AIRCRAFT ACCIDENT REPORT 2/2014

AAIB

Air Accidents Investigation Branch

Report on the accidents to Eurocopter EC225 LP Super Puma

G-REDW
34 nm east of Aberdeen, Scotland
on 10 May 2012

and

G-CHCN
32 nm southwest of Sumburgh, Shetland Islands
on 22 October 2012

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This investigation has been conducted in accordance with
Annex 13 to the ICAO Convention on International Civil Aviation,
EU Regulation No 996/2010 and
The Civil Aviation (Investigation of Air Accidents and Incidents) Regulations 1996.

The sole objective of the investigation of an accident or incident under these Regulations is the prevention of future accidents and incidents. It is not the purpose of such an investigation to apportion blame or liability.

Accordingly, it is inappropriate that AAIB reports should be used to assign fault or blame or determine liability, since neither the investigation nor the reporting process has been undertaken for that purpose.

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Farnborough House
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May 2014

The Right Honourable Patrick McLoughlin
Secretary of State for Transport

Dear Secretary of State

I have the honour to submit the report by Mr P A Sleight, an Inspector of Air Accidents, on the circumstances of the accidents to Eurocopter EC225 LP Super Puma, registration G-REDW, 34 nm east of Aberdeen, Scotland on 10 May 2012 and Eurocopter EC225 LP Super Puma, registration G-CHCN, 32 nm southwest of Sumburgh, Shetland Islands on 22 October 2012.

Yours sincerely

Keith Conradi
Chief Inspector of Air Accidents

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Contents

Appendices

GLOSSARY OF ABBREVIATIONS USED IN THIS REPORT

°C	Degrees Centigrade
°F	Degrees Fahrenheit
°M	Degrees Magnetic
a	Crack length
AAD	Additional Airworthiness Directive
AAIB	Air Accidents Investigation Branch
aal	above airfield level
AD	Airworthiness Directive
ADELT	Automatically Deployable Emergency Locator Transmitter
AFM	Aircraft Flight Manual
agl	above ground level
AMC	Acceptable means of compliance
AMM	Aircraft Maintenance Manual
AMO	Approved maintenance organisation
amsl	above mean sea level
ARCC	Aeronautical rescue co-ordination centre
AS332 Shaft	Shaft manufactured from 16NCD13 steel
ASB	Alert Service Bulletin
ATC	Air Traffic Control
ATCO	Air Traffic Control Officer
bar	Unit of gauge of pressure
BEA	Bureau d'Enquêtes et d'Analyses pour la sécurité de l'aviation civile
BRU	Beacon Release Unit
BST	British Summer Time (UTC+1)
CAA	Civil Aviation Authority
CAM	Cockpit Area Microphone
CAP	Civil Aviation Publication
cm	centimetre
CMM	Component Maintenance Manual
CPI	Crash Position Indicator
CRM	Cockpit Resource Management
CS	Certification Specification
CSI	Controlled Service Introduction
CVFDR	Combined Voice and Flight Data Recorder
CVR	Cockpit Voice Recorder
CWP	Central Warning Panel
daN	deca-Newtons
DDP	Declaration of Design and Performance
DGAC	Direction générale de l'aviation civile
DOA	Design Organisation Approval - Part 21 (J)
EASA	European Aviation Safety Agency
EBS	Emergency Breathing System
EC	Eurocopter
EC225 shaft	Shaft manufactured from 32CDV13 steel
EDR	EuroHUMS Diagnostic Report
ELT	Emergency Locator Transmitter
EMLUB	Emergency Lubrication System
ETSO	European Technical Standard Order
EUROCAE	European Organisation for Civil Aviation Equipment
FAA	Federal Aviation Administration
FAR	Federal Aviation Requirement
FCOM	Flight Crew Operations Manual
FDM	Flight Data Monitoring
FDR	Flight Data Recorder
FEM	Finite Element Model
EID	Electronic Instrument Display
FL	Flight Level
FMECA	Failure Mode Effect Causal Analysis
fpm	feet per minute
ft	feet
g	gravity
GEO	Geostationary
GPS	Global Positioning System
GSC	Ground Station Computer
HAZ	Heat Affected Zone
HOMP	Helicopter Operations Monitoring Programme
HUMS	Health and Usage Monitoring System
Hz	Hertz
IiC	Investigator in Charge
IMC	Instrument Meteorological Conditions
JAA	Joint Aviation Authorities
JAR	Joint Airworthiness Requirement
K	Stress intensity factor
KF	Karl-Fischer
kg	kilogram
Kg	HUMS condition indicator -Tooth damage and general wear
km	kilometre
K_m	HUMS condition indicator - Localised damage to gear teeth
kt	knot
K_t	Stress concentration factor
lb	pound
LEO	Low Earth Orbit
LPC	Licence proficiency check
m	metre

GLOSSARY OF ABBREVIATIONS USED IN THIS REPORT

M'ARMS	Modular Aircraft Recording Monitoring System
mb	millibar
MCP	Max Continuous Power See Appendix D
mg	milligram
MGB	Main Rotor Gearbox
Mhz	Mega hertz
MK	mark
mm	millimetre
MOB	Main Operating Base
MOD-45	HUMS condition indicator bevel gear meshing
MOD-70	HUMS condition indicator oil pump drives
MOPS	Minimum Operational Performance Specifications
MPa	Mega Pascal
N	Number of cycles
n	Safety factor
NE	north east
nm	nautical mile
Nm	Newton metres
NPRM	Notice of Proposed Rule Making
OPC	Operator proficiency check
PCB	Printed Circuit Board
PCMCIA	Personal Computer Memory Card International Association
PIC	Pilot in command
PLB	Personal Locator Beacon
POA	Production Organisation Approval – Part 21 (G)
p_{off}	Low pressure threshold
p_{on}	High pressure threshold
ppm	Parts per million
PSE	Primary Structural Element
psi	pounds per square inch
PSU	Pressure Sensor Unit
PTFE	Polytetrafluoroethylene
R	Ratio R = σmin / σmax
R_a	Average surface roughness
RFM	Rotorcraft Flight Manual
RMS	Root mean square
RMS-r	HUMS condition indicator - General wear and misalignment
RMT	Rule Making Task
RNLI	Royal National Lifeboat Institution
RT	Radio Transmission
R_z	The difference between the highest peak and lowest valley on a surface
S	Stress
SAR	Search and Rescue
SB	Service Bulletin
SEM	Scanning Electron Microscope
SIU	System Interface Unit
TAN	Total Acid Number
TOP	Take Off Power See Appendix D
TOPtran	Take Off Power - Transient See Appendix D
TSB	Transportation Safety Board - Canada
UK	United Kingdom
UMS	Usage Monitoring System
US	United States
UTC	Co-ordinated Universal Time
UTS or σ_{UTS}	Ultimate Tensile Stress
VFR	Visual Flight Rules
VHF	Very High Frequency
VHM	Vibration Health Monitoring
VMC	Visual Meteorological Conditions
VMS	Vehicle Monitoring System
VNE	never exceed airspeed
VOR	VHF omni-range
WG	Working Group
XMSN	Transmission
V_y	Recommended climb speed
d	Depth of corrosion pit
β	factor dependent on the component geometry and loading inputs
ΔK	Change in stress intensity factor
ΔK_{th}	stress intensity threshold below which a fatigue crack will not grow
ρ	Corrosion pit radius
µm	Micro-metre (micron) 1 µm = 0.001 mm
σ	stress
σ_a	alternating tensile stress
σ_{dyn}	alternating tensile stress
σ_e	endurance limit or fatigue limit
σ_m	mean stress
σ_{max}	maximum stress
σ_{min}	minimum stress
σ_p	peak stress
σ_{stat}	mean stress

Figure 1

G-REDW afloat, approximately 8 hours after ditching

Figure 2

G-CHCN afloat, approximately 24 hours after ditching

Air Accidents Investigation Branch

Aircraft Accident Report No: 2/2014

ACCIDENT INVOLVING G-REDW (EW/C2012/05/01)

Aircraft Type and registration:	EC225 LP Super Puma, G-REDW
Registered Owners and Operators:	Bond Offshore Helicopters Ltd
Nationality	British
Date & Time (UTC):	10 May 2012 at 1114 hrs
Location:	34 nm east of Aberdeen

ACCIDENT INVOLVING G-CHCN (EW/C2012/10/03)

Aircraft Type and registration:	EC225 LP Super Puma, G-CHCN
Registered Owners and Operators:	CHC Scotia Ltd
Nationality	British
Date & Time (UTC):	22 October 2012 at 1425 hrs
Location:	32 nm southwest of Sumburgh, Shetland Islands

Introduction

The Air Accidents Investigation Branch (AAIB) was notified at 1112 hrs on 10 May 2012 that an EC225 LP Super Puma, G-REDW, was preparing to ditch in the North Sea approximately 32 nm east of Aberdeen.

On 22 October 2012 the AAIB was notified at 1428 hrs that an EC225 LP Super Puma, G-CHCN, had ditched in the North Sea approximately 32 nm southwest of Sumburgh, Shetland Islands.

In both cases the AAIB deployed a team to Aberdeen to commence an investigation. In accordance with established International arrangements the Bureau d'Enquêtes et d'Analyses pour la sécurité de l'aviation civile (BEA), representing the State of Manufacture of the helicopter, and the European Aviation Safety Agency (EASA), the Regulator responsible for the certification and continued airworthiness of the helicopter, were informed of the accidents. The BEA appointed an Accredited Representative to lead a team of investigators from the BEA and Eurocopter[1] (the helicopter manufacturer). The EASA, the helicopter operators and the UK Civil Aviation Authority (CAA) also provided assistance to the AAIB team.

Owing to the similarities of the circumstances that led to the two accidents, the Chief Inspector of Air Accidents ordered that the investigations be combined into a single report.

Synopsis

While operating over the North Sea, in daylight, the crews of G-REDW and G-CHCN experienced a loss of main rotor gearbox oil pressure, which required them to activate the emergency lubrication system. This system uses a mixture of glycol and water to provide 30 minutes of alternative cooling and lubrication. Both helicopters should have been able to fly to the nearest airport; however, shortly after the system had activated, a warning illuminated indicating that the emergency lubrication system had failed. This required the crews to ditch their helicopters immediately in the North Sea. Both ditchings were successful and the crew and passengers evacuated into the helicopter's liferafts before being rescued. There were no serious injuries.

The loss of oil pressure on both helicopters was caused by a failure of the bevel gear vertical shaft in the main rotor gearbox, which drives the oil pumps. The shafts had failed as result of a circumferential fatigue crack in the area where the two parts of the shaft are welded together.

On G-REDW the crack initiated from a small corrosion pit on the countersink of the 4 mm manufacturing hole in the weld. The corrosion probably resulted from the presence of moisture within the gap between the PTFE plug and the countersink. The shaft on G-REDW had accumulated 167 flying hours since new.

On G-CHCN, the crack initiated from a small corrosion pit located on a feature on the shaft described as the inner radius. Debris that contained iron oxide and moisture had become trapped on the inner radius, which led to the formation of corrosion pits. The shaft fitted to G-CHCN had accumulated 3,845 flying hours; this was more than any other EC225 LP shaft.

The stress, in the areas where the cracks initiated, was found to be higher than that predicted during the certification of the shaft. However, the safety factor of the shaft was still adequate, providing there were no surface defects such as corrosion.

1 On 1 January 2014 Eurocopter changed its name to Airbus Helicopters.

The emergency lubrication system operated in both cases, but the system warning light illuminated as a result of an incompatibility between the helicopter wiring and the pressure switches. This meant the warning light would always illuminate after the crew activated the emergency lubrication system.

A number of other safety issues were identified concerning emergency checklists, the crash position indicator and liferafts.

Ten safety recommendations have been made. In addition, the helicopter manufacturer carried out several safety actions and is redesigning the bevel gear vertical shaft taking into account the findings of the investigation. Other organisations have also initiated a number of safety actions as a result of this investigation.

The following causal factors were identified in the ditching of both helicopters:

a A 360º circumferential high-cycle fatigue crack led to the failure of the main gearbox bevel gear vertical shaft and loss of drive to the oil pumps.

b The incompatibility between the aircraft wiring and the internal configuration of the pressure switches in both the bleed-air and water/glycol (Hydrosafe 620) supplies resulted in the illumination of the MGB EMLUB caption.

The following factors contributed to the failure of the EC225 LP main gearbox bevel gear vertical shafts:

a The helicopter manufacturer's Finite Element Model underestimated the maximum stress in the area of the weld.

b Residual stresses, introduced during the welding operation, were not fully taken into account during the design of the shaft.

c Corrosion pits were present on both shafts from which fatigue cracks initiated:

i On G-REDW the corrosion pit was located at the inner countersink in the 4.2 mm hole and probably resulted from the presence of moisture within the gap between the PTFE plug and the countersink.

ii On G-CHCN the corrosion pit was located at the inner radius and probably resulted from moisture trapped within an iron oxide deposit that had collected in this area.

1 Factual information

1.1 History of the flights

1.1.1 Background to helicopter operations

G-REDW and G-CHCN were engaged on flights in support of the offshore oil and gas industry.

Helicopters utilised in these operations are flown predominantly offshore, over open sea areas, for substantial portions of their flight time. A sequence of flights generally originate onshore, for example from Aberdeen Airport, although on occasions the helicopter could be based on an offshore facility.

Typically, the helicopters would depart onshore and fly to an offshore facility, carry out a rotors-running changeover of personnel (unless refueling was required), perform the same task at other offshore installations nearby, before returning on a longer sector back to the onshore base. Occasionally helicopters would be required to shut down at an offshore facility to support activities such as maintenance.

Passengers are required to wear specific safety equipment and to be given pre-departure safety briefings prior to boarding the helicopter. Most passengers have flown in offshore helicopters before; those inexperienced as an offshore helicopter passenger are identified by the wearing of a green armband.

Aberdeen Airport is a main operating base used by the operators of both G-REDW and G-CHCN.

1.1.2 G-REDW

G-REDW was operating a flight, scheduled to depart at 1030 hrs from Aberdeen Airport, to the Maersk Resilient platform in the North Sea, 150 nm east of Aberdeen.

The helicopter, using callsign Bond 88R, departed Aberdeen at 1045 hrs and established on course towards the Maersk Resilient platform. The helicopter was about 34 nm east of Aberdeen Airport, in the cruise at 3,000 ft with the autopilot engaged and at an approximate speed of 143 kt, when the crew were presented, almost simultaneously, with the following indications:

- a WARN red master light and aural gong
- a CAUT amber master light
- an amber XMSN (transmission) and red MGB.P (MGB loss of lubrication) captions on the Central Warning Panel (CWP)
- an amber MP (main oil pump pressure) and amber S/B.P (standby oil pump pressure) captions and a red flashing digital value for pressure displayed on the Vehicle Monitoring System (VMS)
- a zero indication on the main rotor gearbox (MGB) oil pressure gauge
- an EMLUB SHOT illumination on the MGB control box

The commander assumed control of the helicopter, disengaged the autopilot upper modes, reduced power and stabilized the airspeed at 80 kt, the V_y[1] speed. He then re-engaged the autopilot upper modes, called for the co-pilot to action the checklist and broadcast a PAN call requesting immediate descent to 1,500 ft. The PAN call was acknowledged and clearance for descent was given. The commander then requested an immediate return to Aberdeen.

One minute after the loss of oil pressure, the commander started the turn back towards the coast and initiated a descent. The helicopter was in IMC conditions at the time, with the cloud base reported as being about 600 to 700 ft amsl.

The co-pilot opened the emergency checklist and found the applicable page for '*Total Loss of MGB Oil Pressure*' (Appendix A). He waited until he had the commander's attention and then for his cross-confirmation before carrying out the checklist actions. During this time there were several interruptions from further Air Traffic Control (ATC) transmissions.

The co-pilot activated the emergency lubrication system 1 minute 50 seconds after the initial warning. He continued with the checklist and advised the commander that they were limited to a maximum 30 minutes of flight time and should land as soon as possible. The MGB EMLUB caption then illuminated on the CWP. The co-pilot drew the commander's attention to this and advised him that they were now required to land immediately.

The co-pilot gave the passengers an emergency briefing while the commander carried out the descent and advised ATC that he was descending to 500 ft and may need to ditch. On receipt of this information, the Air Traffic Control Officer (ATCO) directed two helicopters, who were en-route in the vicinity, to the scene. The ATCO then requested several other aircraft to call other agencies in order to reduce the number of aircraft on the VHF frequency.

[1] V_y is the recommended climbing speed, which is 80 kt TAS.

The commander reviewed the situation, noting that they were about 30 nm offshore and heading towards the coast (Figure 3). The co-pilot made several references to the need to land or ditch immediately. The commander briefed the passengers to prepare for a ditching and then called for the ditching checklist. The co-pilot carried out the *'power-on ditching'* checklist (Appendix A) and the commander spoke to the passengers again to remind them of the procedure after ditching.

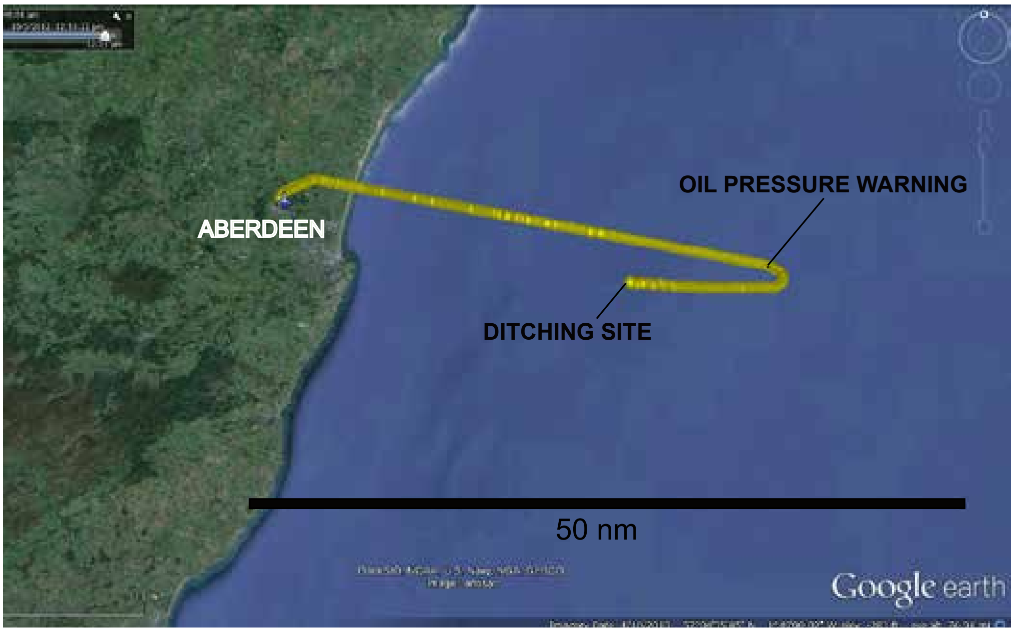


Figure 3

G-REDW accident flight radar track

The co-pilot manually deployed the floats while the commander descended the helicopter slowly and continued to fly towards the coast. Both pilots then noticed an unusual oily smell and the decision to ditch immediately was made by the commander.

