airframe hours at time of accident:	4244 hours
Maximum total weight authorised:	48,500 lb (22,000 kg)
Estimated weight at time of ditching:	42,641 lb (19,342 kg)
Centre of Gravity (CG) range applicable:	313.3 to 336.8 inches aft of datum
Estimated CG at time of ditching:	327.5 inches of datum

The BV 234 (Fig 1) is a twin engined, twin rotor civil version of the CH-47 Chinook, of which approximately 1000 have been built for military use. However, the BV 234 has a longer nose, wider fuselage pods and flight control system differences. The BV 234 LR is the long range version.

1.6.2 Maintenance aspects and component histories

Documentation indicated that the following maintenance actions relevant to the aircraft defects found (para 1.12.2) had been accomplished in accordance with Civil Aviation Authority (CAA) Approved Maintenance Schedule MS 8000 Issue 1 at the noted times. Hours refer to airframe flying hours:

(a) Ramp Check (Daily) – accomplished on the day of the accident 2.8 hours prior to the accident. The check included an inspection of all Lower Boost Actuators (LBAs) for hydraulic fluid leakage and popped jam indicator buttons, and of Upper Boost Actuators (UBAs) for cleanliness.

(b) Check B (100 hour check) – carried out 10 days/50.6 hours prior to the accident. This check included a jam simulation test on both systems of each LBA.

(c) Task C2 (300 hour check) – carried out 27 days/135.3 hours prior to the accident. This check included a 'general condition check of ramp and hinges'. In this context ramp refers to the cargo loading ramp.

LBAs and UBAs were subject to a life evaluation programme controlled by the results of detailed overhaul type inspections of lead time actuators from the fleet at hard time thresholds. Passage beyond the threshold depended on the results of the sample overhaul. The threshold at the time of the accident was 3500 hours.

The known history of relevant components is summarised below, hours refer to component flying hours:

(i) Roll LBA (Serial No (SN) 0013T) – Time since new 2683 hours. Main ram cap screws were replaced (para 1.16.1) by Boeing Vertol during unit repair 1359 hours prior to the accident.

(ii) Pitch LBA (SN 0012) – Time since new 2550 hours. Some or all jam indicator screws were robbed 13 February 1983 for use on another unit. Following screw replacement the unit was installed on G–BISO 1358 hours prior to the accident.

(iii) Yaw LBA (SN 0015 – Time since new 2185 hours. Removed from G–BISR at 1828 hours from new with two fractured System 1 cap screws (para 1.17.1). Repaired and installed on G–BISO 1357 hours prior to the accident.

(iv) Collective LBA (SN 0005) – Time since new 1966 hours. Installed on G–BISO 1359 hours prior to the accident.

Records indicate that the pitch, yaw and collective LBAs were either in process of repair or in storage during late 1982 and early 1983 when cap screws on LBAs installed in the fleet's aircraft were replaced in accordance with manufacturer's recommendations (para 1.17.1). Neither the unit manufacturer nor the operator employed a system to record this replacement in these circumstances, except by the addition of a 'T' suffix to the unit data plate. None of these three units had the 'T' suffix present, suggesting that the replacement was not accomplished on any of them. However, this could not be established positively as the initial telexed Service Bulletin (8-1420-3-4383) from Boeing Vertol did not require the addition of the 'T' suffix. The roll LBA did have a 'T' suffix.

The evidence indicated that both the cargo loading ramp seal and the cargo door seal were original.

1.6.3 *Existing defects*

Aircraft documentation did not suggest that any known relevant deficiencies had been carried forward at the time of the accident. However, an entry in the Technical Log 5 days/22 hours before the accident read:

> 'With No 1 hydraulics ON and No 2 system OFF there is a restriction 3 inches above the detent and the collective becomes very stiff to raise in comparison with No 2 hydraulics selected ON and No 1 system OFF.
>
> Action taken: Controls functioned – found acceptable after controls and systems inspections and checked'.

1.6.4 *Flight Control System Description*

(a) Hydraulics

The BV 234 is provided with two separate flight control hydraulic boost systems (Fig 2). Each comprises a reservoir of 0.66 US Gallon (US gal) nominal capacity mounted in the aft transmission pylon and pressurised by engine bleed air, supplying a 3000 psig pump driven by the accessory gearbox. The output from each pump pressurises a 0.1 US gal accumulator and supplies one half of each of the flight control system dual actuators, after pressure reduction to 1500 psig in the case of the lower actuators. Flight deck control is by a three position switch captioned Boost 1 – Both – Boost 2, which functions by depressurising the non-selected system in the Boost 1 or Boost 2 position. Flight deck indication is limited to a pressure gauge and annunciator caution lights for low pressure and overtemperature for each system.

(b) General

The cockpit dual flying controls are conventional. Pilots' demands in each of the four channels (pitch, roll, yaw, collective) are transmitted mechanically by rod and bellcrank linkage to the input lever on an LBA located in the flight control closet behind the cockpit. Each LBA output lever connects via a mixing unit to input levers on the two UBAs on each rotor head controlling blade pitch angle. Limited authority blade angle control is also exerted, normally automatically, by an LCT system. AFCS inputs are made via a DASH actuator in the pitch channel upstream of the LBA and by dual extensible links in pitch, roll and yaw channels downstream of the respective LBA, acting in series with the mechanical linkages. Trim and spring feel systems are incorporated in each channel between cockpit and LBA, together with a rotor speed droop eliminator in the collective channel providing rotor governor bias as a function of collective demand. Balance springs act on the runs to counteract linkage weight.

(c) LBA description

LBAs are provided in order to reduce cockpit control forces and prevent transmittal of loads in the linkage downstream of the LBA back to the cockpit controls. They are manufactured by Hydraulic Units Inc of Duarte, California. The pitch, roll and yaw actuators are identical (Part Number (PN) 234HS 560-1), and the collective actuator (PN 234HS 560-2) differs only in its linkage geometry and main servo valve flow gain. Each LBA comprises a 1500 psig dual hydraulic servo ram (Fig 3) with features aimed at allowing continued operation in the event of a servo valve spool jam.

The two halves of each LBA are similar. With the shut off valve (SOV) in its normal open position, pressurised fluid enters the unit half, flows into and through the SOV, and then passes to the servo valve and to the bypass valve, which closes against its spring. Servo valve spool displacement in response to input lever movement ports pressurised fluid to one end of the main ram cylinder and ports the other end to return. Resultant ram piston rod displacement drives the output lever and also provides mechanical feedback to cancel the input signal when input and output lever positions match. The servo valve spools are held in contact with the servo valve lever cam by bias springs (nominal preload 0.5 lb). Servo valve gain is 10^6 psi/inch. Most of the operating parts of the servo valve, bypass valve and SOV are manufactured from nitralloy and 52100 tool steel. Both materials are very hard, non-corrosion resistant steels.

In the event of a servo valve spool jamming in its sleeve, input lever movement causes displacement of the sleeve against centring springs, (nominal pre-load 30 lb) thereby porting pressurised fluid to the SOV spool causing it to close against spring force. The resultant cut-off of pressurised fluid causes the bypass valve to open under spring loading and connect both cylinders of the respective side of the actuator to return. Thus the input lever remains free to move in response to pilot demands, hydraulic locking of the disabled main ram piston is eliminated, and the unjammed half of the unit continues to operate. The SOV is designed to hydraulically latch closed by means of direct porting of system pressure (Fig 4, Pressure A) into the SOV actuating pressure chamber (Pressure D) via a cut-out in the spool inner land uncovered when the spool has partially closed. This maintains the actuator half in bypass until the hydraulic boost system is depressurised and repressurised. The servo valve sleeve can also be manually displaced by depressing a jam simulation button on each unit half in order to check the jam protection function. SOV operating pressure is also supplied to a pop-up button type jam indicator on each unit half, manually resettable after cycling actuator supply pressure. The indicator incorporates provision for a microswitch to operate a flight deck indicator, but none was provided. Relevant actuating pressures are:

Less than 48 psig	–	SOV fully open
More than 263 psig	–	SOV fully closed
More than 250 psig	–	Bypass valve fully out of bypass
1100–1200 psig	–	Jam indicator trips

In a non-jam condition leakage into the SOV actuating pressure chamber is dissipated to return via a 0.016 inch diameter screened bleed orifice.

Relevant SOV (Revision C) design dimensions are, (Fig 4):

Axial width of spool inner land (x in Fig 4)	0.190 ± 0.007 inches
Axial width of sleeve inner port (y in Fig 4)	0.180 ± 0.004 inches
ie, Spool overlap	0.010 ± 0.011 inches

In the event of both halves of the LBA entering bypass mode, ram stiffness would be lost, allowing input and output levers to move independently of each other until the LBA linkage bottomed on the valve lever stops provided on the body. Measured control forces in the single and double bypass modes are given in para 1.16.4. Pre-flight control checks utilising each hydraulic boost system in turn constitute the only means available to the crew of detecting that one half of an LBA is in bypass. Abnormal control forces and/or lack of blade movement in response to cockpit control movement over part of the control travel would be experienced when the boost system associated with the non-bypass half of the actuator was depressurised. The relevant part of the Starting Engines Check at the time of the accident was as follows:

> 'Flight controls travel and Flight Boost systems – Check:
>
> a. Check for full travel in each axis with no evidence of binding or restriction with FCS switch at BOOST 1, then BOOST 2.'