As the commander turned the helicopter into wind, the co-pilot transmitted a MAYDAY call stating that they were ditching. This MAYDAY call was initiated whilst another agency was transmitting and was therefore partially blocked; however, the intent of the distress call was understood by the controller. The commander ditched the helicopter; the total flight time was 27 minutes.

The helicopter remained upright, supported by the emergency flotation gear. After shutting down the engines and stopping the rotors, the crew and passengers evacuated the helicopter into one of the two liferafts via the right crew and cabin doors. The two en-route helicopters arrived on the scene and made visual contact with the ditched helicopter, but were not able to establish radio communication.

Two further Search and Rescue (SAR) helicopters were tasked to go to the scene, one from the Miller platform and one from RAF Boulmer, both approximately one hour flight time away. The first SAR helicopter on the scene arrived at 1220 hrs and was able to locate visually the ditched helicopter and liferaft. Six of the occupants were rescued from the liferaft by a SAR helicopter and eight were transferred to a Royal National Lifeboat Institute (RNLI) lifeboat.

1.1.3 G-CHCN

G-CHCN was operating a scheduled flight from Aberdeen Airport to the West Phoenix drilling rig, approximately 226 nm to the north. The crew consisted of a commander and a training captain, acting as co-pilot. It was intended to use the flight for training towards the revalidation of the commander's line training qualification.

The helicopter departed Aberdeen, using callsign HKS24T, at 1322 hrs and turned to the north. The flight was uneventful until about 60 minutes into the flight. Whilst in the cruise at 140 kt and 3,000 ft amsl and with approximately 81% total torque applied, the following indications were displayed:

- a WARN red master light and aural gong
- a CAUT amber master light
- an amber XMSN and red MGB.P captions on the CWP
- an amber MP and amber S/B.P captions and a red flashing digital value for pressure displayed on the VMS
- a zero indication on the MGB oil pressure gauge
- an EMLUB SHOT illumination on the MGB control box

The crew carried out the '*Total Loss of MGB Oil Pressure*' checklist (Appendix A), which required the activation of the MGB emergency lubrication system and the slowing of the helicopter to V_y. The autopilot upper modes were disengaged. Twenty-nine seconds after the initial warning, the crew activated the emergency lubrication system. However, within a minute, the MGB EMLUB caption illuminated on the CWP indicating that the emergency lubrication system had failed.

The illumination of the MGB EMLUB caption required them, in accordance with the checklist, to land immediately, so the crew prepared to carry out a controlled ditching. They transmitted a MAYDAY call to Sumburgh Radar and warned the passengers, so that they could prepare for the ditching.

The helicopter had been flying in VMC on top of a cloud layer and on descending through the cloud the crew became visual with the sea at about 300 ft amsl. They continued their descent to about 50 ft amsl and observed a merchant ship ahead of them (Figure 4).

They used Channel 16, the marine distress channel, to contact the ship and then hover-taxied towards it. The crew completed the '*Emergency Landing – Power ON*' checklist (Appendix A), including manually arming and inflating the floats and selecting the Crash Position Indicator (CPI) to TRANSMIT. They ditched the helicopter successfully, close to the ship. One of the passengers, who was used to working with glycol, commented that at about this time he was aware of a smell of glycol in the cabin.

The helicopter remained upright, supported by the emergency flotation gear. The passengers and crew successfully evacuated the helicopter and boarded two liferafts before being transported to the ship by the fast rescue craft launched from the vessel. The ship's log recorded all crew and passengers safely on board at 1532 hrs with no reported injuries.

Figure 4

G-CHCN accident flight radar track

1.2 Injuries to persons

1.2.1 G-REDW

Injuries	**Crew**	**Passengers**	**Others**
Fatal	—	—	—
Serious	—	—	—
Minor/None	2	12	

1.2.2 G-CHCN

Injuries	**Crew**	**Passengers**	**Others**
Fatal	—	—	—
Serious	—	—	—
Minor/None	2	17	

1.3 Damage to aircraft

Neither G-REDW nor G-CHCN sustained any structural damage as a result of ditching in the North Sea; however, minor structural damage was sustained during the recovery operation. The lower part of both helicopters and a number of electrical and avionic systems had been immersed in salt water.

1.4 Other damage

None

1.5 Personnel information

1.5.1 G-REDW

1.5.1.1 Commander

Age:	40 years	
Licence:	Airline Transport Pilot's Licence	
Licence expiry date:	10 June 2014	
Helicopter Ratings:	EC225 LP/AS332/AS355/ R22	
Operator Proficiency Check:	Valid to 31 October 2012	
Licence Proficiency Check:	Valid to 31 October 2012	
Line check:	Valid to 31 July 2013	
Medical certificate:	Valid to 13 January 2013	
Flying Experience:	Total all types:	3,060 hours
	Total on type:	2,740 hours
	Last 90 days:	140 hours
	Last 28 days:	27 hours
	Last 24 hours:	nil hours
Previous rest period:	38 hrs 45 minutes	

1.5.1.2 Co-pilot

Age:	28 years	
Licence:	Airline Transport Pilot's Licence	
Licence expiry date:	31 October 2012	
Helicopter Ratings:	EC225 LP/AS332/EC1345/ H269	
Operator Proficiency Check:	Valid to 31 October 2012	
Licence Proficiency Check:	Valid to 30 April 2013	
Line check:	Valid to 28 February 2013	
Medical certificate:	Valid to 14 January 2013	
Flying Experience:	Total all types:	798 hours
	Total on type:	569 hours
	Last 90 days:	163 hours
	Last 28 days:	49 hours
	Last 24 hours:	3 hours
Previous rest period:	15 hours 15 minutes	

1.5.2 G-CHCN

1.5.2.1 Commander

Age:	46 years	
Licence:	Airline Transport Pilot's Licence	
Licence expiry date:	25 Jul 2017	
Helicopter Ratings:	EC225 LP	
Operator Proficiency Check:	Valid to 28 February 2013	
Licence Proficiency Check:	Valid to 28 February 2013	
Line check:	Valid to 30 November 2012	
Medical certificate:	Valid to 7 March 2013	
Flying Experience:	Total all types:	11,964 hours
	Total on type:	933 hours
	Last 90 days:	108 hours
	Last 28 days:	6 hours
	Last 24 hours:	4 hours
Previous rest period:	16 hours 15 minutes	

1.5.2.2 Co-pilot

Age:	60 years	
Licence:	Airline Transport Pilot's Licence	
Licence expiry date:	11 February 2013	
Helicopter Ratings:	EC225 LP	
Operator Proficiency Check:	Valid to 28 February 2013	
Licence Proficiency Check:	Valid to 28 February 2013	
Line check:	Valid to 31 March 2013	
Medical certificate:	Valid to 11 February 2013	
Flying Experience:	Total all types:	15,728 hours
	Total on type:	1,334 hours
	Last 90 days:	71 hours
	Last 28 days:	42 hours
	Last 24 hours:	nil hours
Previous rest period:	10 days	

Both pilots on G-CHCN were aware of the accident to G-REDW on 10 May 2012 and had read information on the initial investigation findings. In particular, the commander had noted that the emergency lubrication system was reported as continuing to operate, despite the illumination of the MGB EMLUB caption. Furthermore, the co-pilot had used details of the previous accident as a scenario to support simulator training for other crews. As a result he was, familiar with the actions required for such an emergency.

1.6 Aircraft information

1.6.1 General

Manufacturer:	Eurocopter
Type:	EC225 LP Super Puma
Powerplants:	Two Turbomeca Makila 2A1 turboshaft engines

1.6.1.1 G-REDW

Manufacturer's serial number:	2734
Year of manufacture:	2009
Total airframe hours:	4,141 hrs
Total airframe cycles:	4,399 cycles
Certificate of Registration No:	G-REDW/R1
Registered owner:	Bond Offshore Helicopters Ltd
Date of issue:	27 August 2009
Issuing Authority:	Civil Aviation Authority
Certificate of Airworthiness:	Issued by the European Aviation Safety Agency in August 2009
Airworthiness Review Certificate:	Expired 26 August 2012

1.6.1.2 G-CHCN

Manufacturer's serial number:	2679
Year of manufacture:	2007
Total airframe hours:	5,956 hrs
Total airframe cycles:	6,328 cycles
Certificate of Registration No:	G-CHCN/R1
Registered owner:	CHC Scotia Ltd
Date of issue:	5 March 2008
Issuing Authority:	Civil Aviation Authority
Certificate of Airworthiness:	Issued by the European Aviation Safety Agency in March 2008
Airworthiness Review Certificate:	Expired 6 March 2013

1.6.2 Aircraft description

The EC225 LP first entered service in 2004 and was certified by the EASA[2] against the Joint Aviation Regulations (JAR) 29, change 1, effective 1 December 1999.

The EC225 LP is a twin-engine medium sized helicopter developed from the Eurocopter AS332 L2 Super Puma. The significant difference between the variants, concerning the transmission system, is that the EC225 LP is equipped with a five-bladed main rotor with a spheriflex rotor head and uprated Turbomeca Makila 2A/2A1 engines that can deliver approx 15% more torque to the main rotor system.

It is also equipped with a Modular Aircraft Recording Monitoring System (M'ARMS). M'ARMS incorporates a Combined Voice and Flight Data Recorder (CVFDR), Usage Monitoring System (UMS) and a Health and Usage Monitoring System (HUMS).

G-REDW and G-CHCN were both operated by two pilots and equipped with 19 passenger seats in the main cabin. They were also equipped with an emergency flotation system, a liferaft fitted in each sponson and a deployable crash position indicator (CPI).

1.6.3 Fleet experience

The helicopter manufacturer reported that at the time of the accident involving G-REDW, the AS332 variants[3] had accumulated more than 4 million flight hours and the EC225 LP variant approximately ¼ million flight hours. During this period there had been no reports of cracks occurring on bevel gear vertical shafts fitted to either the AS332 variants or EC225 LP helicopters.

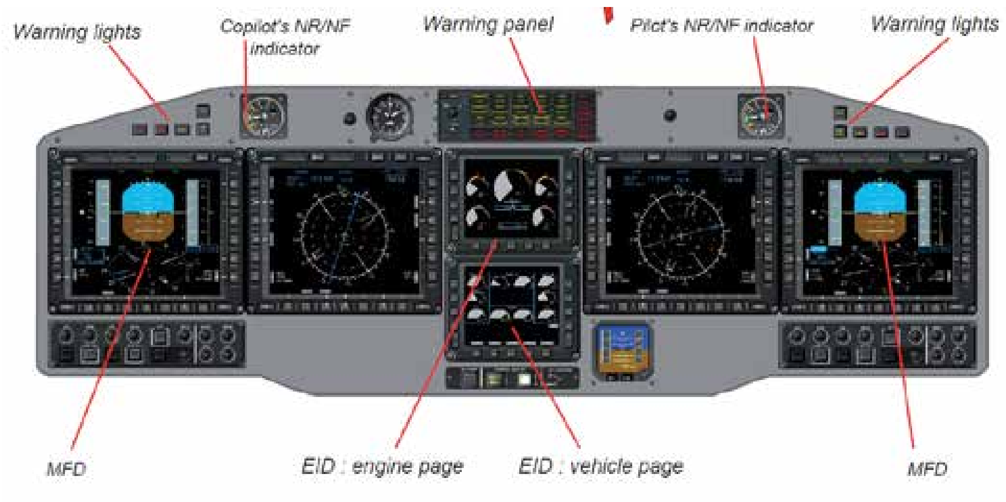
1.6.4 Alerting system

The EC225 LP alerting system uses visual and aural indicators. The visual system uses Red warning and Amber caution lights.

- A Red warning indicates that there is *'a serious operating danger and the pilot must react immediately'*.
- An Amber caution indicates that there is '*a reduction in the possibilities of an essential system or an abnormal operating condition'*.

[2] TCDS Number R002, EC225 LP type certification date 27 July 2004.
[3] This included the AS332 L1 and AS332 L2 helicopters. Collectively, the AS332 L1, AS332 L2 and EC225 LP are referred to as a Super Puma.

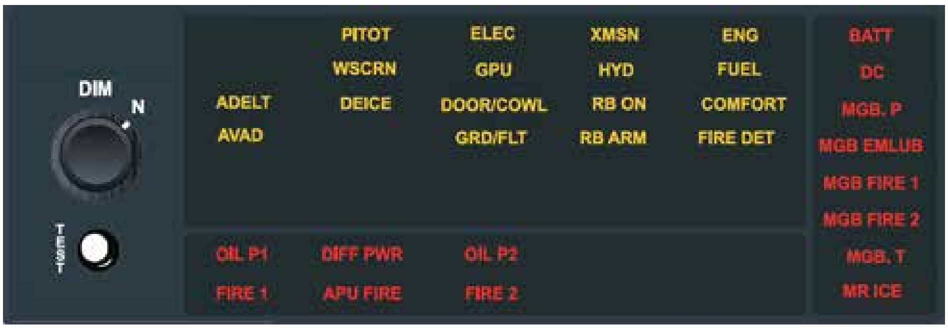
These are brought to the attention of the crew by captions displayed on a CWP and flashing master lights on the instrument panel. The status of the helicopter systems is displayed on Electronic Instrument Displays (EID) located in front of each pilot that forms part of the VMS. The instrument panel on the EC225 LP is shown at Figure 5 and the CWP at Figure 6.



(*Courtesy of Eurocopter*)

Figure 5

Instrument panel on the EC225 LP



(*Courtesy of Eurocopter*)

Figure 6

Central Warning Panel fitted to EC225 LP helicopters

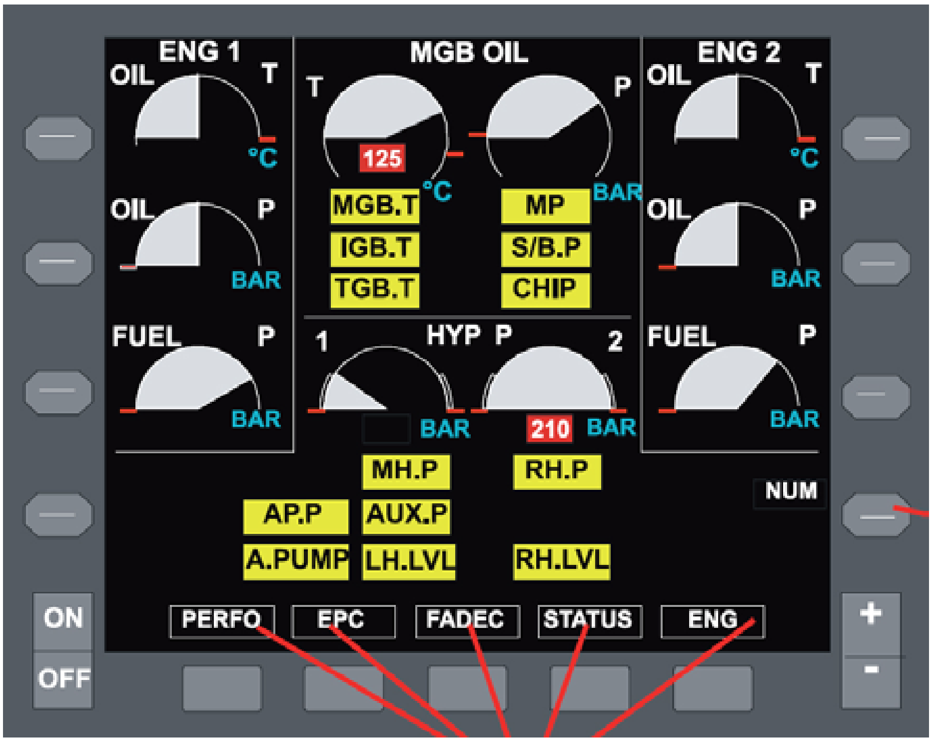
When either a Red warning light or Amber caution light illuminates on the CWP, a master WARN or CAUT light located in front of both pilots, flashes (Figure 7).

(*Courtesy of Eurocopter*)

Figure 7

Master lights on EC225 LP

The EID in front of each pilot displays the system status generated by the VMS and any relevant messages; the pilots can select which pages they wish to view. If a parameter in the VMS is outside the permitted limits marked by the yellow sector on the display, then the parameter is highlighted in amber. If a system parameter is outside the values indicated by the red lines, irrespective of the mode the pilot has selected, the page will be displayed on the EID and the value will appear in digital form and flash red (Figure 8).



(*Courtesy of Eurocopter*)

Figure 8

Electronic Instrument Display on EC225 LP

1.6.5 Transmission

1.6.5.1 Main rotor gearbox (MGB)

The purpose of the MGB is to transmit the power from the engines to the rotors while reducing the engine rotation speed of 23,000 rpm to the nominal main rotor speed of 265 rpm. It also provides the drive to the tail rotor transmission, accessory drives and the main and standby oil pumps.

The MGB consists of four interchangeable modules: the epicyclical reduction gear module, the main module and the left and right accessory modules.

1.6.5.1.1 Epicyclical reduction gear module

The epicyclical reduction gear module is mounted on top of the main module. It consists of two stages through which the rotational speed is reduced and the torque is increased prior to turning the main rotor drive shaft through a splined union.

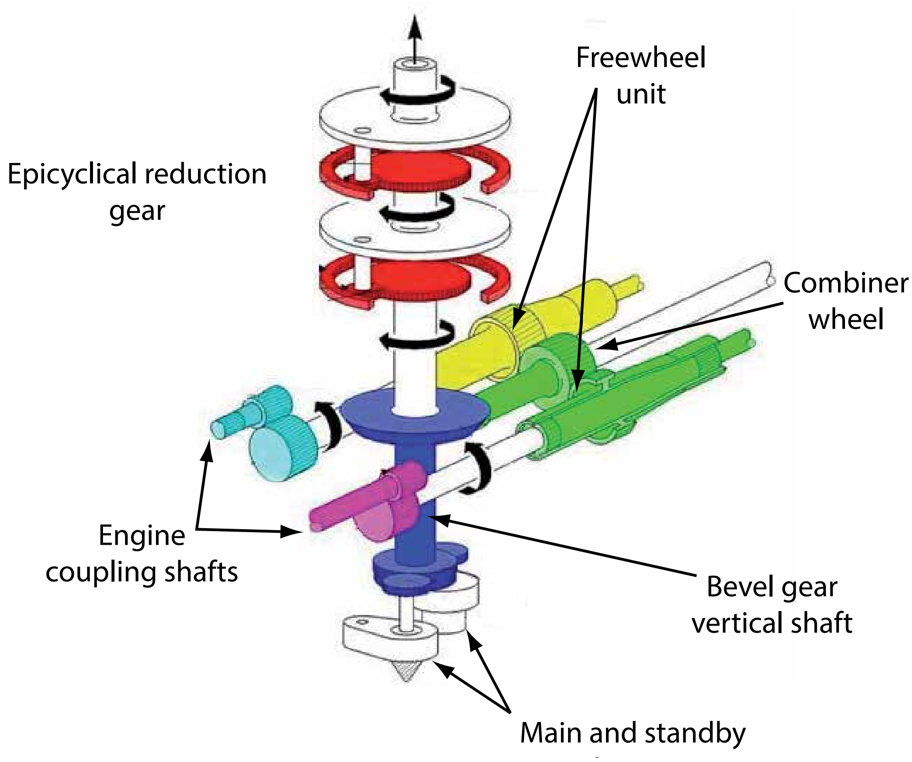
1.6.5.1.2 Main module

The main module is driven by the left and right engine coupling shafts through reduction gears and independent freewheel units. The freewheel units provide the drive to the accessory drives and the combiner wheel drives the shaft to the rear transmission components and the main bevel gear pinion. The pinion drives the bevel gear, which transmits the drive upwards through the first stage sun gear into the epicyclical reduction gear module. Two pinion gears, mounted at the bottom of the bevel gear vertical shaft, drive the main and standby oil pumps (Figure 9).

1.6.5.1.3 Bevel gear vertical shaft

The bevel gear vertical shaft (Figure 9) rotates at 2,405 rpm (40 Hz) and consists of a main bevel wheel and a vertical shaft that are joined together by an electron beam weld (Figure 10). A description of the weld is at Appendix B and the manufacturing process is described at Appendix C. To ensure the integrity of the weld, the disrupted material at the end of the weld is removed by drilling and reaming a 4.2 mm diameter hole, which the design definition allows to be opened up to 4.4 mm. The inner and outer surface of the weld region is then machined to remove the cap and root of the weld.

The shaft is manufactured from a high strength low alloy steel and, as with a number of other steel components used in the gearbox, is not given a surface treatment to protect it from corrosion. Instead, the shaft relies on the oil mist within the gearbox and the application of a protective oil coating during the manufacturing process for corrosion protection.



(*Courtesy of Eurocopter*)

Figure 9

Schematic of EC225 LP MGB

The shaft is supported in the gearbox casing by two upper bearings (one roller and one ball) mounted adjacent to each other above the bevel gear wheel, and a lower roller bearing mounted at the bottom of the vertical shaft above the oil pump drive wheels. Following the failure of the bevel gear vertical shaft, the bevel gear wheel is only supported by the two upper bearings.

Oil jets spray oil through two 29 mm 'lubrication holes' positioned opposite each other on the vertical shaft. Under centrifugal force, this oil forms a layer on the inner surface of the vertical shaft, which is used to lubricate the bevel gear first stage sun gear upper coupling splines. A PTFE[4] plug is fitted in the 4.2 mm hole in the weld in order to prevent this film of oil from leaking through the hole.

[4] Polytetrafluoroethylene.

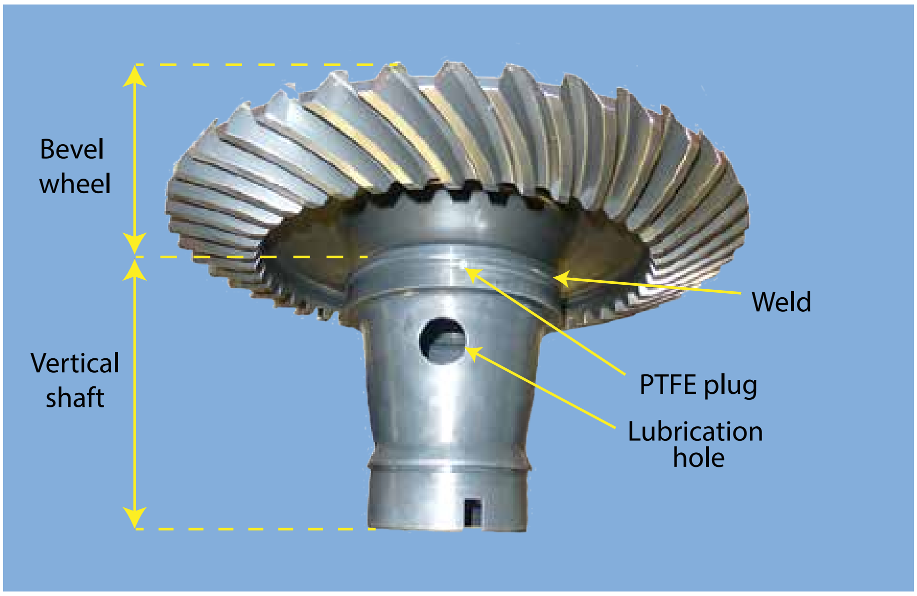


Figure 10

Bevel gear vertical shaft

1.6.5.2 Development of the MGB fitted to the EC225 LP

The MGB fitted to the EC225 LP is of a similar design to the MGB fitted to the AS332 L2, but can deliver approximately 15% more torque to the rotor head. The bevel gear vertical shafts originally designed for the AS332 variants were manufactured from 16NCD13 steel. The gear teeth and pinion splines on these shafts were surface hardened by a process called 'carburising' prior to the bevel gear wheel being welded to the vertical shaft. The manufacturer's design did not require the vertical shaft, or the part of the bevel gear wheel that is welded to the vertical shaft, to be surface hardened.

To accommodate the increased loads on the bevel gear teeth, and the elevated temperatures in the MGB that occur during operation of the emergency lubrication system, it was necessary to change the surface hardening process for the EC225 LP shaft. This required the use of a different high strength steel, 32CDV13[5], that has a similar strength to 16NCD13 steel, and a case hardening process called 'nitriding'. The vertical shaft, which is also manufactured from 32CDV13 steel, is not subject to the nitriding process.

In this report, the bevel gear vertical shaft manufactured from 16NCD13 steel will be referred to as the 'AS332 shaft'[6] and the bevel gear vertical shaft manufactured from 32CDV13 steel will be referred to as the 'EC225 shaft'[7].

5 The bevel gear was manufactured from E32CDV13, Class 4, steel and the vertical shaft from E32CDV13, Class 3, steel.
6 Part number 331A313115.
7 Part number 332A325101.

1.6.5.3 Differences in the geometry of the EC225 and AS332 shafts

The EC225 shaft is 1.2 mm thicker than the AS332 shaft in the area of the weld. Consequently, although the torque is greater in the EC225 gearbox, the maximum stress at the 4.2 mm hole is similar on both types of shaft (Figure 11).

There is also a slight difference in the profile on the inside surface of the EC225 shaft adjacent to the weld that resulted in the introduction of a 3 mm radius. This change in profile was carried out for ease of manufacturing and the new feature is described in this report as the 'inner radius'. The size and profile of the pinion splines and gear teeth is the same on both shafts; therefore, with a 15% increase in torque there will be a proportional increase in the forces on the splines and teeth on the shaft fitted to the EC225. This increase in load has resulted in an increase in the wear of the splines that drive the first stage sun gear.

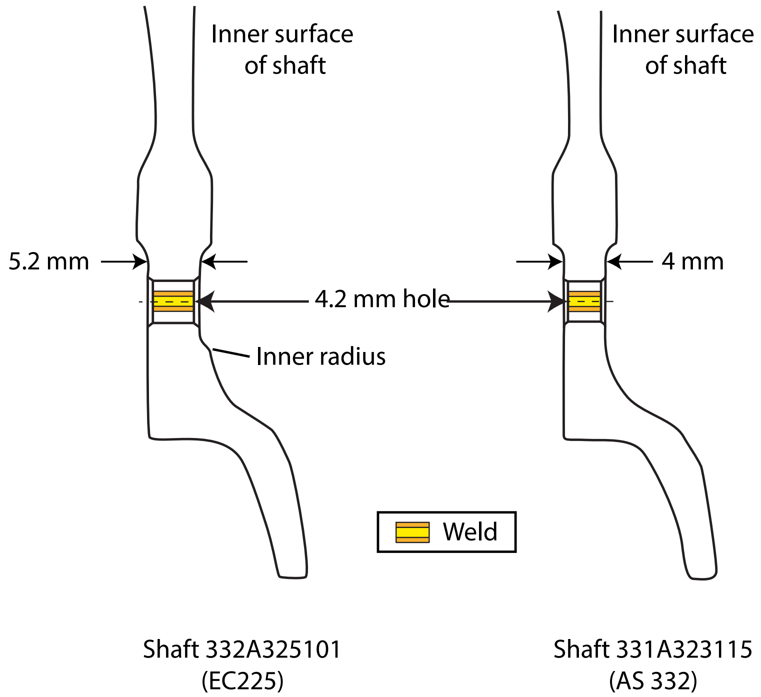


Figure 11

Significant differences between the EC225 and AS332 shafts

An additional difference between the shafts is the upper roller bearings. On the AS332 shaft the roller bearing comes complete with its own internal race. However, on the EC225 shaft, the inner race is integral to the shaft, with the removable bearing assembly only comprising the outer race and rollers.

The EC225 shaft can also be fitted to the AS332 L2. Approximately 732 EC225 shafts had been manufactured when the accident involving G-REDW occurred.

1.6.5.4 Life of the bevel gear vertical shaft

Bevel gear vertical shafts fitted to the EC225 LP have a life of 20,000 flying hours and shafts[8] fitted to the AS332 L2 have a life of 50,000 flying hours. Shafts are overhauled at the same time as the MGB. For the EC225 LP the MGB is overhauled every 2,000 flying hours and for and the AS332 L2 every 3,000 flying hours.

According to the manufacturer, at the time of the accident involving G-CHCN, no shaft manufactured from 32CDV13 steel had flown sufficient hours to reach its second overhaul at 4,000 flying hours. Moreover, approximately 63% of the shafts fitted to the EC225 LP were scrapped during their first overhaul, of which approximately 50% were due to wear on the splines that drive the first stage sun gear. The manufacturer was also of the opinion that the shaft from G-CHCN (M122) was the fleet leader[9].

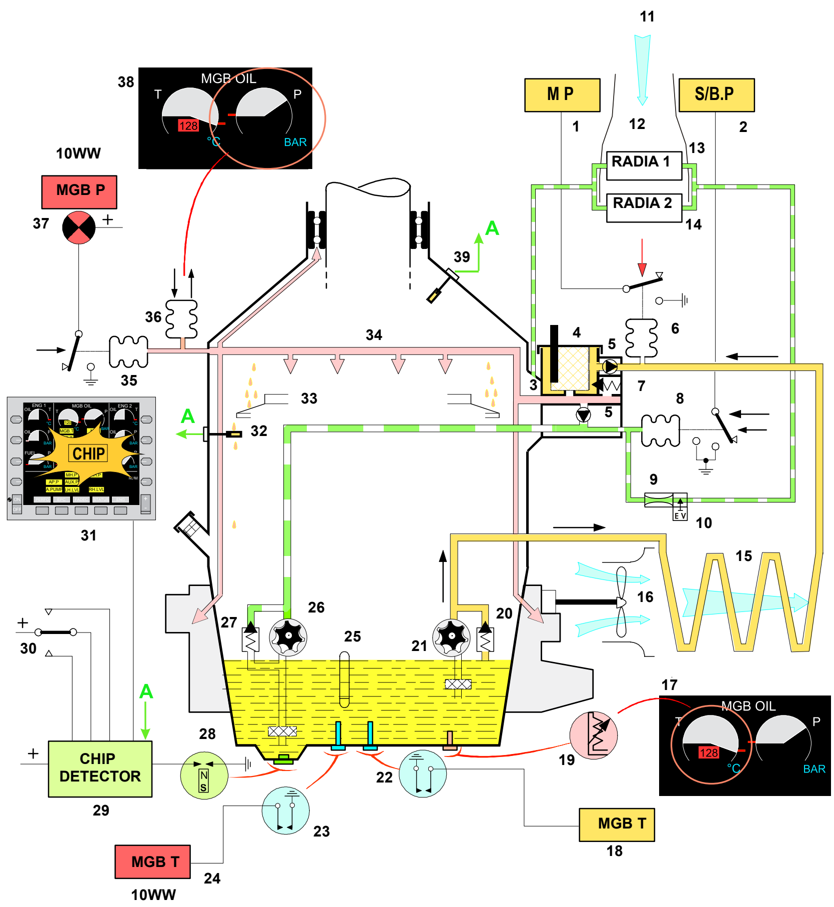
1.6.5.5 MGB oil system

1.6.5.5.1 Overview

The MGB lubrication system has an oil capacity of approximately 30 litres and consists of the main system and a standby system. The main system is supplied by the main oil pump which passes hot oil through an external heat exchanger and filter. In the event of a drop in main oil pressure the standby oil pump automatically delivers oil to the MGB. The oil supply from the standby pump does not pass through the oil filter, hence this oil supply is not filtered. On G-REDW and G-CHCN the oil from the standby oil pump can also be cooled by passing the oil through an oil-to-air heat exchanger.

A vent in the MGB casing and the rotor mast ensures that the pressure within the gearbox casing remains at atmospheric pressure. The EC225 LP is also fitted with an emergency lubrication system. A schematic of the MGB oil system is at Figure 12.

[8] Both the EC225 and AS332 shafts can be fitted to the AS332 L2.
[9] The fleet leader is the shaft that has accumulated the most flight hours.



1 - Cautionary indication on VMS (P<3.7 bars)
2 - Cautionary "Stand-by low pressure" indication on VMS (2 thresholds: P< 2.6 bars or P< 1 bar)
3 - Return line trough the endoscope cap
4 - Oil filter (filtration capacity 25 μ) with clogging indicator
5 - Check valves (set to 0.1 bar)
6 - Pressure switch (P<3.7 bars)
7 - Filter bypass, Δ P 8 bars (opens in the event of clogging)
8 - Dual threshold pressure switch (P< 2.6 bars or P< 1 bar)
9 - Flow diviser part
10 - Electro-valve
11 - Dynamic air scoop
12 - Air duct
13 - Cooler type 2
14 - Cooler type 1
15 - "Oil-to-air" heat exchanger
16 - MGB driven fan
17 - Oil temperature indicator on VMS
18 - Cautionary "excessive oil temperature" indication
19 - Oil temperature probe
20 - Pressure relief valve set to 10 bars
21 - Main oil pump
22 - Oil temperature thermo-switch
23 - Excessive oil temperature thermo-switch
24 - Excessive oil temperature red warning light on 10WW
25 - Oil level sight
26 - Stand-by oil pump
27 - Pressure relief valve set to 3.3 bars
28 - Sump magnetic chip detector
29 - Chip detection and destruction unit
30 - Chip destruction switch
31 - Cautionary indication (on VMS): metallic chips
32 - Epicyclic magnetic chip detector
33 - Oil deflector (prevents pollution due to particles)
34 - Lubrication diffuser
35 - Pressure switch (P<0.4 bars)
36 - Oil pressure transmitter
37 - MGB Oil pressure drop red warning light on 10WW panel (the MGB is no longer lubricated)
38 - Oil pressure indicator on the VMS
39 - Mast magnetic chip detector

(Courtesy of Eurocopter)

Figure 12

Schematic of EC225 LP MGB lubrication system

1.6.5.5.2 Oil delivery

The main and standby oil pumps are mounted in the MGB sump and are driven by separate pinions mounted at the bottom of the bevel gear vertical shaft. In order to prevent the loss of the gearbox oil following a leak from an external pipe or component, the intake for the main oil pump is uncovered when the oil level drops below 8 litres.

The main oil pump supplies oil at a pressure of 10 bar through external pipes to an oil-to-air heat exchanger. The cooled oil then passes through an oil filter and into the distribution ramp where it is directed within the gearbox. The oil returns under gravity to the sump. The standby oil pump supplies oil at a pressure of 3.3 bar and will deliver oil directly to the distribution ramp when the main pump pressure drops below 3.7 bar.

1.6.5.5.3 Indication and warnings

An oil pressure transmitter located in the oil distribution ramp and an oil temperature probe located in the sump, display the MGB oil pressure and temperature on the VMS.

When the oil pressure at the input to the oil filter drops below 3.7 bar, the MP amber caution illuminates on the VMS.

When the main oil pressure drops below 3.7 bar and the standby oil pump drops below 2.6 bar, the parameter S/B.P and MP amber cautions illuminate on the VMS, and the value of the oil pressure displayed on the EID flashes red.

When the oil pressure at the input to the oil filter drops below 0.4 bar, the red warning light MGB.P illuminates on the CWP and the red master warning light flashes.

When the oil temperature in the MGB exceeds 128°C the MGB.T amber caution illuminates on the Vehicle page of the VMS.

1.6.5.5.4 Magnetic chip detectors

The MGB is equipped with two magnetic chip detectors; one mounted in the sump and one mounted on the outer edge of the epicyclical reduction gear casing. A magnetic chip detector is also mounted in the main rotor mast. The warnings generated by the chip detectors are recorded on the HUMS and the CVFDR. None of the warnings are latched.

The sump magnetic chip detector causes an amber CHIP warning light to illuminate on the VMS when conductive debris bridges the magnetic contacts that are spaced 2.28 mm apart on the end of the detector. If the warning light illuminates, the pilot can select PULSE on the chip destruction switch in the cockpit. This action causes a high current to flow through the contact bridging the magnetic contacts which, if the particle is small enough, will destroy it. The warning light will then extinguish.

The epicyclic magnetic chip detector works in the same manner as the sump chip detector and illuminates the CHIP caution on the VMS. However, this detector does not have the 'pulse' facility to destroy small pieces of debris which bridge the gap.

1.6.5.6 Emergency lubrication system

1.6.5.6.1 Certification requirements

The certification requirements (JAR 29.927) required the helicopter to be capable of continued safe flight, at prescribed torque and main rotor speeds, for at least 30 minutes following the loss of the MGB lubrication system. To meet this requirement, the helicopter manufacturer introduced an emergency lubrication system[10] on the EC225 LP. The system uses a mixture of glycol and water, called Hydrosafe 620, to cool and lubricate the MGB.

Certification of the system included a test on a ground rig in which the oil was drained from a MGB and pressurised air (simulating engine bleed-air) and Hydrosafe 620 were sprayed into the gearbox. The test was run for more than 30 minutes and the manufacturer concluded:

> *'Considering all these elements the EC225 loss of oil test demonstrates sufficient safety margin to allow 30 minutes of flight at minimum flight power with the back up system on.'*

Although the emergency lubrication sub-systems were tested individually, no test was carried out on the complete system during certification, either on a test rig or installed on a helicopter.

1.6.5.6.2 Emergency lubrication system description

The emergency lubrication system is manually activated by the selection of the EMLUB SHOT push-button on the MGB control box, located on the cockpit left overhead panel. The SHOT light illuminates when the red MGB.P warning appears on the CWP.

[10] In some of the manufacturer's documents the emergency lubrication system is also described as the back-up lubrication system.