The outer end of each LBA main ram outer cylinder is sealed by a 1.10 inch outside diameter gland (Fig 5) carrying static and dynamic O-ring seals and retained in the body by a triangular cap plate. The 0.275 inch thick cap plate is retained by three 3/16 inch diameter allen headed screws passing through it and screwed into thread inserts in the cylinder end flange. Each screw is wire locked to the body. No screw washers or cap gaskets are used. One screw, designated No 1, is located on a 0.118 inch thick recessed portion of the cap and is situated nearer the cylinder axis than Nos 2 and 3 screws. Hence it reacts approximately 40% of the gland load compared to 30% for each of the other two screws. The load on the gland with full system pressure of 1500 psig in the outer cylinder is approximately 1025 lb. Cylinder pressures to generate a given output lever load are dependant on the degree of mismatch between the two sides of the actuator, ie, lack of unison of the two servo valves.

At full ram travel the trunnion on the ingoing end of the ram contacts the respective cap and stops the ram before the piston in the other half of the actuator contacts the gland. Thus ram overtravel is not reacted by the cap screws. At the time of the accident, alloy steel (Specification FF–S–86) cap screws to Specification NAS 1351–3H10P, with cut threads were specified, with an installation torque (oil lubricated) of 30-35 inch lb.

1.6.5 *Floatation and stability*

In the event of ditching, the fuselage and fuel pods are intended to provide buoyancy and stability. No additional floatation devices are installed. Each production fuselage is tested for water tightness by the manufacturer during final assembly, before fuel pod installation, by submerging it in a tank to a representative maximum gross weight waterline. Maximum allowable total leakage rate is specified as 36 US gal/hour, with individual limits for various sections.

The mainly submerged gap between the hinged cargo loading ramp and its fuselage aperture is provided with a bulbous U-section rubber seal bonded along the sides and forward edge of the aperture. A further seal, reportedly designated as an aerodynamic rather than a water seal, fits in the gap between the cargo loading ramp and the cargo door.

CAA validation of BV 234 Federal Aviation Administration (FAA) Type Certification utilised predictions based on tank testing of a scale model by British Hovercraft Corporation (BHC) that capsize should not occur below Sea State 5. The BHC test report noted that, for all test conditions, the capsize boundary generally lies on a wave height/length ratio between 1:9 and 1:10 'which provides an excellent margin of stability, well in excess of the minimum requirements of the British Civil Airworthiness Requirements (BCARs)'. The model was fitted with static rotor blades. Possible effects of wave impact by rotating blades were not included in the testing, but reportedly formed part of the assessment, although no documentary evidence of this was obtainable.

The Approved Flight Manual and Operations Manual stated in part:

> 'The helicopter fuselage and fuel cell compartments are sealed to provide floatation for an indefinite period. The effective beam and buoyancy of the fuselage and fuel tanks provide stability in heavy sea state conditions. . . .Forward rotor clearance is reduced; however, wave to rotor clearance can be maintained with longitudinal control . . . Rotor shutdown can be accomplished in heavy seas, with minimum risk, by reducing rotor rpm to a value where good control is maintained, and using the action of the waves. As the nose rises on an incoming wave, quickly secure the engines and arm and apply the rotor brake so that the rotors are stopped before impact with the next wave.'

1.6.6 *Safety equipment*

(a) Emergency exits

Extracts from the BCARs, valid at the time of the accident, concerning the requirements for emergency exits and ditching exits are at Appendix 1. The aircraft passenger cabin was fitted with two emergency exits at the front and two at the rear. The forward right exit consisted of the normal entry/exit door, 66 inches high by 36 inches wide (BCAR Type I) in the upper part of which was fitted a jettisonable ditching exit 36 inches high by 24 inches wide (Type III). The forward left emergency/ditching exit situated immediately to the rear of the toilet compartment, was 38 inches high by 20 inches wide (Type III). At the rear of the cabin one 26 inches high by 19 inches wide (Type IV) jettisonable exit was fitted each side alongside the rear rows of seats.

(b) Liferafts

The aircraft was equipped with two RFD Type 30 Mk 4 liferafts with a service capacity of 32 persons each. Each liferaft was contained in a valise measuring 38 x 24 x 8½ inches and weighed 140 lb. They were stowed under the forward row of seats on either side of the aisle. Each liferaft was provided with a painter, the end of which was secured within the underseat storage space. Ditching drills required the liferafts to be deployed manually through the forward exits. A cleat for securing the painter was provided on the outside of the fuselage near the top aft corner of each forward ditching exit.

(c) Life saving jackets and immersion survival suits

Each passenger was provided with a life saving jacket (LSJ) worn throughout the flight, fastened around the waist in a valise on top of a survival suit. The survival suit was a one-piece waterproof garment worn over normal clothing. Each crew member was provided with an immersion suit and an LSJ fitted with a search and rescue beacon (SARBE). The rules for wearing this equipment were stipulated in the Operations Manual, and in the conditions prevailing they were not required to be worn.

(d) Handholds and safety lines

There was no requirement for handholds or safety lines on the aircraft and none were provided.

1.7 Meteorological information

Synoptic situation:

A north easterly airstream covered the area. Pressure was falling but no marked changes were taking place.

Actual weather conditions (provided by an aftercast):

Winds and temperature	–	500 feet 040°/10-15 kt +9°C 2000 feet 040°/10-15 kt +9°C
Cloud	–	3 oktas cumulus/stratocumulus, base 1500 feet, top 2500 feet.
Visibility	–	Mainly greater than 15 kilometres.
Weather	–	Small probability of isolated showers, otherwise nil.
Sea state	–	Fairly smooth with a swell of 2-3 feet and a 7 second period. Sea temperature +8°C.

1.8 Aids to navigation

Not applicable.

1.9 Communications

For the initial part of the return flight to Aberdeen the aircraft operated under the control of Viking Approach. Immediately prior to the first in-flight disturbance communications had been established with Shetland Radar. The aircraft informed Viking Approach that they had a problem soon after it had occurred and the request to continue with Shetland Radar was approved by Viking Approach.

The Viking Approach frequency was recorded and a transcript of the communications on this frequency was completed by the Scottish Field Services Transcription Unit. The Shetland Radar frequency was normally recorded but the recorder was not operating between 1225 hrs and 1243 hrs. The distress calls made by the aircraft were therefore not recorded until after the aircraft had alighted on the surface.

1.10 Aerodrome information

Not applicable.

1.11 Flight Recorders

No recorders were fitted and none were required by UK Regulations. With effect from June 1984 the Air Navigation Order required Transport Category Helicopters with a maximum total weight authorised exceeding 2700 kg or the capability of carrying more than 9 passengers to be fitted with a 4 channel cockpit voice recorder with one channel recording rotor rpm.

1.12 Wreckage and impact information

1.12.1 Recovery

The aircraft remained afloat inverted and was lifted on board an oil support vessel by crane 9 hours after touchdown, using a strop around each rotor head.

1.12.2 Examination

An AIB representative arrived on board the recovery vessel the following day and conducted an initial examination en-route to Aberdeen Docks. Subsequent examination was carried out at Aberdeen Airport.

The evidence indicated that the aircraft did not sustain damage from the water landing but it was subsequently almost totally immersed in seawater. Relatively minor damage to both rotor heads, slight fuselage deformation and holing of the fuselage left side pod resulted during recovery.

Relevant cockpit and system indications and settings found were as follows:

Engine condition levers	–	Both at STOP
Engine fire handles	–	Both pulled
Rotor brake arm switch	–	ARM
Rotor brake lever	–	ON
Generator selector switches	–	Both ON
Transformer rectifier unit selector switches	–	Both ON
Battery selector switch	–	ON
APU selector switch	–	RUN
Flight control boost switch	–	BOTH

Hydraulic pressure gauges:

Utility	–	3120 psig
Flight control boost No 1	–	3000 psig
Flight control boost No 2	–	0 psig

Hydraulic boost system reservoir contents (sight glass):

System 1	–	Approximately normal level, but included some seawater
System 2	–	At or near empty
LBA jam indicators	–	All in (ie, not tripped)
UBA jam indicators	–	All in (ie, not tripped)

A detailed investigation was made of flight control systems, safety equipment and fuselage features relevant to floatation integrity. The following defects were found.

1.12.2.1 *Lower boost actuator cap screws*

Two of the three cap screws in the System 2 half of the roll LBA were found fractured, although the heads were retained in place by the locking wire, and fluid was seen to be seeping past the associated gland with the boost system depressurised. With 1000 psig System 2 inlet pressure applied to the actuator after its removal a dribble of fluid leaked past the gland when the input lever was stationary, increasing to 0.1 US gal/min when the input lever was cycled. The Serial Number of this LBA included a 'T' suffix (para 1.17.1). Detailed investigation was carried out at the LBA manufacturer's facility under AIB supervision. This showed that the LBA had operated for a period with the System 2 gland cocked and that the associated static O-ring was broken in two. Metallurgical examination of the System 2 cap screws at the Royal Aircraft Establishment, Farnborough and Boeing Vertol Materials Laboratories showed that Screws 1 and 2 (Fig 5) had each failed after extensive propagation of multi-origin fatigue cracks. The Screw 1 fracture face, just below the head, exhibited 93% fatigue, while the Screw 2 fracture face, close to the LBA body flange face, exhibited 18% fatigue. Screw 3, which was bent but did not fail, had a fatigue crack extending over 10% of its cross section, again at a position close to the LBA body flange face. The screws conformed to specification except that complete decarburisation of the threads was found on Screws 1 and 3; laps (small folds in the material surface) were found at and below the pitch diameter on Screw 2; and microcracks were found in the thread roots of Screw 3.