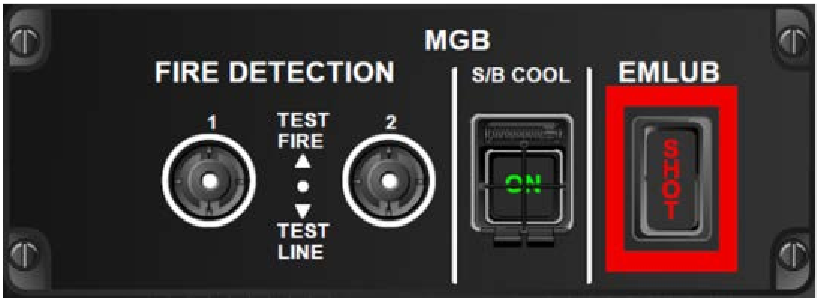


Figure 13

Overhead panel showing EMLUB SHOT button

The emergency lubrication system (Figure 14) comprises:

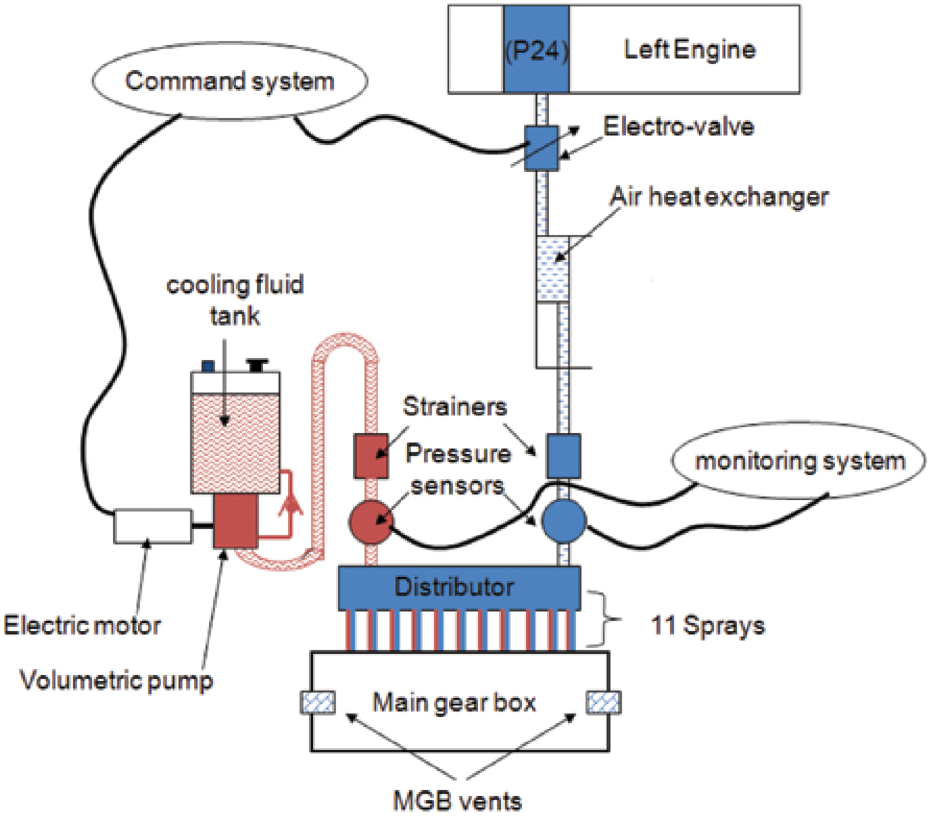
- a bleed-air supply from the left engine via a bleed-air electro-valve and heat exchanger,
- a pumped Hydrosafe 620 supply from an 11 litre reservoir,
- a series of small pipes and nozzles to deliver the Hydrosafe 620 in a spray to the MGB,
- pressure sensors/switches in the bleed-air and Hydrosafe 620 delivery lines,
- a dedicated Printed Circuit Board (PCB) for monitoring and command of the system.

The MGB EMLUB caption will illuminate on the CWP if low pressure is detected by either of two pressure switches, one in the Hydrosafe 620 line and the other in the bleed-air line. It will also illuminate if there is an erroneous signal detected by the PCB. The caption is inhibited for approximately 30 seconds after the emergency lubrication system is activated, to allow the system to reach a steady-state.

The low pressure signal is generated by either the Hydrosafe 620 or bleed-air pressure switches if the pressure does not exceed a specified threshold, p_{on}, when the system is activated, or if the pressure subsequently falls below a specified threshold, p_{off}.

The specified range for p_{on} for each pressure switch is between 0.6 and 1.0 bar (relative to ambient).

The MGB EMLUB caption is not latched therefore if the pressure in both delivery increases above p_{on} and the signal detected by the PCB is valid, the light will extinguish.



(*Courtesy of Eurocopter)*

Figure 14

Schematic of the Emergency Lubrication System

1.6.6 Survival equipment

1.6.6.1 Crash position indicator (CPI)

G-REDW and G-CHCN were both equipped with an externally-mounted, deployable Type 15-503 CPI system. The CPI is a type of Automatically Deployable Emergency Locator Transmitter (ADELT), which is a primary radio location aid designed to activate automatically in the event of an aircraft accident so that the aircraft and its occupants can be located quickly.

1.6.6.1.1 ADELT regulations and certification

JAR-OPS 3.820, *Automatic Emergency Locator Transmitter*, paragraph (b), which was valid at the time of certification of the EC225 LP, provides the operational requirement for an ADELT to be fitted to helicopters which operate over water in a hostile environment:

> *'An operator shall not operate a helicopter in Performance Class 1 or 2 on a flight over water in a hostile environment as defined in JAR-OPS 3.480(a)(12)(ii)(A) at a distance from land corresponding to more than 10 minutes flying time at normal cruising speed, on a flight in support of or in connection with the offshore exploitation of mineral resources (including gas), unless it is equipped with an Automatically Deployable Emergency Locator Transmitter.'*

The certification requirements in JAR-29, valid when the EC225 LP was certified, did not contain any requirements relating to the functionality, location or installation of the components in an ADELT system, and neither do the current CS-29 requirements. However, detailed requirements for the performance of an Emergency Locator Transmitter (ELT) as a stand-alone item were specified in UK CAA Specification No. 16 *'Automatically Deployable Emergency Locator Transmitter for Helicopters'* (Issue 2, December 1991) and in ETSO-2C126[11] *'406 MHz Emergency Locator Transmitter (ELT)'* dated October 2003, both valid at the time of certification.

The Type 15-503 CPI system met the operational requirements of JAR-OPS 3.820 and held approvals in accordance with CAA Specification No. 16 and ETSO-2C126.

ETSO-2C126 states that the ELT must comply with the Minimum Operational Performance Specifications[12] (MOPS) stipulated in EUROCAE[13] document ED-62 '*Minimum Operational Performance Specification for Aircraft Emergency Locator Transmitters (121.5/243 MHz and 406 MHz),* dated May 1990.

ED-62 describes the minimum necessary performance criteria for an ELT and outlines the tests which must be carried out to demonstrate its performance. ED-62 defines an ADELT as follows:

> a. *'This type of ELT is intended to be rigidly attached to the aircraft before a crash and ejected and deployed,*
>
> b. *manually (during the crash sequence), or*
>
> c. *automatically (after the crash sensor has determined that a crash has occurred.'*

11 European Technical Standard Order (ETSO).
12 The minimum necessary performance to satisfy a regulatory requirement.
13 European Organisation For Civil Aviation Equipment – an organisation comprising equipment and airframe manufacturers, regulators and other industry representatives, which deals with aviation standardisation and publishes documents for use in the regulation of aviation equipment.

The current requirements for ELTs are specified in ETSO-C126a, dated July 2012 and ED-62A 'Minimum Operational Performance Specification for Aircraft Emergency Locator Transmitters 406 MHz and 121.5 MHz (Optional 243 MHz)', dated February 2009, which supersedes ED-62. ED-62A defines an ADELT as follows:

> *'This type of ELT is intended to be rigidly attached to the aircraft before a crash and automatically deployed after the crash sensor has determined that a crash has occurred. This type of ELT shall float in water and is intended to aid SAR teams in locating the crash site.'*

Both ED-62 and ED-62A contain the following requirement, applicable to ADELTs:

> *'The equipment shall have provision for manual deployment before a crash, and automatic deployment during a crash.'*

1.6.6.1.2 CPI system description

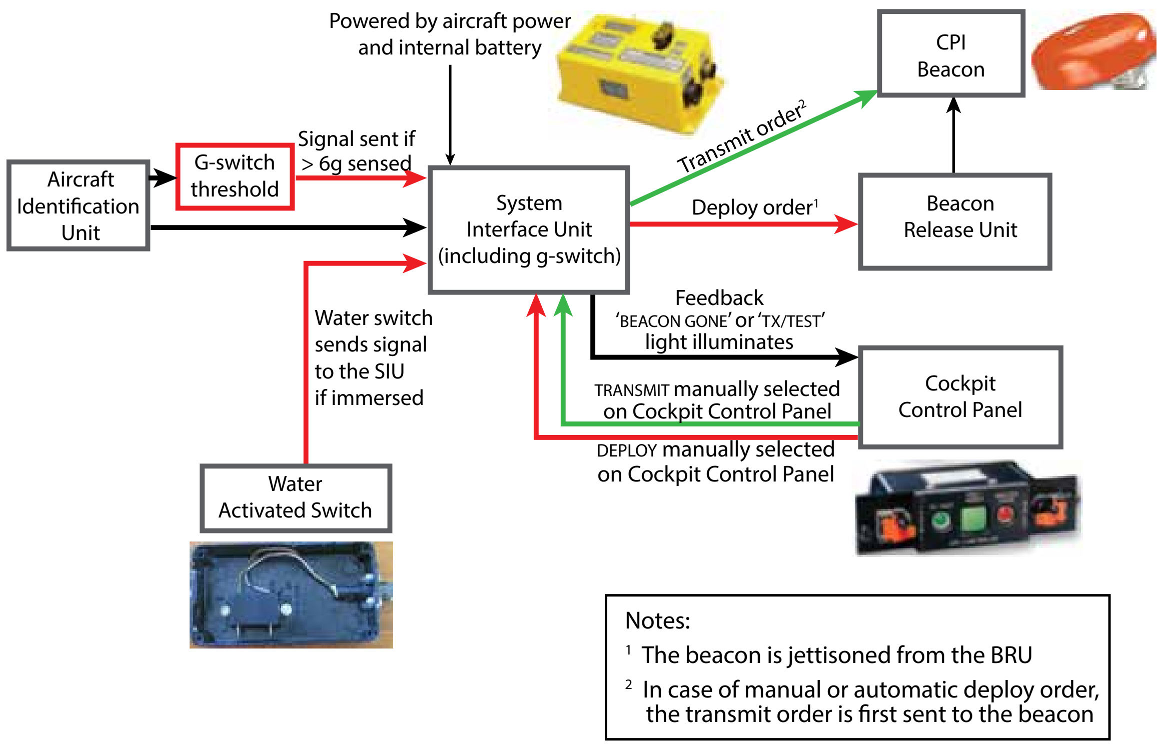
The CPI system consists of a radio beacon, a beacon release unit (BRU), a system interface unit (SIU), a cockpit control panel, a water-activated switch and an aircraft identification unit (Figure 15).

The wiring for the CPI system is integrated with the rest of the helicopter's wiring looms, and is not specifically protected against water ingress. The electrical connectors in the CPI system, however, conform to an industry standard specification[14] which ensures good performance when submerged in water at shallow depths.

The specific CPI system installation and the location and modification standard of the CPI system components, can vary considerably between helicopters and may depend on whether the CPI was fitted at the time of initial manufacture, or retrospectively under a supplemental type certificate. The CPI systems on G-REDW and G-CHCN were installed during manufacture and were of the Type 15-503-134-1 series standard.

On G-REDW, the CPI beacon was externally mounted on the lower left side of the baggage hold at the rear of the main cabin. The BRU was mounted immediately behind the CPI beacon and the SIU and aircraft identification unit were located inside the baggage hold, close to the beacon.

[14] Military Specification Mil-C-26482, Electrical Connectors.



(Modified with permission of Eurocopter)

Figure 15

Schematic of crash position indicator system

On G-CHCN, the CPI beacon was externally mounted on the left side of the tail boom, just aft of the main cabin and the helicopter transport joint. The BRU was mounted immediately behind the CPI beacon and the SIU and aircraft identification unit were located inside the tail boom close to the beacon. Figure 16 shows the location of the CPI components on G-CHCN.

On both helicopters the water-activated switch was mounted just above floor level in the passenger cabin, behind the cabin trim and slightly aft of the left main cabin door aperture. The cockpit control panel was mounted at the rear of the centre console.

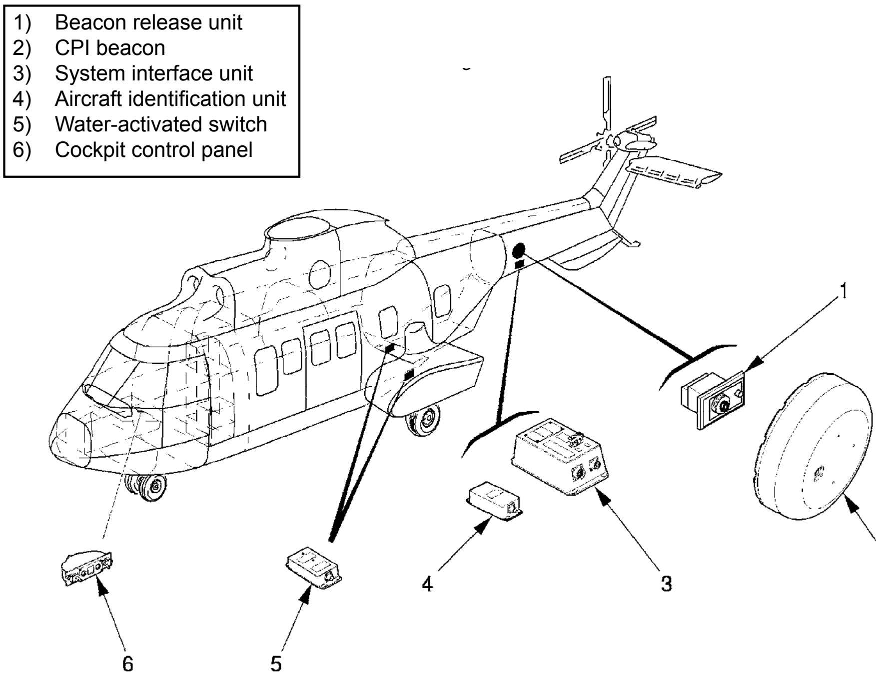


Figure 16

Typical location of CPI components with the CPI beacon on tail boom
(as per the G-CHCN configuration)

1.6.6.1.3 Operation of the CPI

The CPI system receives electrical power either from the helicopter or an internal battery within the SIU, which allows activation of the system for up to two hours after helicopter power is lost.

Deployment of the CPI is achieved by any one of the following:

- an acceleration of more than 6g in any direction detected by g-switches in the SIU, based on information stored in the aircraft identification unit.
- manual operation of the DEPLOY switch on the cockpit control panel.
- immersion of the water-activated switch.

The BRU uses a small actuator and compressed spring to project the beacon away from the helicopter. The beacon is designed to float, with automatic transmission of the beacon signal commencing once it has separated from the helicopter.

Transmission of the CPI can be manually selected by the crew, without deploying the beacon, by selecting the TRANSMIT switch on the cockpit control panel. A green TX/TEST light illuminates on the cockpit control unit when the beacon is transmitting, and a red BEACON GONE light illuminates when the beacon has deployed.

1.6.6.1.4 Water-activated switch

The water-activated switch is a box comprising two exposed electrical contacts, a capacitor and a relay. Two holes in the bottom of the box allow water to enter when it is immersed which allows the contacts to complete an electrical circuit to the BRU. This charges a capacitor and after 5 to 10 seconds it activates the relay which sends a deploy signal to the BRU via the SIU, to deploy the beacon. If the connection between the contacts in the water-activated switch is interrupted during this period, for example due to fluctuations in the water level, the capacitor discharges without activating the relay, thus resetting the delay period until the connection is remade.

1.6.6.1.5 CPI transmission

Once activated, the CPI beacon transmits coded identification signals on 406.025 MHz, which can be detected by the international COSPAS/SARSAT[15] distress alerting system. The transmitted signal from the CPI beacon takes the form of short pulses spaced at approximately 50-second intervals. The beacon also transmits on 121.5 MHz.

The system uses geostationary (GEO) satellites to detect the initial emergency transmission, whilst low earth orbit (LEO) satellites receive signals to enable the approximate position of the point of origin to be established. This can take some time as at least two LEO satellites need to be in receipt of an unobstructed signal for triangulation to take place. Although the satellites are capable of receiving and relaying a Global Positioning System (GPS) position message, neither the G-REDW nor G-CHCN CPIs were GPS-enabled.

1.6.6.2 Liferafts

The EC225 LP is equipped with two double-sided Survitec/RFD Type 18R MK3 inflatable liferafts. Each has a deployable canopy and capacity for 18 occupants, with a nominal overload capacity of 27. They are mounted, together with their inflation systems, in the forward sections of the helicopter's sponsons, on either side of the fuselage.

[15] Cosmicheskaya Sistyema Poiska Avariynich Sudov / Search and Rescue Satellite.

The liferafts can be deployed by any one of three methods:

- Operation of a D-ring, positioned near the top of the bulkhead behind each flight crew position, which deploys and inflates the liferaft on the corresponding side of the helicopter.
- Operation of deployment handles, positioned externally in recesses on each side of the helicopter just aft of each cabin door, which deploys and inflates the liferaft on that side of the helicopter.
- Removing either liferaft cover from its sponson and pulling the inflation D-ring inside.

The main features of the liferaft are shown in Figure 17. A rescue pack is attached to each liferaft by a bridle/rescue pack line. The pack contains a number of items, including flares, water, anti-seasickness tablets, an 'Immediate Action' survival leaflet, an aircrew survival flip-card and a personal locating beacon (PLB).

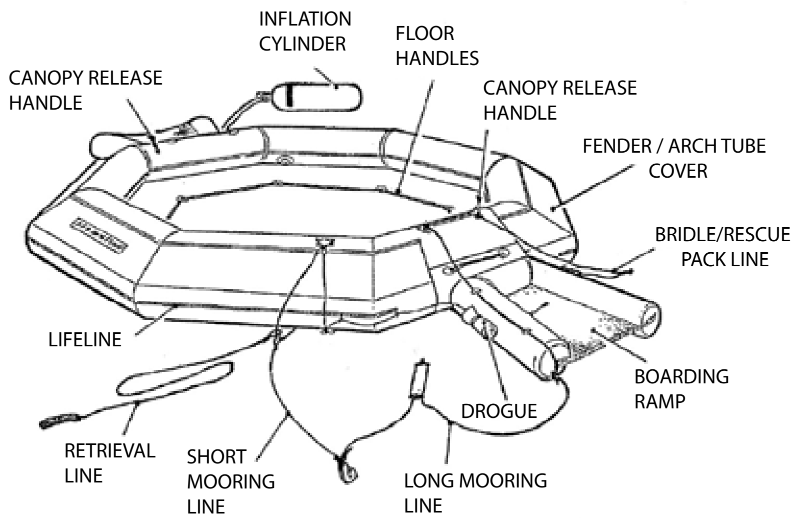


Figure 17

Type 18R MK3 liferaft showing the various lines

When the liferaft is deployed, it remains attached to the helicopter by a 2 m 'short' mooring line and a 12 m 'long' mooring line that are both tied together with a snap hook and connected to a bracket inside the sponson. There is also a retrieval line attached to the raft with a strap on the end that is fastened with Velcro to the inside of the sponson near the door. The short mooring line enables the liferaft to remain close to the helicopter for boarding, while the retrieval line is used to pull the raft back to the door if it starts to drift away. Procedures call for the short mooring line to be cut as soon as all passengers are on board, using the safety knife attached to the inside of the liferaft. The long mooring line is designed to keep the liferaft attached to the helicopter at a distance; co-location of the liferaft and helicopter assists the search and rescue operation. The long mooring line is designed to break if the helicopter sinks.

1.6.6.3 Aircraft exits

The passenger cabin is fitted with two large passenger doors, one each side of the fuselage, approximately midway along the length of the cabin. Normal access for embarkation and disembarkation is through the left main door but both doors are available for emergency use.

For normal operation, the doors initially move outboard from their closed positions in their apertures. They then slide forward, on rails, along the outside of the cabin, towards the fully open position. Once in this position they fit closely alongside the cabin outer skin with the door covering two cabin windows, one of which is an emergency exit, thereby denying its use as an emergency exit [16].

During an emergency evacuation the manufacturer's emergency checklist (Appendix A) requires the cabin doors to be jettisoned. To achieve this, a D-ring is positioned in a recess on the cabin wall, beneath a transparent cover, approximately 23 cm forward of the upper forward corner of the door aperture. The D-ring can be pulled to enable the doors to fall vertically from their attachments. Additional jettison release handles are positioned adjacent to each door aperture in a recess on the outside of the cabin above the leading edge of the sponson.

The helicopter cockpit has two hinged exit doors, positioned on either side of the fuselage. The doors are jettisoned in an emergency evacuation by operating an external or internal jettison handle located on the forward frame of the door aperture, which releases the door's hinge pins. Once the jettison handle has been operated the doors need to be pushed manually for them to depart the helicopter.

1.6.6.4 Flotation equipment

Both G-CHCN and G-REDW were equipped with an emergency flotation system consisting of four cylindrically-shaped inflatable floats. Two of the floats were attached on either side of the helicopter's nose and two were attached on either side of the sponsons. The four floats were inflated by compressed helium

[16] Although an emergency exit is covered when the door is slid open, the remaining exits still meet the regulatory requirements on the minimum number of emergency exits.

contained in three bottles. The system could be armed by selecting a switch on the centre console. Once armed, inflation would be initiated automatically following water immersion, or the crew could manually inflate the floats by pressing a button on the collective control or on the centre console.

In both the G-CHCN and G-REDW accidents the crew manually initiated inflation before the ditching.

1.6.7 Maintenance information

1.6.7.1 History of the bevel gear vertical shafts

1.6.7.1.1 G-REDW

The bevel gear vertical shaft, serial number M385, fitted to the MGB on G-REDW was one of a batch of ten that were given the serial numbers M382 to M391. Both parts of the shaft were welded together in August 2010 and the final inspection at the end of the manufacturing process was carried out in March 2011. No manufacturing concessions were issued for the area of the weld or any of the bearing faces that support the shaft. A description of the manufacturing process is at Appendix C.

The shaft remained in the manufacturer's stores until it was fitted to MGB, serial number M5038, and then G-REDW in March 2012. At the time of the accident, the shaft had operated for 167 flying hours and approximately 20 million cycles[17].

1.6.7.1.2 G-CHCN

The bevel gear vertical shaft, serial number M122, fitted to the MGB on G-CHCN was one of a batch of nine that were given the serial numbers M118 to M126. Both parts of the shaft were welded together in March 2008 and the final inspection at the end of the manufacturing process was carried out in April 2008.

A test specimen is welded and examined at the start and end of each batch to ensure the integrity and dimensions of the weld are within design limits. A non-conformity report was raised for the internal and external width of the weld on the test specimen associated with the batch that included shaft M122. The report stated that the width of the weld was 0.45 mm and 0.3 mm outside the drawing tolerance. A non-conformity was also raised for the upper roller bearing face that was 0.002 mm outside the drawing tolerance. Concessions were issued for these features.

17 One cycle is one revolution of the shaft.

The shaft was fitted to MGB serial number M5081 and then to an EC225 LP helicopter, serial number 2716, in May 2008. After having flown approximately 10 hours, the MGB was removed in order to embody modification 0752522[18]. The MGB was refitted to the same helicopter, which was then given the registration LN-OJE and delivered to the operator in May 2009. The helicopter operated out of Kristiansund in Norway in support of the oil and gas industry in the North Sea until May 2011 when, at 2,032 flying hours, the MGB was removed and sent to an independent overhaul facility. Following overhaul, the MGB was fitted by the operator to G-CHCN in June 2011 which operated from Aberdeen. At the time of the accident the shaft had operated for 3,845 flying hours and approximately 553 million cycles, and was the fleet leader.

1.6.7.2 Test of the emergency lubrication system

Following the replacement of the MGB on G-REDW, the operator carried out a functional check of the emergency lubrication system on 18 March 2012. No faults were reported.

On 1 April 2012 the operator of G-CHCN carried out the 825-hour functional check of the emergency lubrication system. No faults were reported.

The Aircraft Maintenance Manual (AMM) functional check used the helicopter's maintenance panel and tested the system in ambient conditions without the bleed-air being supplied from the engine. As the maintenance panel bypassed the pressure switches to test the emergency lubrication system warning, the pressure switch thresholds and function were not tested.

1.7 Meteorological information

1.7.1 Meteorological information for 10 May 2012 (G-REDW)

The flight crew reported that at the time of the loss of the MGB oil pressure, they were flying in IMC.

Aberdeen ATIS information 'P' issued at 1050 hrs reported a surface wind from 060º at 13 kt, visibility 9,000 m, light rain, scattered cloud at 1,400 ft, scattered cloud at 2,500 ft, broken cloud at 4,400 ft, temperature +8ºC, dewpoint +5ºC and pressure 1007 hPa.

The weather conditions at the accident site as reported by one of the helicopters attending the scene were: showers in the vicinity, good visibility below the cloud with a cloud ceiling from between 600 ft to 700 ft. One helicopter crew reported that the sea state was slight to moderate. The sea surface temperature was 8ºC.

[18] Modification 0752522 was the removal of the ring of magnets in the epicyclic module.

1.7.2 Meteorological information for 22 October 2012 (G-CHCN)

The crew reported flying VFR on top of a layer of cloud with tops of about 2,000 ft amsl at the time of the loss of MGB oil pressure. They remember becoming visual with the sea on descending through the cloud layer at about 300 ft amsl.

The area was subject to a ridge of high pressure with a weak front to the north. This gave light winds of around 5 kt, little precipitation and patchy cloud, some of which was either very low cloud or fog. The air temperature at the surface was about 9ºC.

An aftercast reported the sea state as moderate with a swell from a west-south-westerly direction of between 1.5 and 2.0 m in height. The crew estimated the swell they experienced to be 1 m. The sea surface temperature was reported as being 11ºC.

1.8 Aids to navigation

Not applicable to this investigation.

1.9 Communications

Both G-REDW and G-CHCN were equipped with VHF radios, including a separate VHF radio for maritime communications. Records of radio transmissions between both helicopters and ATC were available from the ATC recording media and were also recorded on the CVFDRs of both helicopters.

G-REDW received a radar service from Aberdeen ATC on frequency 134.100 MHz during the flight. This frequency covers a sector of airspace extending out to 80 nm to the east of Aberdeen Airport. There was no alternative frequency available in the area of the accident for Aberdeen ATC to utilise. Therefore, G-REDW could not be transferred to a discrete frequency, nor could other aircraft operating on 134.100 MHz be transferred, unless they were within range of another service.

G-CHCN was in contact with Sumburgh radar ATC on 131.100 MHz at the time of the accident. The crew used the separate VHF maritime radio, emergency channel 16, to contact the ship that subsequently rescued the passengers and crew.

1.10 Aerodrome information

Not applicable to this investigation.

1.11 Flight Recorders

Both G-REDW and G-CHCN were equipped with a Honeywell AR-Series combined CVFDR[19], which records 25 hours of data and 2 hours of audio. The audio recordings include the commander and co-pilot's communications, radio transmissions, passenger announcements and audio from the Cockpit Area Microphone (CAM).

1.11.1 G-REDW CVFDR

The CVFDR data indicated that G-REDW lifted off from Aberdeen at 1047:34 hrs and was just under 18 minutes into the flight (34 nm east of Aberdeen Airport) before the first indication of the loss of the MGB oil pressure. Figure 18 is a plot of the salient flight data parameters, starting 1 minute 13 seconds before the loss of the MGB oil pressure. The following pertinent information was obtained from the data:

UTC	Event	Time from MGB low oil pressure warning
11:05:12	MGB oil pressure begins to reduce.	
11:05:13	MGB oil pressure low warning.	00:00
11:05:14	CWP main warning.	+00:01
11:05:15	MGB sump chip detected.	+00:02
11:05:54	"PAN PAN PAN PAN PAN PAN" radio call made.	+00:41
11:06:31	Helicopter descends from 3,000 ft amsl.	+01:18
11:07:03	Emergency lubrication system activated. (From CVFDR co-pilot states that the EMLUB activation light is "ILLUMINATED").	+01:50
11:07:37	CWP main warning corresponding to emergency lubrication system failure detection.	+02:24
11:08:24	Landing gear down selected.	+03:11
11:09:01	Peak MGB oil temperature of 127.5°C recorded.	+03:48
11:10:41	Epicyclical chip detected.	+05:28
11:12:44	Helicopter levelled off at 200 ft amsl.	+07:31
11:13:35	"MAYDAY" radio call made.	+08:22
11:14:08	Helicopter ditched.	+08:55
11:14:32	CVFDR stopped recording.	+09:19

[19] Honeywell AR-Combi CVFDR Part Number 980-6021-066.

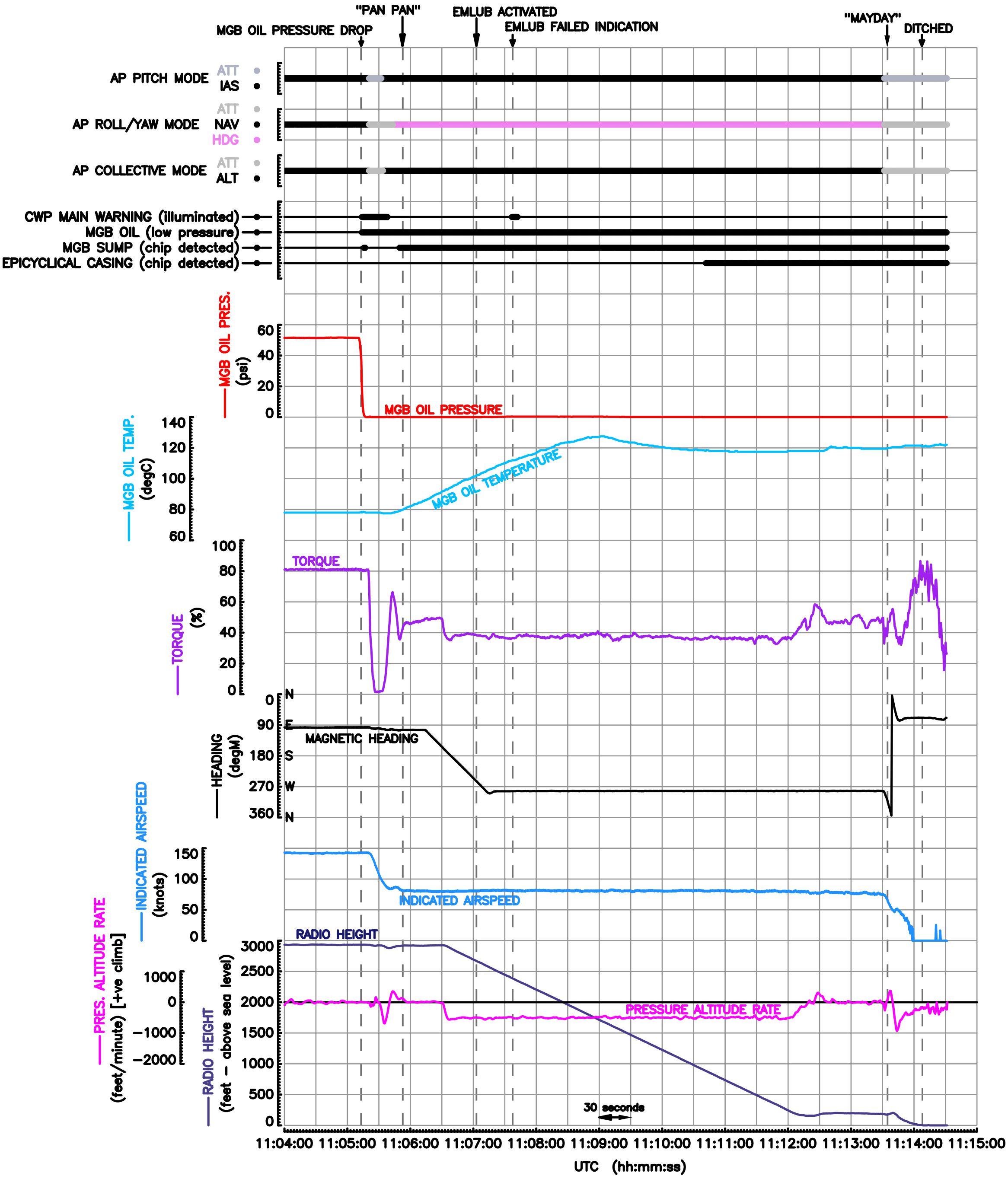


Figure 18

G-REDW flight data extract

1.11.2 G-CHCN CVFDR

The CVFDR data indicated that G-CHCN lifted off from Aberdeen at 1322:57 hrs and was just under 59 minutes into the flight (138 nm north of Aberdeen Airport) before the first indication of the loss of the MGB oil pressure. Figure 19 is a plot of the salient flight data parameters, starting 1 minute 57 seconds before the loss of the MGB oil pressure. The following pertinent information was obtained from the data:

UTC	Event	Time from MGB low oil pressure warning
14:21:27	MGB oil pressure begins to reduce.	
14:21:28	MGB oil pressure low warning.	00:00
14:21:29	CWP main warning.	+00:01
14:21:30	MGB sump chip detected.	+00:02
14:21:42	Helicopter descends from 3,000 ft amsl.	+00:14
14:21:57	Emergency lubrication system activated.	+00:29
14:22:31	CWP main warning corresponding to emergency lubrication system failure detection.	+01:03
14:22:38	“MAYDAY” radio call made.	+01:10
14:23:38	Landing gear down selected.	+02:10
14:24:42	Peak MGB oil temperature of 123.5°C recorded.	+03:14
14:24:57	Epicyclical chip detected.	+03:29
14:25:26	Helicopter (initially) levelled off at 25 ft amsl into the hover.	+03:58
14:28:34	Helicopter ditched.	+07:06
14:28:56	CVFDR stopped recording.	+07:28

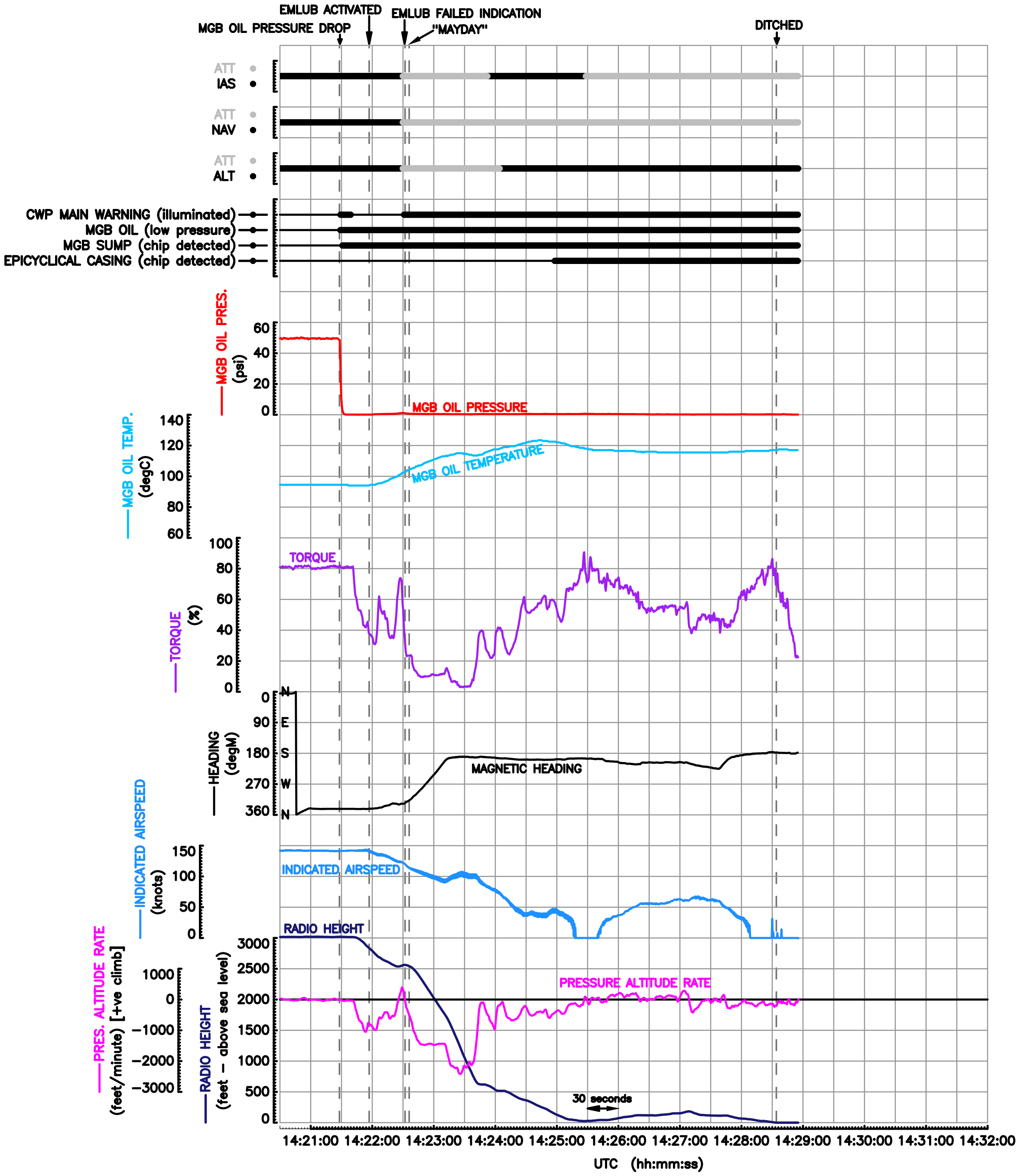


Figure 19

G-CHCN flight data extract

1.11.3 Operating histories

Both operators used a Helicopter Operations Monitoring Programme (HOMP), which is a helicopter version of the fixed wing Flight Data Monitoring programme. The data (a subset of the CVFDR flight data) for this programme was recorded onto a PCMCIA[20] memory card situated in the control panel below the HUMS Helicopter Monitoring Interface control unit. The same card also recorded the HUMS data. A review of this historical flight data enabled an analysis of each helicopter's operating history to be made.

1.11.3.1 G-REDW operating history

G-REDW flew on 34 of the 54 days between the MGB having been installed and the accident occurring. Throughout this period it was based onshore and was subject to the following:

- 86 engine start and stops,
- 63 flights and 155 sectors,
- 219 engine hours and 156 flight hours.

A breakdown by flight time of the 155 sectors flown by G-REDW from Aberdeen is shown at Figure 20:

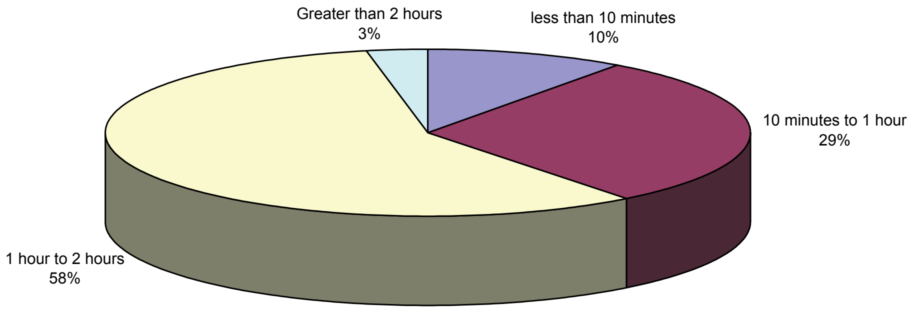


Figure 20

Breakdown by flight time of the 155 sectors flown by G-REDW out of Aberdeen

An analysis of the recorded data for these sectors revealed that during this period the helicopter operated for:

- 0.3% of the time at a power setting above MCP[21],
- 77.4% of the time at an engine power setting between MCP and 80% torque,
- 22.3% of the time at an engine power setting below 80% torque.