Following cap screw and gland replacement both halves of the LBA were rig functionally checked and partially stripped. No evidence was found of additional faults except for corrosion of System 1 components (para 1.12.2.3). Subsequently cylinder pressures and/or output load characteristics during a number of events were also measured when the unit was utilised for ground and flight tests by the manufacturer to determine cap screw loading characteristics (para 1.16.1).

Incorrect screw preload was believed to be a possible factor in screw failure and so tightening torque checks on the other cap screws from G–BISO were made. Results are given in Fig 5. Break-out torque could not be established for around 40% of the intact screws as the allen key rotated relative to the screw head socket at a torque less than that required to drive the screw. This generally occurred at 50 inch lb or less. No evidence of gross undertorque was found, with the exception of the No 2 cap screw of System 2 of the collective LBA, which was found to have negligible tightening torque, implying little or no preload. Metallurgical examination of this cap screw revealed a severely distorted thread profile, together with laps on the thread flanks. Scanning electron microscope dispersive X-ray analysis showed that this screw was manufactured from a corrosion resistant steel not in accordance with the approved material specification for this application.

1.12.2.2 *Lower boost actuator bypass condition*

X-ray photographs of the collective LBA immediately after removal from the aircraft and before it was subjected to any testing or dismantling showed that the System 1 SOV was partially closed. The jam indicator was not tripped. A detailed examination was carried out at the LBA manufacturer's facility under AIB control. The SOV spool was found displaced approximately 0.06 inches from full open, out of a total stroke of 0.18 inches and was seized in this position. A total force of 17 lb in the SOV opening sense was required to dislodge the spool, which was found to be extensively coated between lands with tough gelatinous, rust-coloured deposits, with rust-like discolouration of much of the outer surface of one of the lands. After deposit removal the spool was found to be free to slide in its sleeve. SOV spring free length and spring rate (60.3 lb/inch) were within design limits. Rig functional checks were conducted to examine the operation of both halves of the LBA. With the System 1 SOV spool reinstalled but shimmed to the partially closed position found, the System 1 half of the LBA was found to be in bypass mode, with a pressure at the jam indicator port of 960 psig. The System 1 jam indicator operating pressure was within limits at 1180 psig. With the SOV spool installed unshimmed, normal behaviour of both LBA halves in normal operating mode was obtained. However, for both Systems, during initial jam simulation tests the simulation button would not remain depressed and the respective LBA half did not latch in bypass mode. Some correct results were obtained for both Systems in the course of a large number of jam simulation tests but not consistently. Both SOV spools were found to be of the Revision C type (para 1.17.2). The System 1 SOV was found to conform to nominal drawing dimensions, and spool overlap was thus 0.010 inch (para 1.6.4(c)).

On completion of functional checks the collective LBA was strip examined. No evidence was found of excessive leakage across servo valve spool or sleeve lands, or of blockage of the bleed orifice for the SOV actuating pressure chamber. The examination did reveal corrosion within System 1 (para 1.12.2.3), together with some scoring of the System 1 servo valve spool lands. No evidence was available to indicate when the scoring had occurred, or whether it reflected sufficient interference between spool and sleeve at some stage to have caused a jam.

1.12.2.3 *Hydraulic boost system corrosion*

Collective and roll LBA strip eight days after the accident revealed extensive corrosion and corrosion type deposits on System 1 components, particularly in the collective LBA servo valve and SOV and in the roll LBA servo valve, SOV and bypass valve. The pattern of corrosion in some areas had clearly been influenced by fluid flows through the unit and was not consistent with post-accident effects. No evidence of pre-ditching corrosion within System 2 was found. Partial strip of the pitch and yaw LBAs, both hydraulic boost system pumps and the three extensible links revealed signs of corrosion in both systems, but no evidence was found to suggest whether this had occurred before or after the ditching.

Fluid samples from both boost systems were analysed by the Directorate General of Defence Quality Assurance at Harefield, with particular attention to determining levels of chlorine and chlorinated solvents (eg, trichloroethane). These substances were not found in excessive concentrations. The fluid was consistent with Aeroshell 4, with the exception of free water found present in the samples from both systems, consistent with seawater having entered via reservoir pressurization systems while the aircraft was floating inverted. Further work directed at determining the cause of the corrosion found is described in para 1.16.2.

Corrosion products found within LBAs were analysed by Northrop Aircraft Materials Laboratory, Hawthorne, California. Iron oxide and significant concentrations of chlorine were found, together with some traces of potassium and zinc. This was consistent with the corrosion having been induced by either chlorinated cleaning solvents or saltwater.

1.12.2.4 *Lower boost actuator jam indicator mounting*

The pitch LBA System 1 jam indicator was found to be loose on its adaptor plate, with a gap averaging 0.021 inches between the indicator flange and adaptor plate surfaces. All four attaching screws were loose and could be turned between 50-180° clockwise with less than 2.5 inch lb torque before beginning to tighten. There was no provision for screw locking. Examination showed that the indicator flange and adaptor plate were held 0.017 inches apart by contact of the indicator diaphragm with the adaptor plate piston guide.

1.12.2.5 *Upper boost actuator leak*

Clear evidence was found that hydraulic fluid had leaked from the System 1 main ram gland of the swivelling UBA on the aft rotor head prior to the accident. Extensive streaking and gelatinous type deposits were present around the gland and on the outer surface of the System 1 cylinder. The operator reported that seepage from UBAs when unpressurised was not uncommon, but that normally no leakage occurred when pressurised. When rig tested after the accident the leak could not be reproduced.

1.12.2.6 *Aircraft floatation and stability*

The portion of the aircraft that would be submerged or remain only a short distance above the waterline after a water landing was examined in detail in an attempt to assess water-tightness. Visual inspection indicated a good standard of sealing of fuselage skin, including drain plugs, together with the radome door, cockpit exit doors, main cabin door, electrical bay access panels, external hook door, landing light bay and aerials.

However, a number of paths whereby water would, or possibly could, have entered the aircraft were found:

(a) Cargo loading ramp fuselage seal

This seal was found extensively damaged in the region of its right hand forward corner. A 7 inch long portion was missing, with the seal either side of this longitudinally split over a total distance of approximately

30 inches. The damage did not appear to be consistent with the effects of water egress during the recovery lift, although this possibility could not be dismissed.

It was also observed that with the ramp closed the lateral section of the seal contacted the forward top corner of the ramp, in contrast with the longitudinal side sections of the seal which were compressed against the flat face of the ramp top surface. This arrangement of the lateral seal appeared less positive, and the possibility that deformation by water pressure could have unseated it and allowed leakage, could not be dismissed. The distance of the lateral seal below the initial static waterline was estimated at 19 inches.

(b) Cargo door seal

This seal was severely damaged over its central areas with portions missing, rendering an approximately 26 inch length ineffective. The damaged portion was estimated to be 10 inches above the initial static waterline.

(c) Fuel tank pod interspace drains

Both 3/8 inch diameter threaded holes in the undersurface of each fuel tank pod (ie, four holes per aircraft) were found unblanked, in accordance with CAA requirements (para 1.17.3). These holes directly communicated the interspace between each fuel pod and the bladder tank within it to overboard. Forward and aft holes were estimated to be 18 inches and 23 inches respectively below the initial static waterline. The total fuel load at the time of ditching was estimated at 7200 lb, giving a fuel head at the holes that was approximately 20 pounds/square foot less than the head of seawater outside.

(d) Forward right hand landing gear torsion box drain

An open ¼ inch inside diameter pipe communicating the interior of the sealed torsion box to overboard was found. Access difficulties prevented complete torsion box examination, but it was reported that the torsion box interior communicated with the fuselage underfloor region via two drain holes. The highest portion of this flow path was estimated to be approximately 12 inches below the initial static waterline.

With the aircraft afloat, water would have entered the heater exhaust, heater drains, and engine and APU fuel supply line shroud drains within the cabin, and would have covered the fuselage sidewall cutouts (4 per aircraft) for fuel pod mounting beams. Access difficulties precluded complete inspection in some areas, but no evidence was found that any of these paths was open to the cabin interior below the initial waterline.

No evidence was found to suggest that rotor blades contacted the sea while rotating.

1.12.2.7 *Safety equipment*

(a) Liferaft mooring

Both liferaft painters were found severed. Evidence was found that both painters had chafed and cut on the corner of the forward right hand ditching exit aperture. The outer panel of the airstairs door which formed this surround was of kevlar reinforced plastic, with square, sharp edges. The aircraft ends of the painters were recovered attached to the front row seats and not to the mooring cleats.

(b) Emergency exit markings

Exits were marked internally by illuminated signs and decals. External decals marking emergency exits and providing operating instructions for these exits were severely faded and generally illegible, with the exception of two of the operating instruction decals. Of these, one was incorrectly located and not comprehensible in context, and the other was upside down.

1.13 Medical and pathological information

Not relevant.

1.14 Fire

No pre or post accident fire occurred.

1.15 Survival aspects

When, at 1241 hrs, the aircraft settled on the water, the three crew members donned their LSJs and the cabin attendant also removed the left hand liferaft from its stowage with the aid of two passengers. A little while later, while the aircraft was water taxying towards the North Cormorant Rig, the cabin attendant and the two pilots in turn put on their immersion suits. At 1257 hrs the first rescue helicopter, a Bell 212 based in the Brent Field, arrived on the scene, followed 20 minutes later by the first surface vessel.

By this time it was evident to the crew of G–BISO that the aircraft was taking on a lot of water and flooding by the stern and it was decided to evacuate the passengers. By 1323 hrs the first liferaft had been launched through the forward right ditching exit with the painter secured around the arm of one of the passenger seats. After some passengers had entered the liferaft through the forward right exit it was either dragged or blown out of reach. More passengers went through the rear right exit and clambered forward along the top of the sponson in order to reach the liferaft. Approximately 9 passengers had boarded when the painter parted allowing the liferaft to drift behind the aircraft. The second liferaft was also launched through the forward right exit and the painter similarly secured. Two passengers had entered this liferaft when its painter also parted and by 1329 hrs both rafts had drifted clear of the aircraft.

Between approximately 1332 and 1348 hrs the remaining passengers escaped through the rear right exit into the water and drifted behind the aircraft where they were picked up either by surface vessels or by one of 3 rescue helicopters. During this phase communications between the crew were made difficult because the intercommunication system was suffering from the ingress of sea water. However, the rate at which passengers left the aircraft was regulated to enable the helicopters and surface vessels to pick them out of the water with the minimum of delay. Following the evacuation of the cabin attendant at 1350 hrs the two pilots started the APU, shut down the engines and were able to stop the rotors without the blades striking the sea. They then escaped through the forward right exit at approximately 1401 hrs and the aircraft capsized about two minutes later.

After the rescue was complete a number of the passengers stated that their survival suits had leaked badly. There was no statutory requirement for passengers to wear survival suits and these did not form part of the aircraft or its equipment but were worn by passengers on the instruction of the charterer who also provided the suits. There was no formal specification which the survival suits were required to meet but the existing standard for military immersion suits specified a maximum 10 minute leak rate of less than 50 grams.

Of the 44 passenger survival suits in use on the accident aircraft, 40 were subsequently tested by the Royal Air Force Institute of Aviation Medicine. Of these only one suit had a leak rate of less than 50 grams/10 minutes and 21 suits had leak rates in excess of 1000 grams/10 minutes.

1.16 Tests and research

1.16.1 Lower boost actuator cap screws

A stress analysis of the gland retaining arrangements was carried out by the LBA manufacturer, including the effects of cap plate bending. The aircraft manufacturer conducted a series of bench fatigue endurance tests on NAS 1351 type screws, both in isolation and when installed in an LBA, together with some tests on the effects of screw overtorque. Both analysis and tests considered the effects on screw fatigue life of main ram cylinder pressure fluctuations being more severe than had been previously anticipated; of lack of screw preload; and of screw thread decarburisation. The aircraft manufacturer also conducted aircraft ground and flight tests, using G–BISO's roll LBA after appropriate refurbishment installed in the yaw channel, to determine main ram cylinder pressure and/or output load characteristics during various aircraft events and manoeuvres.

The analytical evaluation indicated that screw fatigue strength was marginal, with predicted stresses in the radius between screw head and shank close to yield when the LBA was subjected to proof pressure of 2000 psig. The testing clearly showed that lack of screw preload had a major detrimental effect on fatigue strength. In the most adverse conditions of LBA cycling over the full system pressure range (0-1500 psig), screws in the No 1 position typically failed after approximately 0.12×10^6 cycles, if untorqued; while no correctly torqued No 1 screws failed during a series of seven tests run for between $1–3 \times 10^6$ cycles. Where the tests were run with all three screws untorqued the No 1 screw invariably failed first, followed by one of the others after between $2–10 \times 10^3$ further cycles.

A degree of correlation between extent of screw thread decarburisation and fatigue strength was found, but the effect was minor. The limited testing on the effects of gross screw overtorque suggested that the screw would probably fail before the thread insert or actuator body was severely affected. No evidence was found of a gross overtorque influencing the subsequent capability of the insert to maintain screw preload.

Flight tests revealed greater main ram cylinder pressure fluctuations than anticipated in one side of the actuator, as a result of servo valve mismatch, ie, small differences in the relative position of servo valve spool and sleeve between the two sides. The evidence suggested that this resulted from small dimensional differences, together with the effects of one servo valve spool unseating from the input lever cam under dynamic conditions, ie, when a rapid control input was made. Both effects were likely to be present in all LBAs. Also apparent was an appreciable level of load fluctuation at a frequency of 3/rev (three times rotor frequency) in LBA output linkages in some conditions, apparently generated by the effects of airframe vibration. Inadequate information was available to enable estimation of the resultant effects on cap screw loading, but the manufacturer assessed that significant fatigue damage from this cause would be very unlikely.

1.16.2 *Hydraulic system corrosion*

The most likely cause of the corrosion found on hydraulic boost System 1 components was considered to be contamination by chlorinated solvents, which in combination with minute amounts of water produces corrosive acids. Once corrosion is initiated chlorine ions released by the process, in combination with dissolved water present, can maintain the reaction. The possible ways in which water and chlorinated solvent cleaners could enter aircraft hydraulic systems were assessed, and hydraulic samples from the operator's ground servicing rig and the other five BV 234s in the operator's fleet were analysed. A fleet inspection for LBA SOV corrosion was also conducted.

The possibility of contamination of the reservoir of the operator's rig via an unsealed access panel incorporated in the flat top surface of the reservoir was identified. It was also considered that condensation in the reservoir was a possible source of water contamination, particularly as reservoir contents had to be kept low when the rig was not in use to allow for the emptying by gravity of the aircraft reservoir into the rig reservoir that occurs when the rig is in use.

A number of chlorinated solvents were widely used by the operator for cleaning purposes, as is normal practice. Solvent constituents generally consisted of about 95% of 1.1.1 trichloroethane, plus an inert stabiliser. The only possible means of significant water ingress to aircraft hydraulic systems in service appeared to be via reservoir pressurization lines when precipitation, spray, or engine wash water passed through the engines. Available evidence did not suggest that appreciable quantities were entering the systems in this way, at least not on a frequent basis.

Concentrations of water and chlorinated solvents were well below allowable levels in initial rig samples, but significantly above in samples taken 3 weeks after the accident, at 675 parts per million (ppm) of water and 400 ppm of trichloroethane, compared to reject levels of 150-200 ppm and 300 ppm respectively. No unacceptable water concentration levels were found in samples from the 10 other systems of the BV 234 fleet, but 5 had trichloroethane levels at or above the reject threshold. Subsequent regular fleet checks generally showed acceptable contamination levels. No other cases of severe SOV corrosion in the fleet were found.

Threshold chlorinated solvent levels were understood to have been derived from work done by the United States Air Force (USAF) following cases of aircraft system malfunction as a result of corrosion of low chrome steel hydraulic components, particularly spool valves (Air Force Wright Aeronautical Laboratories Report AFWAL–TR–82–4027 of June 1982, unclassified). Substitution of hydrocarbon solvents for aircraft and rig hydraulic system cleaning was implemented. Available information suggested that chlorinated solvents could promote corrosion not only by their introduction in liquid form into hydraulic fluid, but also by leaving residues on cleaned components after drying.

1.16.3 *Mathematical model*

Aircraft behaviour with one side of the collective LBA in bypass mode and the other experiencing temporary loss of system pressure was investigated by the aircraft manufacturer using a computer model. For a 4 second loss of pressure with an assumed typical pilot response, aircraft parameters varied over the following ranges during the pressure fluctuation cycle:

Vertical Acceleration at cockpit	–	0.6–2.0g (absolute)
Pitch attitude	–	1.5° nose up to 3.9° nose down
Pitch rate	–	± 15°/second
Rotor RPM	–	91–107%

1.16.4 *Cockpit control force checks*

Cockpit control force checks in pitch, roll and collective channels were measured on a BV 234 at Aberdeen. The approximate force at the pilot's handgrip to displace each control in Mode 2 (ie, one half of LBA Normal and the other in bypass) and Mode 3 (ie, both halves of LBA in bypass) with the magnetic brake released was as follows. (On achieving Mode 3 in the pitch and roll channels the pilot's cyclic control stick will attempt to move forwards and to the left respectively. With the collective channel in Mode 3 the pilot's control lever will attempt to move down towards the detent.)

Pitch:	Mode 2	1–2 lb in both directions over full 13 inches of travel.
	Mode 3	1–2 lb aft over first 1.5 inches from full fwd. 4–6 lb aft over next 6 inches. 4–9 lb aft over next 5 inches.
Roll:	Mode 2	1.5–2 lb in both directions over full left-right travel of 9 inches.
	Mode 3	0–1 lb over first 1 inch from full left. 17–19 lb over next 8 inches.
Collective:	Mode 2	5–6 lb in both directions over the 7 inches of travel from the detent to full up.
	Mode 3	3–5 lb over first 2 inches up from detent. 16–18 lb over next 5 inches up from detent.