[20] Personal Computer Memory Card International Association.
[21] Maximum Continuous Power, Appendix D describes the engine power settings.

1.11.3.2 G-CHCN operating history

G-CHCN flew on 114 of the 130 days[22] prior to the accident. During this period it was based offshore and operated from an oil platform for 50 days during the following dates: 17 June 2012 to 22 July 2012 (36 days)[23], 26 July 2012 to 5 August 2012 (11 days), and 25 to 28 September 2012 (4 days). During the 114 days the helicopter was subject to the following:

- 210 engine start and stops,
- 210 flights and 580 sectors,
- 567 engine hours and 329 flight hours.

While based offshore the helicopter was subject to:

- 119 engine start and stops,
- 355 sectors,
- 220 engine hours and 67 flight hours.

A breakdown of the 355 sectors flown offshore by G-CHCN is shown at Figure 21. The 4% of the sectors that last longer than one hour were the transit flights to and from Aberdeen.

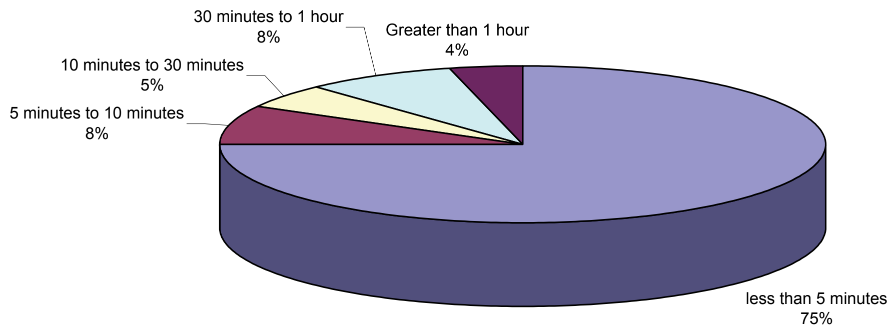


Figure 21

Breakdown by flight time of the 355 sectors flown by G-CHCN while based offshore

While operating from Aberdeen the percentage of time that G-CHCN spent at the different power settings was similar to G-REDW. However, whilst based offshore a typical power spectrum for a 5-minute flight was:

- 2.7% of the time at a power setting above MCP,
- 2.3% of the time at an engine power setting between MCP and 80% torque,
- 95% of the time at an engine power setting below 80% torque.

22 This period was chosen in order to help with the work carried out in understanding the initiation and growth of the crack.

23 This is atypical of the time the operator would normally base a helicopter offshore: a two-week rotation of helicopters was normal.

1.11.4 Health and Usage Monitoring System (HUMS)

1.11.4.1 Vibration Health Monitoring (VHM) Regulatory requirements

On 1 June 1999, the CAA issued Additional Airworthiness Directive (AAD) 001-05-99 that made the installation and use of Vibration Health Monitoring (VHM) mandatory for UK registered helicopters issued with a Certificate of Airworthiness in the transport category and having a maximum approved seating configuration of more than nine. The acceptable means of compliance with the AAD was originally specified in Civil Aviation Publication (CAP) 693, which was superseded in September 2010 by CAP 753, 'Helicopter Vibration Health Monitoring'. This document provides guidance on both the design and operation of VHM systems.

Following their formation in 2003, the EASA reviewed the requirements for VHM. They concluded that the National Aviation Authorities should, where necessary, introduce national VHM requirements for 'demanding' operations, such as those operations in the North Sea.

For the EC225 LP these requirements are met by the use of HUMS.

1.11.4.2 HUMS Alert threshold philosophy

HUMS is intended to detect wear and degradation of rotating systems with a low propagation rate, and the activity is undertaken in addition to the schedule maintenance programme.

HUMS requires the Design Organization to set threshold values for each indicator above which an alert is generated. While the thresholds need to be set above the ambient noise levels, if they are set too low then the rate of false alarms can result in an unacceptable maintenance burden. It is normal for the manufacturer to revise the threshold levels, and introduce new alerts, as a result of knowledge gained from statistical analysis of vibration levels across the fleet.

The guidance to operators given in CAP 753 states that the period between the successful download and assessment of any primary VHM indicator, used for monitoring the engine and rotor drive system components, should not exceed 25 hours. This interval is reduced to 10 hours for components or indicators that require 'close monitoring' where, for example, an indicator value has exceeded a 'maintenance action' threshold or shows signs which warrant increased attention.

1.11.4.3 HUMS configuration on EC225 LP

On the EC225 LP the HUMS forms part of the M'ARMS and uses accelerometers to capture the vibration of rotating components; eight accelerometers are fitted to the MGB.

The system processes the raw signal from the accelerometers to produce the condition indicators, which are then used to monitor the vibration levels of individual components. The acquisition cycle for one complete set of samples typically lasts between 30 and 40 minutes, although some accelerometers are sampled more frequently.

At the end of each flight, as the helicopter is shutdown[24], the system downloads the HUMS data onto a PCMCIA card. The PCMCIA card can only store HUMS data for a maximum of five complete acquisitions.

The number of acquisitions will be correspondingly less on flights where insufficient time is available to capture five complete acquisitions, or where insufficient time is spent in certain flight phases particular to certain condition indicators, or if an acquisition is rejected.

On flights where more than five complete acquisitions are made, the system chooses five to download that are distributed evenly throughout the flight rather than use the five most recent ones. However, the first and last acquisitions are always kept.

The HUMS data is transferred from the PCMCIA card to the system's Ground Station Computer (GSC). On the GSC the condition indicators are calculated and reviewed by engineering personnel to identify, for example, any indicators that may have exceeded their thresholds.

1.11.4.3.1 EC225 LP alert thresholds

Thresholds are critical values for condition indicators which are set to alert the user of significant changes in their values. Two types of alert exist:

- Amber alerts give an advance warning of a potential problem. This prompts the close monitoring of the indicator and maintenance inspections.
- Red alerts indicate that a more serious problem has potentially been found and maintenance action is required before the helicopter is released for flight.

[24] The session starts once N1 > 5% on both engines and stops on engine shutdown when N1 < 5% and NR < 85%.

Alerts are normally generated when two out of five consecutive indicator values exceed their respective threshold. Two types of thresholds exist:

- Learned thresholds are a function of the mean and standard deviation of the indicator values recorded to date. They are particular to an individual helicopter and based typically on the last 25 flight hours.

- Fixed or Maximum thresholds are defined by the helicopter manufacturer Design Office. These are fleet-wide and can, if required, be set lower than the learned thresholds.

1.11.4.4 G-REDW operator's internal HUMS procedures

At the time of the accident, the procedure was for the PCMCIA card to be removed and downloaded when the helicopter returned to the Main Operating Base (MOB) and the engines were shut down. The operator's procedure also required the period between downloads not to exceed 10 flight hours. If the helicopter was on 'HUMS close monitor', the crew would initiate the HUMS data transfer to the PCMCIA card at every return to MOB, even if the engines were not shut down.

It was the responsibility of the licensed aircraft engineer signing for the flight servicing to download the data onto the GSC and to analyse the indicator data to establish if any alerts had been generated. If alerts were found, then the engineer would raise an entry in the technical log and follow the advice in the AMM.

The operator's HUMS engineer would review all the indicators on each helicopter in the fleet once every fortnight. This exercise was intended to confirm that the system was serviceable and to identify any underlining trends, or significant deviations, which might not have generated an alert.

Following the accident to G-REDW, the procedures were changed such that the HUMS data was required to be downloaded to the PCMCIA card and analysed every time the helicopter returned to the MOB. Additional checks were also introduced, after every flight, for the indicators relating to the bevel shaft and wheel to check for any anomalies and/or rising trends. If any alerts or anomalies were identified, further flights were suspended until a positive maintenance action had been carried out. If a Eurocopter Diagnostic Report (EDR)[25] had been raised, regardless of the guidance in the AMM, the operator's procedure would be to ground the helicopter until the problem had been discussed with the helicopter manufacturer.

25 An EDR is a report that is transmitted to the helicopter manufacturer to provide information arising from the analysis of the HUMS data.

Due to the increased number of helicopters in the operator's fleet, they were no longer able to carry out the fortnightly fleet check of all indicators on their helicopters. However, these checks were replaced with monthly spot checks.

1.11.4.5 G-REDW HUMS download

On 26 April 2012, the operator's HUMS engineer carried out the fortnightly review of the HUMS data on G-REDW. He assessed that the system was serviceable and there were no rising trends or unusual deviations on any of the indicators.

On the last flight on 9 May 2012, the engineer who downloaded the HUMS data noted that an amber alert had been generated for MOD-45 (condition indicator relating to the bevel gear meshing). An entry was raised in the technical log and the engineer, as guided by the GSC, followed the flow chart in AMM 45.11.08.811.008, which required the accelerometer system to be checked in accordance with AMM 45.11.08.211. The engineer concluded that the condition of the fibre washer on the accelerometer might have resulted in an erroneous vibration signal. The washer was replaced and, in accordance with the flow chart, the helicopter was released for flight. The HUMS engineer also reviewed the data and noted that only the last two data points had exceeded the threshold. This was not considered to be sufficient to determine if there was a rising trend. There was no requirement in the AMM for the helicopter to be placed on 'close monitoring'.

When G-REDW returned from its first flight of the day on 10 May 2012, the HUMS was downloaded and examined, were it was found that amber alerts had been generated for MOD-45 and MOD-70 (condition indicator relating to oil pump drives). The engineers sought advice from the HUMS engineer who recommended that they continue to follow the flow chart in the AMM. As the accelerometer system had been examined the night before, the engineers checked the sump magnetic chip detector which was found to be clear. The HUMS engineer checked all the condition indicators for the bevel shaft (13 indicators), the oil pump gears (10 indicators) and the combiner gear (13 indicators). He noticed that during the last flight there had been a slight rising trend on indicators K_g (tooth damage and general wear), RMS-r (general wear and misalignment) and K_m (localized damage to gear teeth). However, none of these indicators exceeded their thresholds. All other indicators were normal.

In line with the flow chart, the helicopter was released on a 10-hour close monitoring and an EDR, covering the amber alerts that had occurred on the last two flights, was sent to the helicopter manufacturer's HUMS support team. The operator's HUMS engineer also attempted to speak to the relevant personnel within the HUMS support team, but the accident occurred before he was able to do so.

The helicopter manufacturer subsequently advised the investigation that, based on the information available at the time, they would have advised the operator to complete the first 10-hour cycle of 'close monitoring', after which they would review the data and determine if it was necessary to either carry out a boroscope inspection or replace the gearbox.

The PCMCIA card was removed from the helicopter by the crew before they exited it after the ditching. The loss of electrical power to the helicopter following the ditching meant that the HUMS data for the accident flight was not written to the PCMCIA card.

1.11.4.6 G-CHCN Operator's internal HUMS procedures

Prior to the accident to G-REDW, the operator of G-CHCN carried out a daily check of the HUMS data at the MOB. It was then the responsibility of the licensed aircraft engineer, signing for the flight, to transfer the data onto the GSC and analyse the data to establish if any alerts had been generated. When operating offshore, a laptop would be used as a mobile GSC. If any alerts had been generated, the engineer would follow the advice in the AMM and raise an entry in the technical log. The operator's Continuing Airworthiness Department would also be informed using their HUMS Technical Support Request, which could lead to an EDR being sent to the helicopter manufacturer.

Following the accident to G-REDW, the Operator placed all their helicopters on 'close monitoring' which required a HUMS download and check after each return to the MOB. Guidance was issued by the helicopter manufacturer in a SIN[26] dated 6 July 2012, that restricted the 'close monitoring' for shafts that fell within a range of serial numbers. However, the shaft fitted to G-CHCN fell outside this range and, consequently, the HUMS download reverted back to a daily basis.

1.11.4.7 G-CHCN - HUMS Download

On the morning of 21 October 2012, the helicopter flew a short test flight. Following this flight, the PCMCIA card containing the HUMS data was downloaded onto the GSC. This was the last time that the HUMS data was downloaded until after the accident. The helicopter then flew a further flight consisting of two sectors in the afternoon.

The following day, which was the day of the accident, the helicopter flew a flight consisting of two sectors in the morning. This flight started and ended at Aberdeen.

26 SIN 2470-S-00, issued 6/7/2012. Progress of investigations following ditching of an EC225 helicopter in the North Sea in May 2012.

The next flight was the accident flight and after the ditching, the crew removed the PCMCIA card from the helicopter. The loss of electrical power to the helicopter following the ditching meant that the HUMS data for the flight was not written to the PCMCIA card. However, the HUMS data from the previous two flights, which had not been downloaded onto the GSC, was recovered from the card.

It was subsequently established, from the HUMS data, that no alerts would have been generated for the flight on 21 October 2012 following the last HUMS download.

For the first sector flown on the morning of 22 October 2012 (the day of the accident), the HUMS data acquisition generated two data points each for the MOD-45 and MOD-70 indicators. No alerts would have been generated for the MOD-70 indicator; however, the MOD-45 indicator exceeded the amber threshold with the first point and the red threshold with the second point.

On the next sector, three more data points were generated. The MOD-45 indicator values had increased in magnitude and would have exceeded the red threshold. Only the last two MOD-70 indicator values had increased in magnitude, of which the last would have exceeded the red threshold.

1.11.4.8 History of MOD-45 and MOD-70 indicators

Figures 22 and 23 compare the MOD-45 and MOD-70 indicators for G-CHCN and G-REDW. The indicator values are plotted with respect to flying hours relative to the time at which the MGB oil pressure was lost; the period covered by each figure is 30 flying hours. Also plotted are the threshold values of these indicators unique to each helicopter and applicable at the time of each accident.

At the time of the first accident in May 2012, the MOD-45 and MOD-70 indicators only had amber thresholds. These were learned thresholds, which for G-REDW were 0.19 for MOD-45 and 0.14 for MOD-70. For G-CHCN they were 0.10 for MOD-45 and 0.14 for MOD-70. The manufacturer defined fleet-wide maximum amber alert was set at 0.6 for both indicators.

In July 2012, Eurocopter published EC225 Service Bulletin No 45-001. This introduced a learned and maximum red threshold for both of these indicators, and lowered the fleet-wide maximum amber threshold values for both indicators. For MOD-45 the amber alert was reduced to 0.3 and a red alert of 0.4 was introduced. For MOD-70 the amber alert was reduced to 0.4 and a red alert of 0.5 was introduced. Note that the fleet-wide maximum for the amber alert was still higher than the learned thresholds for both helicopters.

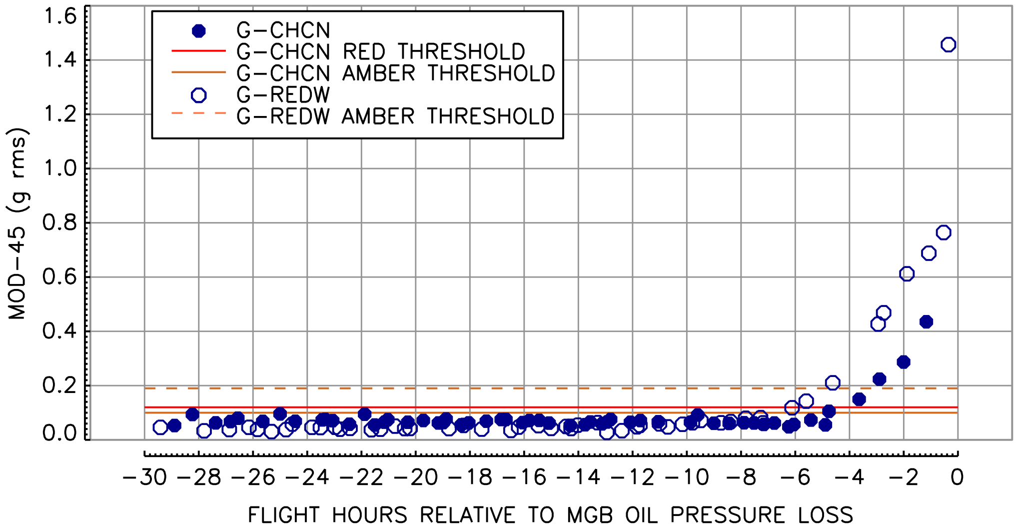


Figure 22

Comparison MOD-45 condition indicator between G-CHCN and G-REDW

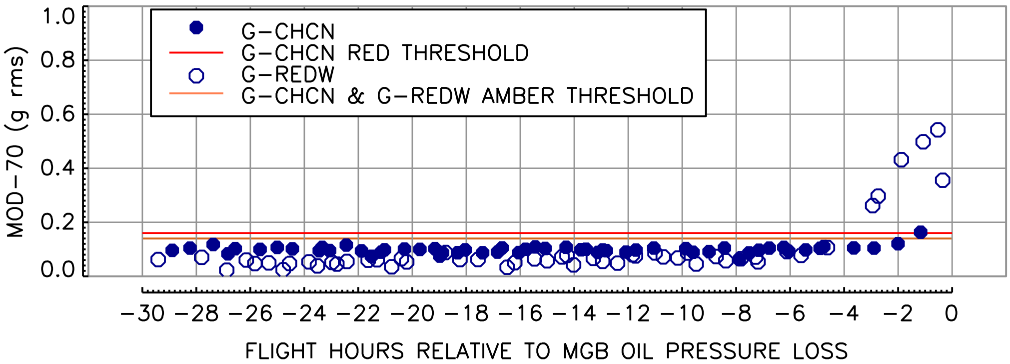


Figure 23

Comparison MOD-70 condition indicator between G-CHCN and G-REDW

On 21 November 2012, Eurocopter published an Emergency Alert Service Bulletin (ASB). This removed the maximum amber alert threshold for MOD-45 and lowered the fleet-wide maximum red alert threshold to 0.2. No change was made to MOD-70 indicator thresholds.

For G-REDW the MOD-45 indicator exceeded its learned amber threshold (0.19) 4.62 flying hours before the loss of the MGB oil pressure. No red threshold existed at the time of the accident involving G-REDW. The MOD-45 indicator for G-CHCN exceeded its learned amber threshold (0.10) 4.75 flying hours and its learned red threshold (0.12) 3.63 flying hours prior to the loss of oil pressure (Figure 22).

For the MOD-70 indicator, the first instance that it exceeded the learned amber threshold (0.14) for G-REDW was 2.95 flying hours before the loss of MGB oil pressure (Figure 23). However, for G-CHCN only the last recorded value of this indicator, which was captured 1.17 flying hours before the loss of the MGB oil pressure, exceeded both its amber (0.14) and red (0.16) learned thresholds.

1.12 Wreckage and impact information

1.12.1 General

A detailed examination of both helicopters was carried out at each of the operator's maintenance facilities in Aberdeen. Components were then taken to various manufacturer facilities for further examination and testing.