1.16.5 *Floatation and blade clearance*

Following the accident a number of tank tests and a demonstration river landing, including shutdown, were conducted by the aircraft manufacturer on a BV 234 with approximately 2 calendar years/500 airframe hours in service. This was done in order to verify floatation integrity and predictions of waterlines and rotor blade clearances. The manufacturer's findings were utilised in the assessment of G–BISO's watertightness and to make the following estimates of forward rotor blade clearance from a calm seawater surface for the following conditions:

Gross weight	–	42,500 lb
CG	–	Central (327.5 inches aft of datum)
Cockpit controls	–	Neutral (collective in detent)
AFCS	–	Off
LCT	–	Fully retracted

Minimum blade clearance:

Rotor speed 100%	– 70 inches
Rotors stopped, flying control hydraulic systems pressurised	– 37 inches
Rotors stopped, flying control hydraulic systems depressurised	– 25 inches

Clearance could be affected by a number of factors. Research uncovered little documentary evidence on which to base an assessment of the possibility and likely effects of a strike on the sea by a rotating blade. BCAR's do not specifically address rotating blade strike after ditching.

Features relevant to watertightness were also inspected on four other BV 234's. One exhibited cargo loading ramp seal damage; two showed cargo door seal damage; and on all four the forward seal portion contacted the ramp corner, rather than the face, all as for G–BISO.

1.17 **Additional information**

1.17.1 *Lower boost actuator cap screw background*

Original CH–47 Chinook main ram gland retention features were not comparable with those of the BV 234. For BV 234 Type Certification LBA cap screws were not considered to be in a fatigue application and fatigue qualification testing was not required. On 27 November 1982 two System 1 cap screws of a BAH aircraft's pitch LBA were found fractured after complete loss of System 1 hydraulic pressure. This led to discovery of an LBA manufacturer's assembly drawing error, whereby cap screw tightening torque was specified at 125-150 inch lb rather than 30-35 inch lb. The screw failures were attributed to the effects of overtorque, and the aircraft manufacturer commenced replacement and correct torquing of cap screws on all LBAs in its control (Critical Action Request Expedite CARE CD 234 H0005, 29 November 1982), and recommended operators to do the same by means of telexed Service Bulletin 234-65-1034 on 13 December 1982. The service bulletin was reissued by post on 2 February 1983. The final service bulletin issue also required the addition of a 'T' suffix to the unit's serial number, although the initial telexed issue did not. The change was incorporated on British registered aircraft by an operator's Special Check (No H–1303) raised for each aircraft, but clear provision for modification of uninstalled units did not exist. Compliance was not made mandatory by either FAA or CAA.

Fluid leakage from the roll LBA (SN 0024T) of a Norwegian operated aircraft on 6 February 1984 led to discovery of a fractured System 1 No 1 cap screw that had reportedly been replaced in accordance with the service bulletin. No further details of investigation into this case could be found.

Approximately 3 weeks after G–BISO's accident, the operator found one cap screw with 'zero torque' and another with damage to its head socket that prevented it from being effectively torqued (BAH Preliminary Defect Report 25–5–84, SN 0059). Both defects were on an LBA being prepared for installation on an aircraft and were found during cap screw replacement in accordance with Service Bulletin 234–65–1034. The operator reported that the unit had been received shortly before from Boeing Vertol with all cap screws wirelocked and sealed.

A second BAH Preliminary Defect Report (30–5–84, SN 0056T) indicated that shortly afterwards a new LBA drawn from the operator's stores was found prior to installation to have four cap screws torqued to only 3 inch lb.

1.17.2 *LBA SOV background*

The LBA anti-jam feature was required by the Certificating Authorities to be provided for the civil version of the Chinook as an added safety feature, although military experience had not shown jamming to be a problem, and therefore CH–47 service experience was in general not relevant to this system. The design of the SOV spool (Revision C) had a vee notch in the inner land for porting pressurised fluid to latch the spool closed after a servo valve jam (Fig 4). Modified spools (Revision D) with the vee notch replaced by a flat were incorporated in a number of LBAs in late 1983 after they failed to latch into bypass during the jam simulation portion of the manufacturer's production acceptance test. G–BISO's SOV spools were all of the Revision C type.

Three weeks after G–BISO's accident, revised pre-flight checks of the flight control system on another BV 234 revealed that the System 1 half of the pitch LBA was in bypass. The jam indicator did not trip. The actuator was replaced by a manufacturer refurbished replacement unit, which then failed to latch into bypass during a jam simulation test. Both these actuators had Revision C type SOV spools.

1.17.3 *Floatation background*

Removable blanking plugs were originally designed to be fitted in the two 3/8 inch diameter holes provided in the undersurface of each fuel pod for pod/bladder interspace overboard drainage. Permanent plug removal was required at CAA validation of FAA Type Certification in order to provide ready indication of bladder leakage. No documentary evidence was received to indicate what consideration was given at the time to possible repercussions on floatation integrity.

2. Analysis

2.1 Aircraft control difficulties

Clear evidence was found that extreme control difficulties were encountered as a result of two unconnected serious defects found in the flight control system.

The leak from the roll LBA System 2 main ram gland found after the accident, in the absence of any hydraulic system damage as a result of the ditching or recovery, together with the evidence that the unit had operated for a time with the gland partially displaced, indicated that System 2 had lost fluid via the LBA prior to the accident. System 2 contents at the time of the accident could not be established directly and no contents indication was available to the crew. However, crew reports, together with the zero cockpit pressure gauge reading found alongside normal readings for System 1 and the the Utility System were fully consistent with the System 2 reservoir having been empty at the time of capsize, when APU smothering by the sea would have caused almost simultaneous loss of hydraulic pump drives and pressure gauge electrical supplies. The System 2 pressure fluctuations observed by the crew were indicative of reservoir depletion causing the pump inlet to intermittently suck dry as the hydraulic system total volume varied with accumulator and UBA activity, both of which, in turn, would have occurred in this case because of the system pressure fluctuation (see below).

The System 1 SOV of the collective LBA was found seized partially closed after the accident, and the evidence indicated (para 2.3) that the System 1 half of the actuator was locked into a bypass mode as a result. No possible connection could be established between the SOV malfunction and either the loss of System 2 contents or any effects of the ditching and recovery.

Either of these malfunctions alone should have had no significant effect on aircraft control. The violent undemanded manoeuvres that in fact occurred as System 2 pressure was lost and restored were fully consistent with the System 1 half of the collective LBA having already been de-activated at this time. In this case as System 2 pressure was lost, the LBA output lever would fall under the download imparted by balance springs and downstream linkage weight, with the pilot's collective lever stationary, and impart a demand to the UBAs and hence cause aircraft vertical manoeuvres not demanded by the pilot. Progressive pilot demand to counteract this, combined with the undemanded LBA output lever movement, would eventually cause the LBA servo valve operating lever to bottom and, by providing a fulcrum, effectively connect input and output levers directly, but in a grossly mismatched state. In this condition System 2 repressurization would cause full rate drive of the LBA until the mismatch was eliminated, thus imparting a large vertical manoeuvre step demand to the UBA in the opposite sense to previously. Large rotor rpm variation would also occur during system depressurization and repressurization as the droop compensator, being driven by the LBA input linkage, would fail to sense the undemanded signals being generated downstream by the intermittent LBA deactivation. Computer modelling of this double failure case predicted aircraft manoeuvres and rotor rpm variations of the same order as those reported by the crew.

It was not clear whether control could have been maintained had System 2 pressure been lost permanently. A form of manual connection through the collective LBA would have obtained (see above), but in this state 5 inches of the total collective lever travel of 8 inches would effectively have been lost.

No indications of either malfunction were available to the crew, except by virtue of the System 2 pressure fluctuations and the violent disturbances experienced. The time before the onset of the disturbances at which each failure occurred could not be definitely established. However, the evidence suggested that hydraulic System 2 contents were probably lost over a period of 10-20 minutes. No indication of the leak was available in the cockpit. A cockpit indication of reservoir contents would thus have provided the crew with an appreciable warning time of the impending loss of one of the two hydraulic systems vital to the control of the aircraft. This could have been expected to result in termination of the flight before system pressure was lost, particularly as the aircraft was only ten minutes into the flight when the first pressure loss occurred.

The System 1 SOV of the collective LBA may have been seized partially open for some time (para 2.3), even though the point at which the actuator half entered bypass as a result could not be determined. No cockpit indication of an LBA bypass condition was provided. There was provision on the LBA for a sensor in the form of a jam indicator microswitch, but it became apparent during the investigation that a warning system utilising this switch would not have provided a warning in this case, as evidenced by the failure of the jam indicator button to operate. This was the result of LBA design features whereby the actuating pressures for both the SOV and the bypass valve were greatly below the actuating pressure for the jam indicator. Had an effective cockpit indication system of an LBA malfunction been provided, it would have given the crew an immediate warning that a serious flight control defect was present, quite possibly long before hydraulic System 2 started leaking.

No hydraulic system reservoir contents indication, low level warning, or indication of LBA malfunction was provided to the crew.

2.2 Lower boost actuator cap screw failure

The leakage past the roll LBA System 2 main ram gland resulted from failure of two of the three cap screws because of extensive fatigue cracking. The evidence indicated that the most highly loaded screw (No 1) failed first, and that the increased loading thereby falling on the other two screws caused these to fatigue, eventually resulting in fracture of the No 2 screw.

The No 1 screw did not conform to specification, in that the thread exhibited complete decarburisation. This is known to reduce fatigue strength, but testing subsequent to the accident indicated that the effect would not have been major. Additionally, some evidence suggested that the range of cylinder pressure cycling, and hence cap screw loading, may have been higher than anticipated as a result of mismatch between the two halves of the servo valve, probably largely due to the relatively low design preload of the spool bias springs (0.5 lb), but no reason was found to suggest this would have been particularly severe for G–BISO's roll LBA. The LBA design was susceptible to mismatch as it employed a relatively high servo valve gain (10^6 psi/inch) implying large main ram cylinder pressure variation for small relative displacement of servo valve spools. The possibility was also considered that fluctuating loads induced in the LBA output linkage by airframe vibration had contributed to the cap screw fatigue. Appreciable LBA cylinder pressure fluctuations from this cause were apparent in flight test measurement in some conditions, but available evidence suggested that fatigue damage to roll LBA cap screws from output linkage induced loads was very unlikely.

The endurance testing did show that screw preload had a very major effect on cap screw fatigue life. This life could not be estimated with any accuracy for G–BISO's failed cap screws because of difficulty in realistically assessing the loading environment experienced in service. A best approximation did suggest that, even under the most adverse conditions of full system pressure cycling (0-1500 psi), cap screws should have been able to withstand, as a minimum, many times the flight hours at which G–BISO's screws failed, if correctly torqued; whereas a No 1 screw, if untorqued, would suffer early failure. This would shortly be followed by failure of one of the other two screws, whether torqued or untorqued.

It was therefore concluded, in the absence of any major intrinsic defect in the failed screws, that the most likely cause of failure was lack of preload in the No. 1 screw, with excessive screw decarburisation and primary valve mismatch as contributory factors. All cap screws were found correctly wirelocked. The possibility was considered that previous overtorquing as a result of the assembly drawing error had been a factor, perhaps causing damage to the body or thread insert that had subsequently allowed correctly torqued screws to slacken in service. However, while the erroneous instruction called for 125-150 inch lb torque, it was found during the investigation that in practice generally only around 50 inch lb could be applied before local yielding caused the allen key to rotate within the internal hexagonal socket of the screw head. It was therefore doubtful if actuators had in fact experienced cap screw over-torque of more than around twice the correct value of 30-35 inch lb. Available evidence suggested this would not have caused permanent actuator damage.

Inability to achieve specified torque would have been expected to lead to discovery of the assembly drawing error, but this had apparently not occurred.

It was therefore concluded that the No. 1 cap screw had probably not been properly tightened at the time it was last disturbed, although no positive evidence for this was found. Some support for this conclusion was, however, added by the reports that shortly after the accident two LBAs received from the aircraft manufacturer were found during preparation for installation to exhibit gross cap screw undertorque. In one case one screw was affected and in the other case four screws. The manufacturer indicated that G–BISO's roll LBA cap screws were replaced by them in January 1983 following discovery of the assembly torque error.

With regard to the time at which the cap screws failed it was likely that the gland leakage would not have started until both screws fractured. From consideration of the measured leakage rate and likely pilot's roll control activity it was considered probable that the No 2 screw failed in the order of 10-20 minutes prior to the onset of the aircraft disturbances. The post accident fatigue tests indicated that the No 1 screw failure would have occurred a number of flying hours prior to the No 2 screw failure. However, the head portion of the No 1 screw was retained in place by its wire locking, with the fracture out of sight, and the failure would have been almost impossible to detect in the crowded control closet during the daily Ramp Check visual inspection.

A previous case, in November 1982, of loss of one hydraulic boost system as a result of fracture of two LBA cap screws was apparently attributed to screw overtorque. In view of the limited torque that can be achieved without stripping the screw socket, the possibility that the failure in fact resulted from screw undertorque could not be dismissed. In spite of the potential seriousness of the discovery at this point that all LBA cap screws had apparently been grossly overtorqued, the action recommended by the manufacturer to rectify the situation was not made mandatory by either FAA or CAA. The operator did institute a programme to embody the service bulletin recommending screw replacement, but there is no record of incorporation on G–BISO's pitch, yaw or collective LBAs. At the time of fleet incorporation of the bulletin they were all in repair or storage and because of a maintenance control system shortcoming the bulletin was scheduled only on actuators installed on an aircraft. The operator believes there is a possibility that the change was incorporated in response to the telexed service bulletin.

A further case of cap screw failure, to another operator's aircraft occurred after incorporation of the service bulletin, but no results of any investigation into the cause could be found.

2.3 Lower boost actuator in bypass mode

The System 1 SOV of the collective LBA was found during the investigation to be corroded and seized partially closed. With the exception of the corrosion, no evidence was found that the actuator deviated from the design standard in any respect.

While some corrosion could have occurred after the accident as a result of salt water contamination, corrosion products elsewhere in the actuator showed the influence of fluid flows and had clearly been deposited prior to the accident. The SOV normally closes against spring loading only in the event of a servo valve jam and there was no reason why the loss of hydraulic System 2, or the ditching or recovery should have had any effect on SOV spool position. As, in addition, testing showed that the System 1 half of the actuator was in bypass with the SOV spool held in the position found, but operated correctly (with the exception of the jam indicator) after the SOV was freed, it was concluded that at the time the control difficulties were encountered the SOV was already seized in the partially closed position, and the actuator half was in bypass as a result.

The overlap designed into the SOV meant that, with the spool having travelled only part-way closed, the supply pressure port would be completely blanked and no pressure would be available to operate the bypass valve or the indicator, or to complete closure of the SOV itself. (Figs 3 and 4.) The system apparently relied for its correct operation on the spool being carried by momentum across this overlap band, thereby connecting the jam indicator and the spool actuating pressure chamber directly with the supply pressure port; but it seemed likely that in some cases the spool could stall at a point where its actuating pressure was reduced by progressive inlet port blanking sufficiently to just balance the increasing spring force. This pressure was estimated at 110 psig, well below the nominal 250 psig required to maintain the bypass valve in its normal (non bypass) position. The SOV spool would not be latched in this situation and would revert to normal when the jam situation was removed. Such a stall condition was found during the investigation when both halves of the collective LBA failed to latch into a jam condition in most jam simulation tests, and similar problems were experienced with installed units after the accident and with a number of units during production acceptance tests in 1983. As a result of the latter events a spool modification had been introduced to provide SOV underlap, thus eliminating the possibility of complete inlet port blanking, but this was not incorporated generally, nor recommended for units in service.

No evidence found suggested that G–BISO's collective LBA had failed to pass regular jam simulation tests prior to the accident. However, with SOV operation marginal, slight changes in conditions, such as an increase in spool friction as a result of corrosion, could well have been sufficient to cause the spool to stall in a jam situation. Furthermore the relatively light spring loading on the spool (estimated at 6 lb at the stall position) suggested that a small degree of corrosion could readily have jammed the spool in this position.

The time at which the SOV seized could not be established. In the absence of any evidence of defects, such as high servo valve leakage or bleed orifice blockage that could have caused a build-up of SOV actuating pressure, it was concluded that the most likely cause of SOV spool displacement was a servo valve jam, real or simulated. No clear signs that an actual jam had occurred were found, but none would necessarily be expected. The last jam simulation test was 41 flying hours prior to the accident. The Technical Log report 5 days/ 22 flying hours before the accident, indicating that the System 1 side of the collective LBA was found during the pre-flight control check to be temporarily in bypass, was consistent with the SOV having been partially closed at this time, but alternatively a temporary jam could have occurred. No further reports or abnormalities were found until the accident.

However, the means available to flight or ground crew of detecting that an actuator half was in bypass were of dubious effectiveness, particularly for such vital components. As no flight deck indication of a bypass situation was provided, it would have been apparent to the flight crew only when the other half of the actuator was depressurized, a situation demonstrably dramatic in flight, but providing rather more subtle indications during pre-flight control checks. In the latter situation the two clues available would be lack of blade pitch change response and/or abnormal control forces. However, the check in use at the time of the accident did not require confirmation of blade response, and, as normal practice was apparently for collective and cyclic controls to be functioned simultaneously, lack of response in one axis would not have been apparent. Differences in control force versus displacement profiles for a bypass compared to a normal situation varied in each axis, being very marked for roll and yaw channels but rather less so for pitch and collective. Hence, the possibility that System 1 of the collective LBA was in bypass during a number of flights prior to the accident and not detected during pre-flightchecks could not be dismissed.

The pop-up button type jam indicator on each microswitch housing, actuated by the same pressure source as the SOV, was intended to provide indication to ground crew during daily inspection that a bypass situation had occurred, but records indicated that it had not been reported tripped at any time in the weeks prior to the accident and nor was it found tripped afterwards. However, the pressure required to trip the indicator (1180 psig measured) was vastly in excess of both the estimated 110 psig required to move the SOV spool against its spring into the overlap band, and the 250 psig required to hold the bypass valve fully out of the bypass position. The indicator was therefore not capable of detecting that the actuator half was in bypass unless the SOV operated correctly.

Analysis of hydraulic system corrosion products could not positively differentiate between chlorinated solvent and seawater induced reactions, and the situation was complicated by the seawater contamination that had indeed occurred after the ditching. Although excessive chlorinated solvent levels were not found in fluid samples from G–BISO it was quite possible that a previous contamination episode could have initiated a corrosive attack, which could then have been sustained by normal levels of dissolved water present after the fluid became decontaminated. Reduction in the contaminant level by virtue of normal maintenance and replenishment could occur, particularly as extensive mixing of aircraft and rig system contents would take place during these operations.