1.12.2 Initial examination of G-REDW

G-REDW remained upright and was salvaged onto a recovery vessel approximately 22 hours after it ditched (Figure 1). Whilst afloat the sea state increased to a Force 5 and waves were seen to break over the windshields, the tail boom dipped into the water and the main rotor blades caught the top of the waves.

There was no structural damage on the lower surfaces of the helicopter to indicate that it had landed heavily in the water. Some aerials had been damaged and minor damage had occurred to the structure and the left pilot's lower transparency during the recovery operation.

The cockpit and cabin doors had been jettisoned; all the cabin windows were still in place. The flotation bags were intact and fully inflated, although the forward cell in the left main bag 'felt soft'.

There was a strong smell of hot oil around the helicopter. A mixture of oil and water covered the transmission decking and had been sprayed around the inside of the sliding cowling. The oil level in the MGB was at the bottom of both sight glasses.

1.12.3 Initial examination of G-CHCN

G-CHCN remained upright and was salvaged onto a recovery vessel approximately 25 hours after it ditched (Figure 2). Whilst afloat the sea state did not change significantly. The video taken during the recovery showed that the water level was just above the height of the passenger seat base, and all four flotation bags appeared to be fully inflated.

There was no structural damage on the helicopter's lower surface to indicate that it had landed heavily on the water. The left cockpit door was missing whilst the right cockpit door was still in place but with its jettison pins pulled. The passenger doors were open and had not been jettisoned. The right lower forward cockpit window was broken and the left upper cockpit window was cracked; this damage had occurred prior to the recovery of the helicopter.

As with G-REDW, a mixture of oil and water covered the transmission decking and had been sprayed around the inside of the sliding cowling. A test of this fluid revealed the presence of glycol, which is a constituent of Hydrosafe 620. The oil level in the MGB was at the bottom of both sight glasses.

1.12.4 Condition of the MGB on G-REDW and G-CHCN

Examination of the MGBs fitted to G-REDW and G-CHCN identified no visual evidence of heat distress or significant damage to any other components in the MGB other than the bevel gear vertical shaft. A small amount of wear debris was found in the epicyclic module that the manufacturer stated was normal for a gearbox that had been in service.

The MGB sump magnetic chip detectors from both helicopters had captured material from the failed shafts.

The epicyclic magnetic chip detectors were removed from each of the MGBs on both helicopters after they had been recovered to Aberdeen and before the gearboxes were removed and stripped. No metallic particles, large enough to bridge the gap, were found on the epicyclic magnetic chip detectors or in the recesses in which they were located.

Small quantities of very fine metallic particles were found on the end of the epicyclic chip detectors and in the fluid remaining in the chip detector recesses. Inspection of the gearboxes could find no evidence of damage to any of the components in the epicyclic module. Small quantities of very fine metallic dust were found on the bottom of the epicyclic modules. Although a considerable amount of metallic debris was found in the MGB sumps, the oil filters were relatively clean which indicated that the debris in the sump had been generated after the drive to the main and standby oil pumps had been lost. Therefore, the debris generated following the failure of the shaft could not have been transmitted by the oil distribution system to the epicyclic modules.

The components in both MGBs had been correctly manufactured and assembled and the dimensions were within the design tolerances. The oil pumps in both gearboxes turned freely by hand. Glycol was found throughout the inside of the gearbox casings and on all the gears and bearings.

1.12.5 Bevel gear vertical shafts

1.12.5.1 Condition of the bevel gear vertical shaft fitted to G-REDW

The bevel gear vertical shaft on G-REDW failed as a result of a 360° circumferential crack that initiated at the inner countersink of the 4.2 mm hole in the weld. A representation of the location of the crack is at Figure 24.

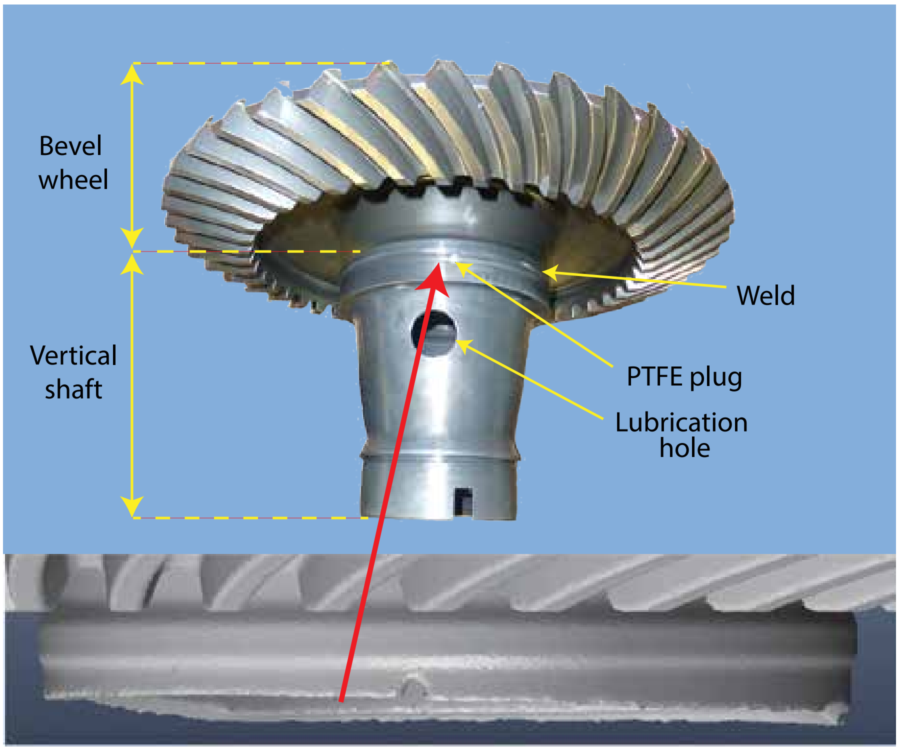


Figure 24

Graphic showing the location of the circumferential crack on G-REDW

This fracture allowed the vertical shaft to drop downwards by approximately 6 mm until its movement was arrested by the outer race of the lower roller bearing making contact with the retaining lip that had been machined into the vertical shaft. Smear marks on the fracture surfaces indicate that there had been some rotating contact between the two parts of the shaft after failure had occurred. The teeth on the main oil pump drive had sustained some damage that occurred after the shaft had failed; there was similar damage on the standby oil pump drive. The PTFE plug was still fitted in the 4.2 mm hole. Looking down the shaft, the 29 mm lubrication hole was positioned 38° clockwise from the 4.2 mm hole.

The bearing retainer for the lower roller bearing, which also forms the bearing's outer race, had fractured as a result of the bearing having been forced downwards. Light asymmetrical marks were found on the bearing cages fitted to the upper roller and ball bearings that are believed to have occurred in the time between the shaft failing and the gearbox ceasing to rotate. A large number of metallic, and some non-metallic, debris was found in the gearbox sump; no other debris was found anywhere else in the gearbox. The metallic debris was identified as coming from the bevel gear vertical shaft and the pinion and gears that drive the oil pumps. There was some fretting damage on the splines of the first stage of the sun gear; there was no evidence of wear anywhere else in the gearbox.

From an examination of the weld using a Scanning Electron Microscope (SEM), a review of the x-rays and results of the inspections performed during manufacturing, it was established that the weld had been correctly formed. The weld was within the design tolerances. While the 4.2 mm hole was of the correct size, and correctly aligned in the weld, there was evidence of tooling marks and a spiral scratch that ran along the length of the bore. The geometry of the countersinks was found to be outside the design specifications; there were also a number of 'scoops' in the inner countersink (Figure 25).

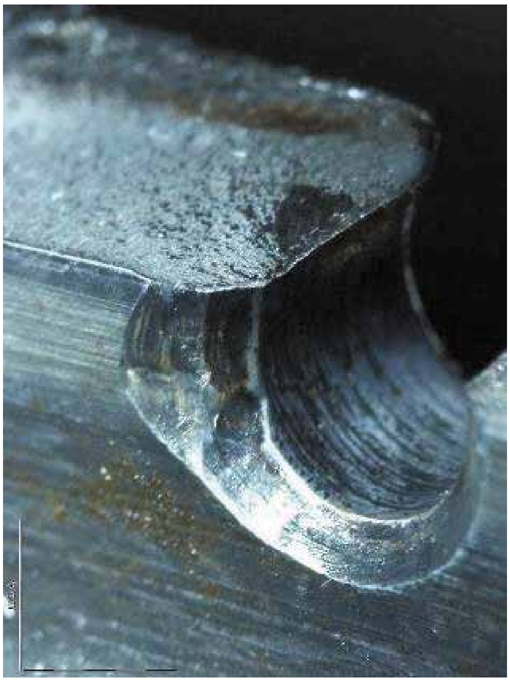


Figure 25

Condition of the 4 mm hole and countersink on G-REDW

There were patches of very small corrosion pits in the walls of the hole and around the inner countersink, in the area where there is a gap (crevice) between the PTFE plug and the countersink. These corrosion pits were only initially detected using a SEM (Figure 26).

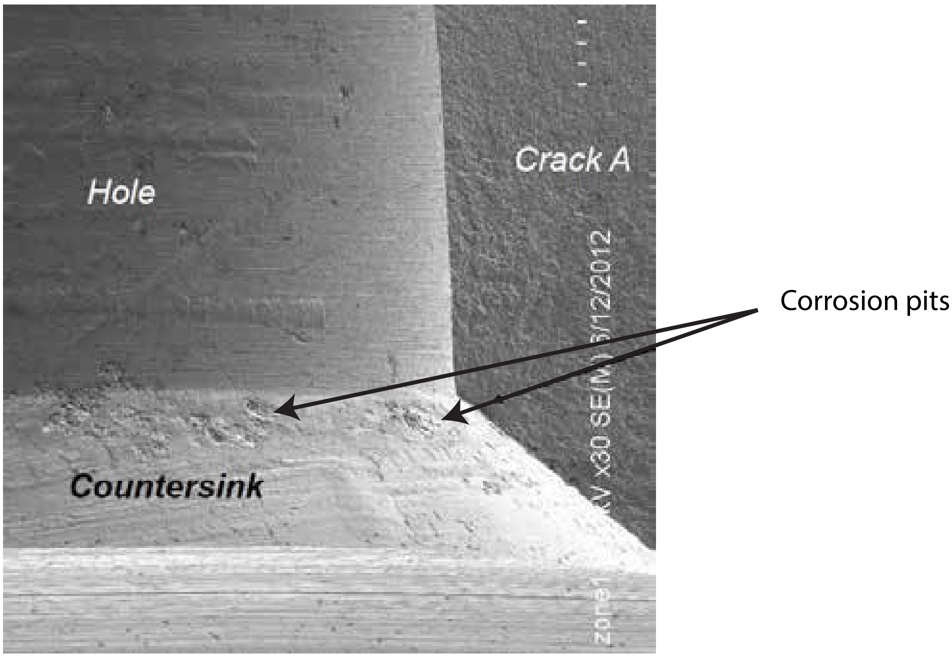


Figure 26

Corrosion pits on inner countersink on G-REDW

The average surface roughness (R_a) of the part of the hole, on the lower section of the shaft, was measured as 1.695 μm[27]. However, one end of the hole was much rougher than the other, with the R_a being 2.50 μm and 0.29 μm, respectively. The deepest feature was of the order of 60 to 70 μm. The design drawings specify a R_a of 3.2 μm for the shaft and 1.6 μm for the bore of the 4.2 mm hole.

Following the accident to G-REDW, the manufacturer examined eighteen EC225 shafts with serial numbers between M308 and M559. There was some variability in the geometry of the countersinks on the 4.2 mm holes and a number were found to be outside the design tolerance of 90° ± 1°. There was also evidence of tooling marks in the bore of a number of these holes.

1.12.5.2 Condition of the bevel gear vertical shaft fitted to G-CHCN

The bevel gear vertical shaft fitted to G-CHCN failed as a result of a 360º circumferential crack that initiated on the inner radius and passed through part of the 4.2 mm hole in the weld. While the PTFE plug was still fitted in the 4.2 mm hole, a small slither of PTFE was found trapped between the plug and countersink which had left a small gap between these parts. As with G-REDW, the shaft had moved downwards causing the loss of the drive to both oil pumps. However, unlike G-REDW the teeth on the oil pump drives were undamaged and the bearing retainer for the lower bearing remained intact.

[27] Measured using a Talysurf profile meter with an ISO-2CR filter

Looking down the shaft, the 4 mm hole was positioned 28º clockwise from the 29 mm lubrication hole and the cracks started at a position 45º clockwise from the 4.2 mm hole.

There were signs of wear on the splines that drive the first stage sun gear. There was also evidence of the rollers on the upper roller bearing having slipped along the outer race and there were light marks, similar to those seen on G-REDW, in the cage on the roller bearing. The lower roller bearing displayed no unusual marks. A large number of metallic, and some non-metallic, debris was found in the gearbox sump; no other debris was found elsewhere in the gearbox.

A red deposit was present on the inside of the bevel gear[28] part of the shaft which appeared to be slightly more concentrated on the inner radius and above and below the splines (Figure 27). Apart from on the inside of the first stage sun gear and in the fluid found in the gearbox sump, this deposit was not seen on any other part of the bevel gear vertical shaft or in the MGB. The deposit was found to contain iron oxide that had most probably been generated as a result of wear debris from the splines that drive the first stage sun gear.

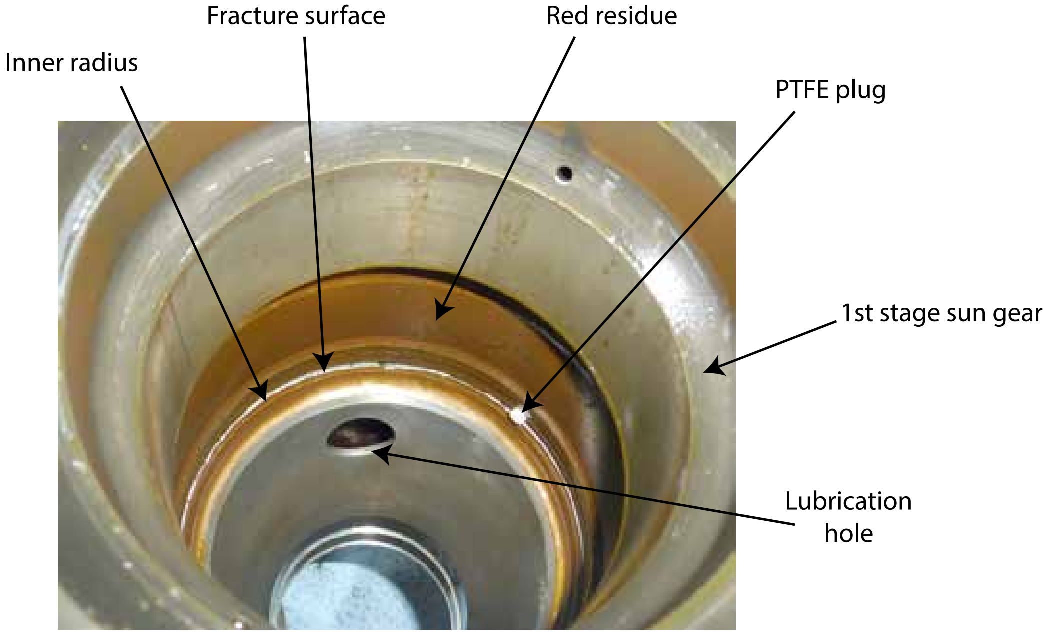


Figure 27

Red deposit on inner surface of the shaft fitted to G-CHCN

When the deposit was removed, corrosion was observed visually on the shaft, in the areas above and below the splines. Small areas of corrosion pits and machining marks were also present around the inner flange that had been machined to remove the root of the weld (Figures 28 and 29).

28 Figure 10

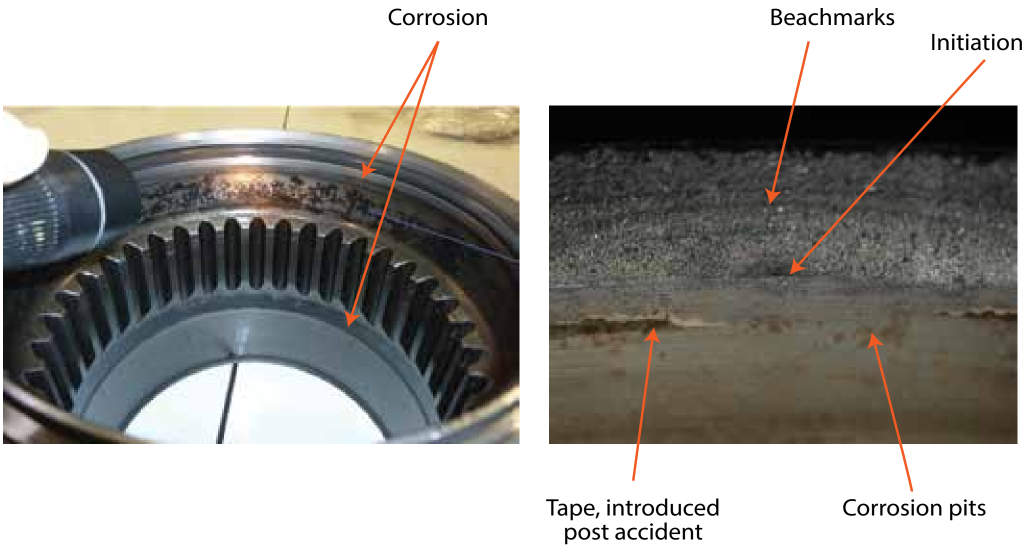


Figure 28

Evidence of corrosion pits on G-CHCN

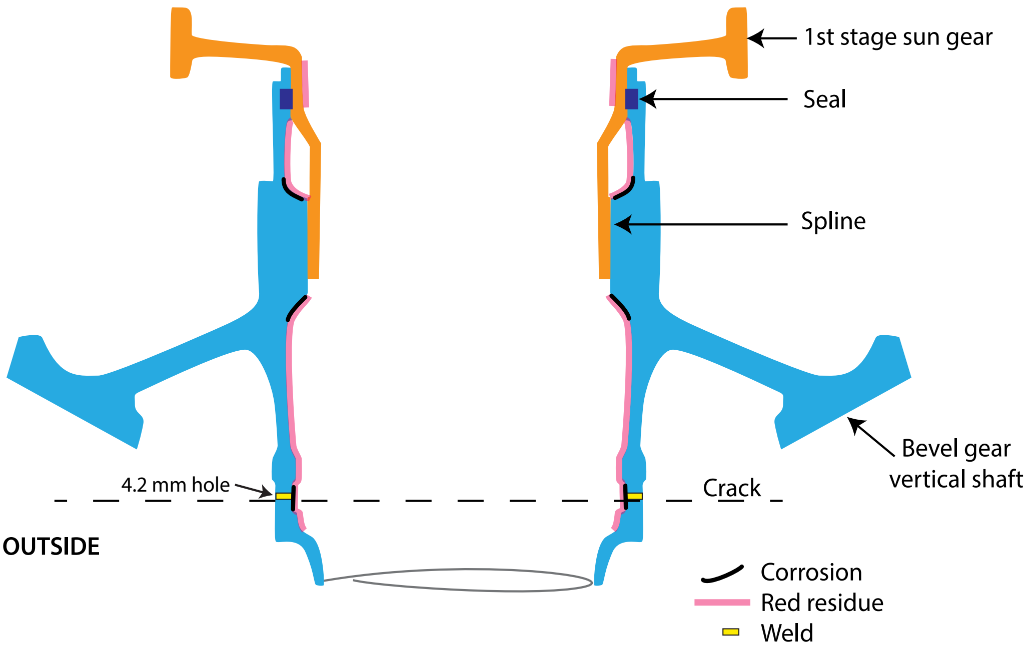


Figure 29

Location of corrosion on bevel gear vertical shaft G-CHCN

Corrosion similar to that seen on G-REDW was also found on the inner countersink of the 4.2 mm hole in the weld (Figure 30). There was no evidence of corrosion elsewhere on the shaft.