Although the cause of the hydraulic system corrosion found could not be positively established, the excessive levels of trichloroethane found in fleet and rig samples taken shortly after the accident suggested that this, in combination with water dissolved in the fluid, was the most likely cause. The lack of corrosion resistance of the material used for the various valves within the actuator was also an essential factor. The most likley source of the contamination was from cleaning solvents used during aircraft and hydraulic rig servicing, a number of which consisted very largely of trichloroethane; and features of the hydraulic system servicing arrangements were less than optimum in minimising the possibility of contamination. Some information sources suggested that corrosion could result from residues remaining on component surfaces after chlorinated solvent evaporation, while others believed that hydraulic system contamination with liquid solvent would be required.

A similar problem had apparently been encountered by the USAF with a different aircraft type some years previously, and hydrocarbon solvent cleaners had been substituted as a result. However, the potential in some applications for corrosion as a result of chlorinated solvent contamination did not seem to be generally appreciated.

2.4 Other flight control system defects

The No 2 cap screw of System 2 of the collective LBA was of an unapproved type, but how it came to be installed could not be determined. The almost complete lack of preload torque found was believed to be due to the severe thread distortion that had occurred.

The cause of the loose System 1 jam indicator on the pitch LBA was not established, but lack of positive means of locking the four mounting screws must have been a factor. The looseness was not sufficient to cause hydraulic fluid leakage. Had screws continued to back off far enough to allow complete indicator detachment, the leakage rate would probably have been minor normally, but significant in the event the actuator entered bypass.

The evidence of leakage from the aft swivelling UBA suggested that System 1 as well as System 2 may have experienced a significant leakage rate prior to the accident. However, the leak was not reproducible on test, and neither the leakage rate, nor the time for which it had persisted could be established.

None of these defects was relevant to the accident but did represent three additional potentially significant faults in the primary flying control systems, and emphasised the desirability of some form of hydraulic contents indication being available to the flight crew.

2.5 Floatation and stability

Although the ditching was on to an almost calm sea and left the aircraft undamaged it took on water rapidly and capsized 82 minutes after touchdown, in contrast with the Operations Manual predictions that the aircraft should remain afloat indefinitely and have stability in heavy sea state conditions. No evidence was found to suggest that intrinsic stability characteristics of the fuselage/fuel pod combination were suspect, and the capsize clearly resulted from the destabilising effects of the seawater taken on board.

Water ingress into the fuselage appeared to have been by a number of paths, with the cargo loading ramp seals probaly a primary source. A damaged portion of the ramp seal found, located well below the initial static waterline, would have provided a very significant leakage path directly into the cabin. The possibility that the damage resulted from water egress during the recovery could not be dismissed, although this was considered unlikely, and it was significant that another aircraft inspected shortly after the accident exhibited almost identical seal damage. In both cases the damage was in an area where strains on the seal during ramp operation could be expected to be at their maximum. It appeared possible that ramp seals could remain unseaworthy for some time as, although the Maintenance Programme required a 'general condition check of ramp and hinges' at 300 hour intervals, ramp seals were not directly specified, and no clear requirement for seal inspection or maintenance apparently existed.

The possibility was also considered that the manner in which the hinged ramp forward lateral seal was seated, ie, against a corner of the ramp rather than its surface, allowed leakage. The manufacturer's tank tests conducted on each production fuselage, and on a used aircraft after the accident, did not suggest that this feature caused any significant leakage. However, the tests could not be totally representative of the conditions experienced by G–BISO, particularly with regard to the varying head of water under wave and swell action that would be experienced in an actual seaborne case, and the possibility that some leakage into the fuselage resulted from this source could not be dismissed.

The cargo door seal was in a damaged condition with a large central portion missing, in company with a number of other aircraft inspected. While this was categorised as an aerodynamic rather than a water seal, it was only around 10 inches above the static waterline, and could fairly readily be intermittently covered by wave and swell action, particularly as any water taken on board caused the rear end of the fuselage to settle. It was considered likely that water entered G–BISO's fuselage by this path, but not in major quantities initially.

Continuous leakage into the fuselage probably took place via the forward right hand landing gear torsion box drain and the holes communicating the torsion box with the fuselage underfloor region, which were believed to be unblanked. No other fuselage leakage paths were identified, but the possibility that inspection did not reveal some routes could not be dismissed.

Cabin flooding was concentrated towards the rear. A bilge pump draining water from this area would have improved habitability and significantly delayed the capsize, but none was required and none was fitted.

Fuel tank pods were intended to be watertight and hence make a major contribution to buoyancy and stability, but both pods would have progressively flooded via two 3/8 inch diameter open holes found in their undersurfaces and located well below the initial waterline at a level where the head of seawater externally exceeded the head of fuel internally. Had the fuselage remained buoyant, pod interspace flooding via the holes would have ceased when the water depth inside the pods reached 3-4 inches. However, with the fuselage also taking on water and gradually sinking the pods would have continued to flood as well. In all likelihood the fuel bladders would have progressively collapsed as a result, with their contents being expelled through the vents. The aircraft apparently capsized before the waterlevel in the pods rose high enough to start expelling fuel. The holes in the pods were produced when permanent removal of the original screwed blanking plugs was required for CAA Validation of the FAA Type Certificate in order to improve the ability to detect fuel leakage from the bladders, apparently without the possible critical repercussions on floatation and stability being appreciated.

No rotor blade strike on the sea occurred but rotor disc clearance from the surface was of concern to the crew, even though the sea was relatively calm. North Sea conditions clearly are frequently much more severe, but attempts to assess the limiting conditions under which it would probably not be possible to avoid a blade strike on the sea before rotors stopped turning, and the likely effects of such a strike, were frustrated by lack of information, and no meaningful conclusions were reached. It did appear that for any type of helicopter a rotor blade sea strike following ditching was likely to have a significant effect on survivability, but there are no BCARs specifically concerned with the possibility of blade strike.

2.6 Ditching exits

For the ditching case BCARs required a rotorcraft of passenger seating capacity from 40 to 59 to be fitted with two Type III exits, one per side, located above the water level. The BV 234 exceeded this requirement, being provided with one Type III and one Type IV exit on each side, all above the water level.

This accident has shown that there is no great difficulty in achieving a successful escape of a full passenger load through these exits, following a controlled ditching in which the aircraft remains upright on the water. The eventual capsize was caused by flooding of the airframe and consequent loss of buoyancy and stability. It may therefore be argued that the examination of the emergency escape from a capsized BV 234 has no place in this report, particularly as tank testing has shown that the waterborne stability characteristics when the aircraft is watertight are generally superior to other North Sea helicopters. This is achieved without recourse to relatively vulnerable blow-out floatation bags. However, the tank tests did demonstrate that a capsize was possible in Sea State 5, which is not a rare occurrence in the North Sea. Also, the effects on stability of a rotor blade wave strike in some conditions is of possible concern, albeit not quantifiable. It is therefore considered appropriate to discuss underwater escape.

The problems of escaping from a fuselage floating inverted are considerable. A US Army report on aircraft crash survival published in 1980 (Aircraft Crash Survival Design Guide, Volume V, Aircraft Post-crash Survival. USARTL TR–79–22E) included a section on underwater escape. Based on interviews with helicopter ditching survivors it found that the major difficulty encountered was due to water rushing in through the exits. Difficulties in reaching and operating hatches, disorientation and darkness were quoted as additional major factors. Underwater escape tests completed by the Royal Navy suggest that the maximum number of trained personnel likely to escape from one hatch is four. In such circumstances the occupants of a BV 234 passenger cabin would have only four escape exits available. It would therefore appear that, if a capsize should occur after a controlled ditching into the North Sea, a significant number of passengers would be trapped in the fuselage and fatalities would be inevitable.

The CAA have stated that, at the time of certification of the BV 234, BCARs did not aim to provide exits to allow occupants egress from a capsized aircraft. In the BV 234 the ratio of passengers to exits is 11 : 1, whereas in other North Sea helicopters the ratio was typically 5 : 1 at the time of the accident, and more recent modifications to some helicopters has improved this ratio. Consequently, although the BV 234 exceeds the ditching exit requirements specified in BCARs, a passenger's potential for escape from the capsized aircraft is significantly worse than that from other North Sea helicopters. It is therefore considered that the adequacy of BCARs concerning emergency exits should be reviewed.

The CAA have stated that increasing the window size in the BV 234 to provide additional Type IV exits would require such major structural changes as to be prohibitive. Tests conducted by the Royal Air Force Institute of Aviation Medicine (Report No 528 – January 1986) have shown that the minimum dimensions of an aperture through which a 95th percentile male can escape, while wearing the standard survival clothing used by North Sea passengers, is 17 inches by 14 inches. Even though such an exit is significantly smaller than a Type IV exit, consideration should be given to the possibility of modifying some of the BV 234 windows to provide additional exits of such minimum dimensions.

2.7 Liferafts

A capsize immediately after the ditching would have precluded the launching of either of the internally stowed liferafts. Those passengers and crew who were able to escape from the aircraft would therefore have been deprived of a major aid to their subsequent survival. Some of them would have been able to cling to the upturned hull but no hand holds or safety lines were available. Two externally mounted liferafts and safety lines along the fuel sponsons have now been introduced.

Although the Flight Manual and crew drills state that the rotors should be stopped before evacuation, the decision had been made to watertaxy towards the nearest rig, and not until later did it become obvious that the aircraft was taking on water. The liferafts were launched through a forward ditching exit but before most of the passengers could enter them both liferafts, in turn, separated from the aircraft and drifted away when the painters were cut on the sharp kevlar edge of the emergency exit aperture. This occurred because the painters were secured inside the cabin rather than on the external cleat. The effect of the rotor downwash, combined with the forward motion of the aircraft through the water, would have increased the tension in the painter and exacerbated the cutting effect of the sharp kevlar edge.

The length of the painter would have allowed the liferafts to float away from the emergency exit and the crew could have secured it to the external cleat provided above the exit. However, the cleat is neither apparent from inside the cabin nor located in an optimum position to maintain a liferaft close to the exit and the crew could have had little reason to believe that it was unsatisfactory to simply shorten the painter by securing it to a seat. A short painter to hold the liferaft close to the aircraft, and protection from the kevlar edges has now been introduced.

2.8 **Safety equipment**

A control malfunction such as occurred in this case required the attention of both of the pilots until after the ditching, thus leaving little opportunity for them to put on either immersion suits or LSJs. In the event of an immediate capsize the large jettisonable doors by each pilot's seat would provide them with every opportunity to escape from the aircraft. However, the lack of LSJs and immersion suits would seriously reduce their subsequent survival chances. The cabin attendant may also have little opportunity to put on an immersion suit and LSJ before the ditching and be equally poorly placed after escape. More serious than the loss of the LSJs would be the loss of the three emergency radio beacons (SARBE) which they contain, as this would be likely to delay location and rescue, particularly in poor weather.

The requirement for North Sea helicopters to be fitted with an Automatically Deployable Emergency Locator Beacon (ADELB) which was introduced on 1 March 1986 has largely overcome this deficiency, but a policy of wearing LSJs continuously would be a further improvement.

2.9 **General**

In common with all twin-engined helicopters currently operating over the North Sea, the BV 234 is provided with only two flight control hydraulic systems and no manual reversion. The investigation revealed a number of flight control system defects and a background of deficiencies concerned with flight control systems, in LBA design and manufacture; in aircraft design; in operating and maintenance procedures; and in investigation and resolution of potentially serious LBA faults encountered prior to the accident. The extent of this background, as described in preceding sections, was indicative of a widespread lack of appreciation of the particular need for a high standard of integrity of each flight control system on such an aircraft.

The evidence also indicated some lack of awareness of the need to maintain aircraft watertightness in service. The hull formed a fairly complicated floatation vessel with numerous hatches, detachable panels and drains below waterline. While nearly all of these were in fact well sealed, it was clear that close attention would be necessary to maintain floatation integrity.

It was noted that subsequent to the accident a considerable number of measures aimed at rectifying both the flight control system and floatation deficiencies have been taken.

3. Conclusions

3.a *Findings*

1. The crew members were properly licensed and adequately experienced to conduct the flight.

2. The aircraft had been maintained in accordance with an approved maintenance schedule and the Certificates of Airworthiness, Registration and Maintenance were valid at the time of the accident.

3. Violent undemanded manoeuvres accompanied by rotor rpm fluctuations occurred without warning because the System 2 side of the collective Lower Boost Actuator (LBA) intermittently de-activated as a result of system fluid depletion when the System 1 side of the actuator was already in bypass mode.

4. Virtually complete in-flight loss of collective control may well have been experienced had the crew not elected to ditch shortly after the onset of the disturbances.

5. Hydraulic System 2 contents were depleted by fluid loss from the roll LBA as a result of failure of two of the three cap screws.

6. The cap screws failed as a result of fatigue damage, consistent with the effects of one or both screws having been undertorqued at their last installation. Excessive screw thread decarburisation and servo valve mismatch were contributory factors.

7. A number of LBA cap screws not related to the accident aircraft were found grossly undertorqued following the accident.

8. Prior to the accident the effects of a gross LBA cap screw assembly error and cases of in-service cap screw failure were not effectively investigated or resolved.

9. It was probable that the initial cap screw failure was not detectable by normal daily ground crew inspection. No information on hydraulic system contents was available on the flight deck to warn of impending loss of one flying control hydraulic system, in spite of the serious degradation in flight control system integrity caused by such a loss.

10. LBAs were designed such that partial Shut Off Valve (SOV) closure was sufficient to place the actuator in bypass, without the jam indicator being actuated, and the System 1 side of the collective LBA was in this condition, with the SOV seized partially closed.

11. The normally open SOV had probably attempted to close because of a servo valve jam at an undetermined point prior to the accident and stalled part way as a result of its overlap feature, possibly assisted by the effects of corrosion. It failed to reopen after the jam condition disappeared because the return spring was by design relatively weak and unable to overcome corrosion induced friction.

12. A number of LBAs were found both before and after the accident to fail the jam test. SOV spool modification introduced to eliminate overlap was not incorporated or recommended generally prior to the accident.

13. The SOV corrosion probably resulted from hydraulic system contamination with trichloroethane cleaning solvents used during aircraft, component, and/or hydraulic rig servicing. A contributory factor was the low level of corrosion resistance of the SOV materials.

14. No flight deck warning of an LBA bypass condition was provided, in spite of the serious degradation in flight control system integrity caused by such a condition. The actuators incorporated provision for jam indicator microswitches, but the LBA design was such that a warning system based on these would not have provided an indication in this case.

15. Pre-flight flying control system checks were inadequately specified in the Flight Manual and conducted in a way that did not maximise the chances of detecting an LBA bypass condition. The possibility that an unrecognised bypass condition on the collective LBA had persisted for a number of flights could not be dismissed.

16. Three additional flying control system defects were identified which, although not relevant to the accident, could potentially have had serious repercussions on flying control systems integrity, particularly in the absence of flight deck indication of hydraulic system contents.

17. The aircraft ditched undamaged on an almost calm sea, but capsized 82 minutes after touchdown after taking on large quantities of sea water. This resulted from major leakage into fuel pods via unblanked drain holes, and into the fuselage probably via previously damaged rear cargo ramp seals and a landing gear torsion box drain. The possibility that other unidentified leakage paths existed could not be totally dismissed.

18. Assessment of the conditions in which a rotor blade strike on the sea after ditching would probably not be avoidable, and of the likely effects, was precluded by lack of information.

19. The BV 234 has a high passenger to ditching exit ratio although ditching exit provisions are well in excess of British Civil Airworthiness Requirements. These did not aim at ensuring the occupant's egress from a capsized aircraft.

20. Both liferafts released from the aircraft prematurely when their painters were secured inside the cabin and cut on a sharp edge of the emergency exit surround. The external cleat provided but not used was not best positioned to produce an optimum liferaft mooring position for boarding. Contributory factors to the painter cutting were the forward motion of the aircraft and the effect of downwash.

3.b *Cause*

The accident was caused by two separate flying control system malfunctions that produced intermittent loss of collective control. Lack of floatation integrity led the aircraft to capsize after ditching.

4. Safety Recommendations

1. The CAA in conjunction with other airworthiness authorities re-assess, based on system failure rates in service, the ability to achieve an acceptable level of safety in public transport operations with helicopters provided with only two flight control hydraulic systems and no manual reversion.

2. The CAA in conjunction with Boeing Vertol review the design of the BV 234 hydraulic flight system components with particular reference to:

   (a) Possible deletion of the Lower Boost Actuator jam protection feature in the light of CH–47 Chinook experience.

   (b) Modification of the jam indicator to monitor directly the bypass valve position and provide a flight crew warning of a bypass condition pending any action on (a) above.

   (c) The loading environment experienced by the Lower Boost Actuator cap screws and their resultant fatigue lives.

   (d) The integrity of Upper Boost Actuator seals.

3. The CAA in conjunction with other airworthiness authorities consider requiring on all hydraulic flight systems:

   (a) The locking of all bolts the insecurity of which can lead to fluid loss.

   (b) The use of improved corrosion resistant materials and the introduction of means to prevent contamination from chlorinated solvents.

   (c) Flight crew warning of low fluid level.

4. The manufacturer and operator amend their procedures for embodying modifications so as to ensure that components installed on aircraft after the main embodiment programme are also modified, and that the operator maintains complete records on all critical components.

5. The CAA conduct a review into public transport helicopter floatation and stability after ditching in respect of the following:

   (a) Identifying and where possible plugging leaks due to design, modifications, and service use.

   (b) The installation of a bilge pump system on aircraft used on long overwater flights.

   (c) The effects of a wave strike by a rotor blade.

6. The operator amend the BV 234 ditching drills to reflect a realistic assessment of the likely floatation capabilities.

7. The operator modify the pre-flight, flight control system drills to optimise detection of a Lower Boost Actuator bypass condition.

8. The CAA conduct a review of the number and type of exits and the safety equipment required for public transport helicopters with particular reference to:

   (a) External location of liferafts and provision of effective restraint during passenger boarding of them.

   (b) Providing flight deck and cabin crew with immersion suits and life saving jackets suitable for permanent wear.

9. The CAA make mandatory all manufacturer's Service Bulletins which implement solutions to safety critical defects.

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