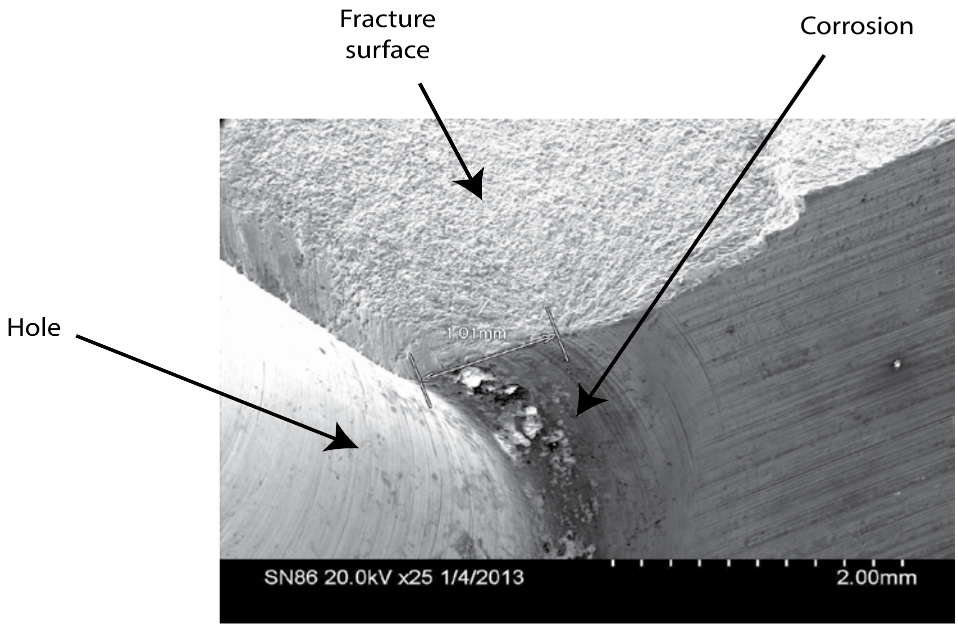


Figure 30

Corrosion pits on inner countersink on G-CHCN

Examination of the contents of the MGB oil filter from G-CHCN was carried out after it had been agitated in an ultrasonic bath and the contents filtered onto an 11µm Millipore. The amount of debris was considered to be normal with: 65% of the debris consisting of inorganic/organic material including carbon particles, fibres and siliceous particulate; 25% of the material appeared to have been generated from the bearing tracks, balls and rollers; 8% was predominately silver-based and the remaining 2% was from aluminium-based materials. No significant quantity of iron oxide, as identified in the red deposit, was found in the filter.

1.12.5.3 Examination of fracture surfaces

The failure of both shafts was consistent with the shafts bending (flexing) as they rotated, with approximately 99% of the fracture surface having failed in fatigue and 1% in overload. There was no evidence of corrosion on any of the fracture surfaces.

1.12.5.3.1 Fracture surface on G-REDW

Examination of the fracture surface on G-REDW revealed the presence of three cracks identified as ‘A’, ‘B’ and ‘C’. Cracks ‘A’ and ‘B’ both exhibited visual evidence of beachmarks that are associated with fatigue crack propagation. The first crack to develop was identified as Crack ‘A’, which initiated at a corrosion pit approximately 60µm deep located in the inner countersink of the 4.2 mm hole on the fusion line of the weld (Figure 31). Crack ‘A’ then propagated close to the fusion line, on the inside of the shaft, up to a minimum of 61 mm and was observed to be in the parent material at a distance of 135 mm from the point of initiation. A second crack, Crack 'B', then initiated at a small defect in the internal surface of the hole. The propagation of Crack 'A' may have caused the initiation of Crack 'B' and the deviation of Crack 'A' into the HAZ may have been as a result of the propagation of Crack 'B'. Crack 'B' ran into a third crack identified as Crack 'C' (Figures 31 and 32).

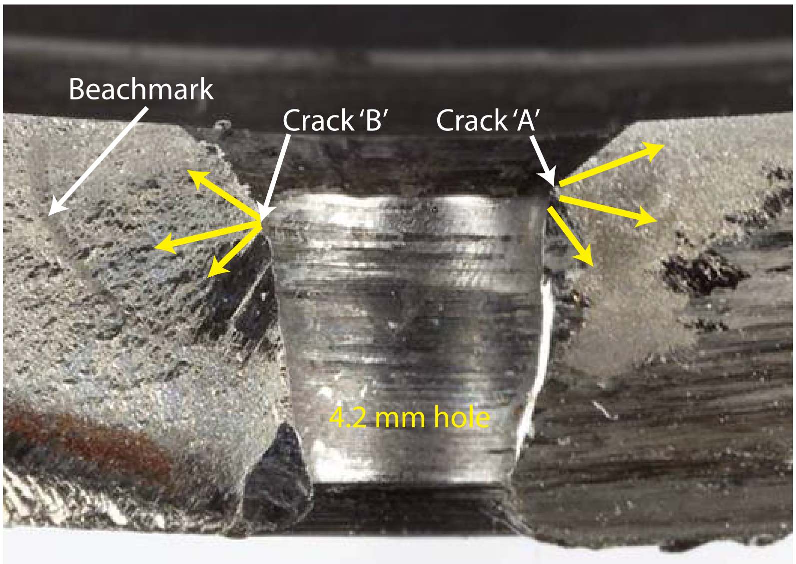


Figure 31

Location of cracks 'A' and 'B' on G-REDW

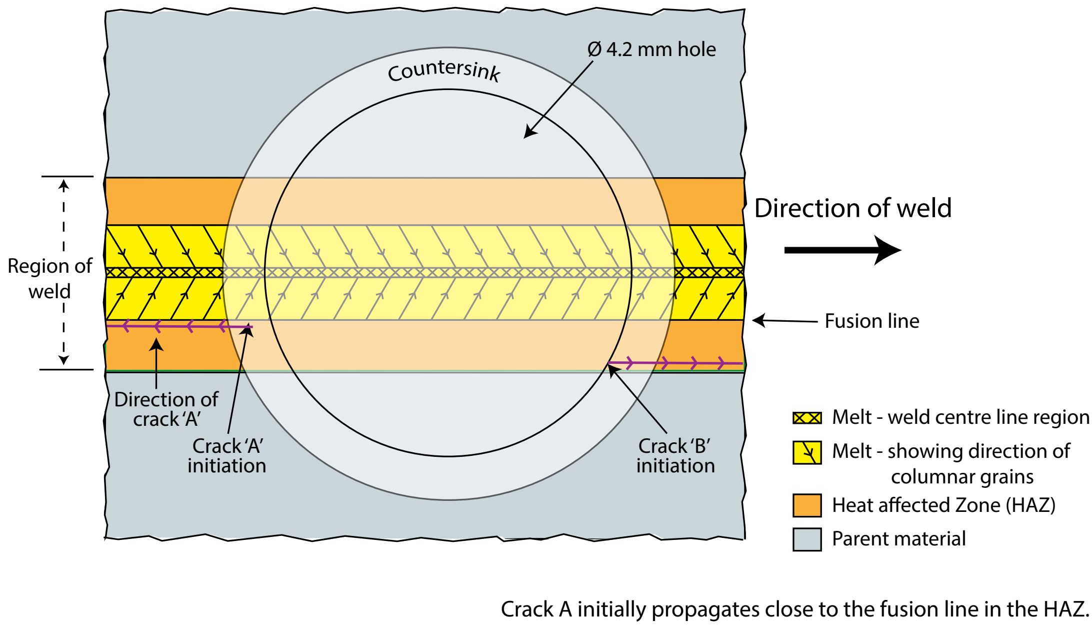


Figure 32

Location of crack initiation and propagation on G-REDW

Crack 'A' extended around the circumference for approximately 250º, Crack 'B' extended in the opposite direction to crack 'A' along an arc of approximately 80º (Figure 33). Crack 'C', which showed no evidence of beachmarks, ran between Crack 'A' and 'B' along an arc of approximately 30º. The inner part of the fracture surface was flat and perpendicular to the axis and what appeared to be a 45º shear lip ran around the outer edge. Striations[29] were also detected, by the use of a SEM, on the fracture surfaces of Crack 'A' and 'B'.

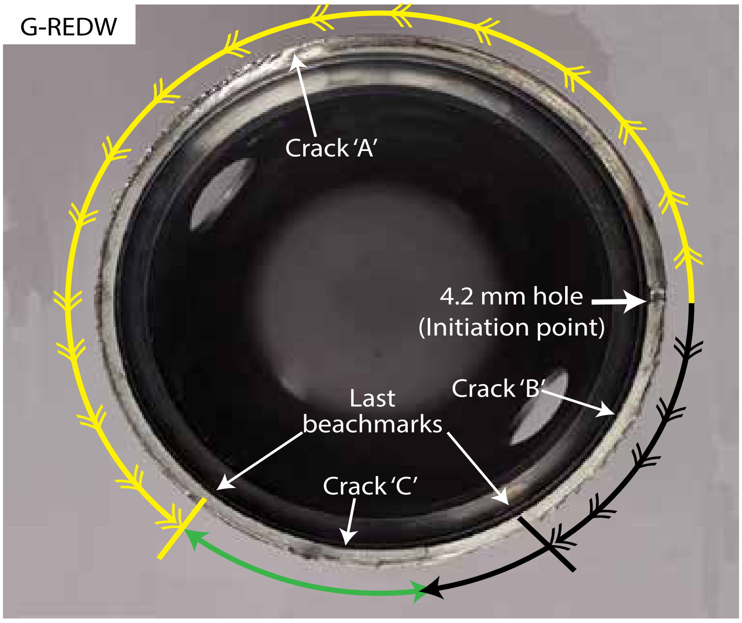


Figure 33

G-REDW fracture surface

1.12.5.3.2 Fracture surface on G-CHCN

On G-CHCN, the principle crack initiated at a corrosion pit 60µm deep located on the inner radius in the parent material (Figure 34). The initiation point was approximately 45º around the circumference of the shaft from the 4.2 mm hole. The crack then propagated in both directions in the parent material in the lower[30] part of the shaft, into the 4.2 mm hole, before it re-initiated on the other side of the hole. As the crack extended around the circumference of the shaft it changed planes a number of times, but always remained in the parent material. In addition, multiple crack initiation sites were identified on the inside of the shaft with small semi-elliptical cracks propagating from the surface. It was assessed that these small cracks initiated and grew as a result of the increase in stress caused by the two major cracks extending around the circumference of the shaft. Both major cracks exhibited evidence of striations and beachmarks, which are associated with fatigue cracking (Figure 35).

[29] See 1.12.5.6 for explanation of striations.

[30] The bevel gear vertical shaft consists of two parts welded together; the lower part of the shaft is called the 'vertical shaft'. The crack on G-CHCN propagated on this part of the shaft adjacent to the weld.

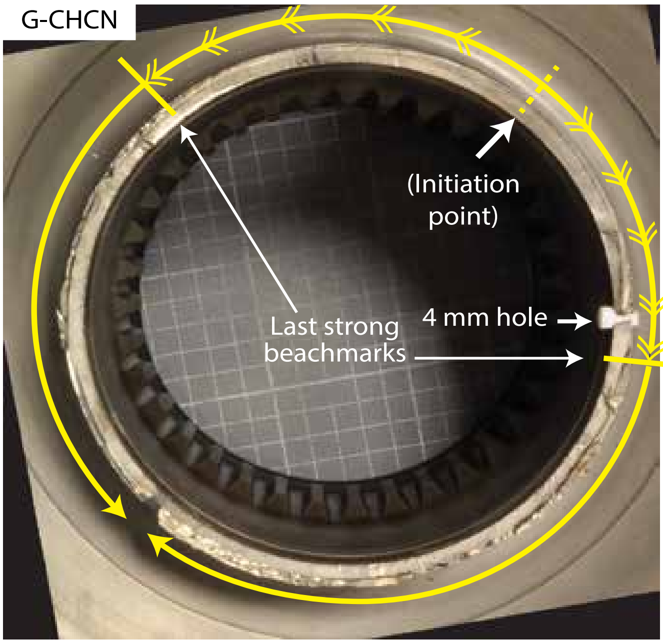


Figure 34

G-CHCN fracture surface

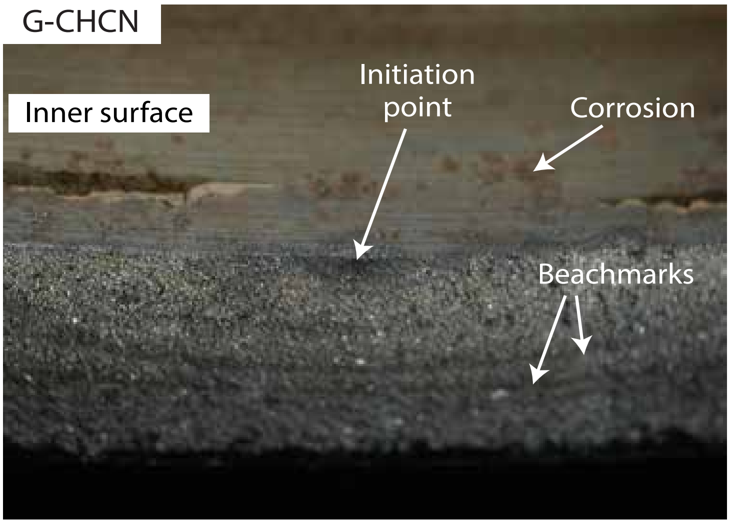


Figure 35

Location of crack initiation on G-CHCN

The machining marks on the inner flange of the shaft, in the area of the weld, appeared to be more pronounced than the marks left on other parts of the shaft. Additionally, there were numerous small tears in the surface consistent with a machine tool vibrating as it cut the metal during the manufacturing of the shaft. Localised pits of corrosion were also present in this area, particularly in the inner radius, and appeared to have developed in the crevices of the machining marks in this location (Figure 36).

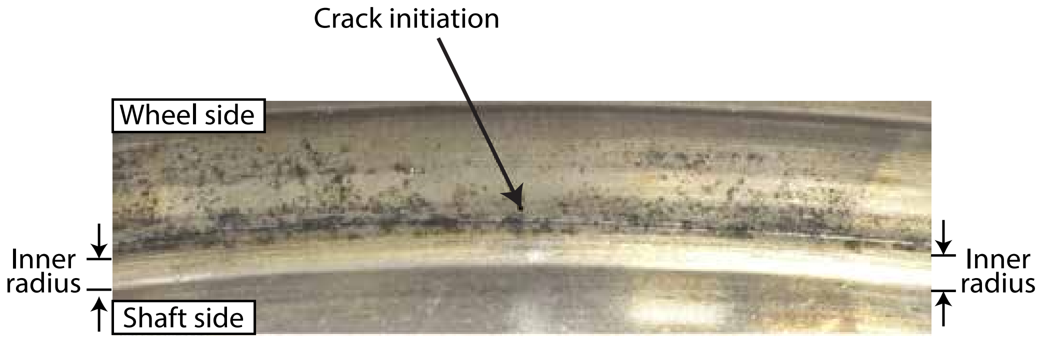


Figure 36

Localised corrosion on inner radius of the shaft on G-CHCN

1.12.5.3.3 Detailed examination of the fracture surface

The fracture surfaces on the bevel gear vertical shafts fitted to G-REDW and G-CHCN displayed trans-granular fracture which is typical of high cycle fatigue. There was a small amount of inter-granular fracture on G-REDW, which may be associated with the weld; however, there was no evidence of inter-granular fracture on G-CHCN. There was no evidence of corrosion on any of the fracture surfaces.

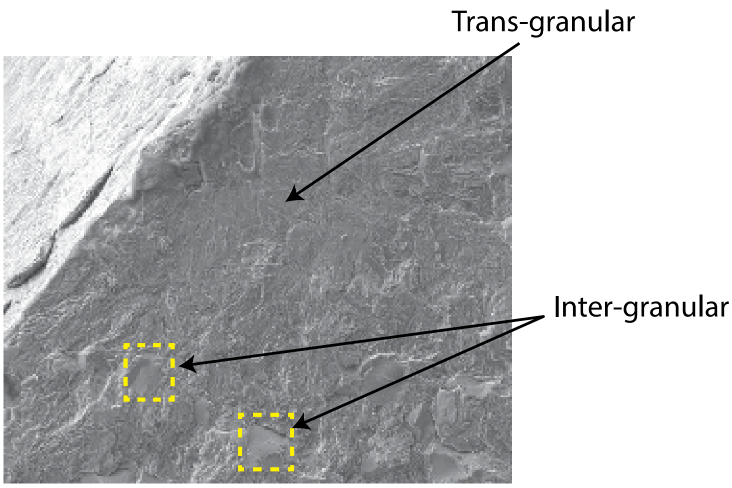


Figure 37

SEM image of fracture surface on G-REDW

1.12.5.4 Surface roughness of the inner flange

A measurement of the surface roughness of the inner flange adjacent to the fracture surface was carried out on samples taken from the shafts fitted to G-REDW and G-CHCN. The measurement was undertaken using a Taylor-Hobson Form Talysurf Series 2 machine with an ISO-2CR filter, and a bandwidth of 300:1.

The R_a is measured taking five consecutive points along the surface. The manufacturing design drawing specifies a general surface finish with a maximum R_a of 3.2µm. The maximum R_a was measured as 2.56 µm for G-REDW and 1.56 µm for G-CHCN. R_z, which is the difference between the highest peak and lowest valley, was also measured and the maximum value was found to be 11.39 µm for G-REDW and 12.03 µm for G-CHCN. There is no design specification for R_z.

1.12.5.5 Detailed examination of the corrosion pits

The corrosion pits at the crack initiation sites on G-REDW and G-CHCN were similar in size and both showed evidence of corrosion products at the root tips. A detailed three-dimensional survey was undertaken on the corrosion pits on G-CHCN and it was established that their depth was approximately 60 µm, and their length and width were approximately 600 µm and 200 µm.

The profile across part of a corrosion pit, mapped by the manufacturer, on the surface of G-CHCN is shown in Figures 38(a) and 38(b). From the profile in Figure 38(a) Cranfield University[31] assessed that the root radius of the corrosion pit was at most 5 µm, which could give a theoretical stress concentration factor (K_t) of the order of 8 to 9[32].

[31] Fatigue lives of corrosion pitted gearbox shafts: Review of investigations and calculations relating to the failures of EC225 Gearbox shafts on G-REDW and G-CHCN. September 2013, Professor P E Irving, Cranfield University.

[32] See Section 1.18.7.5.2.

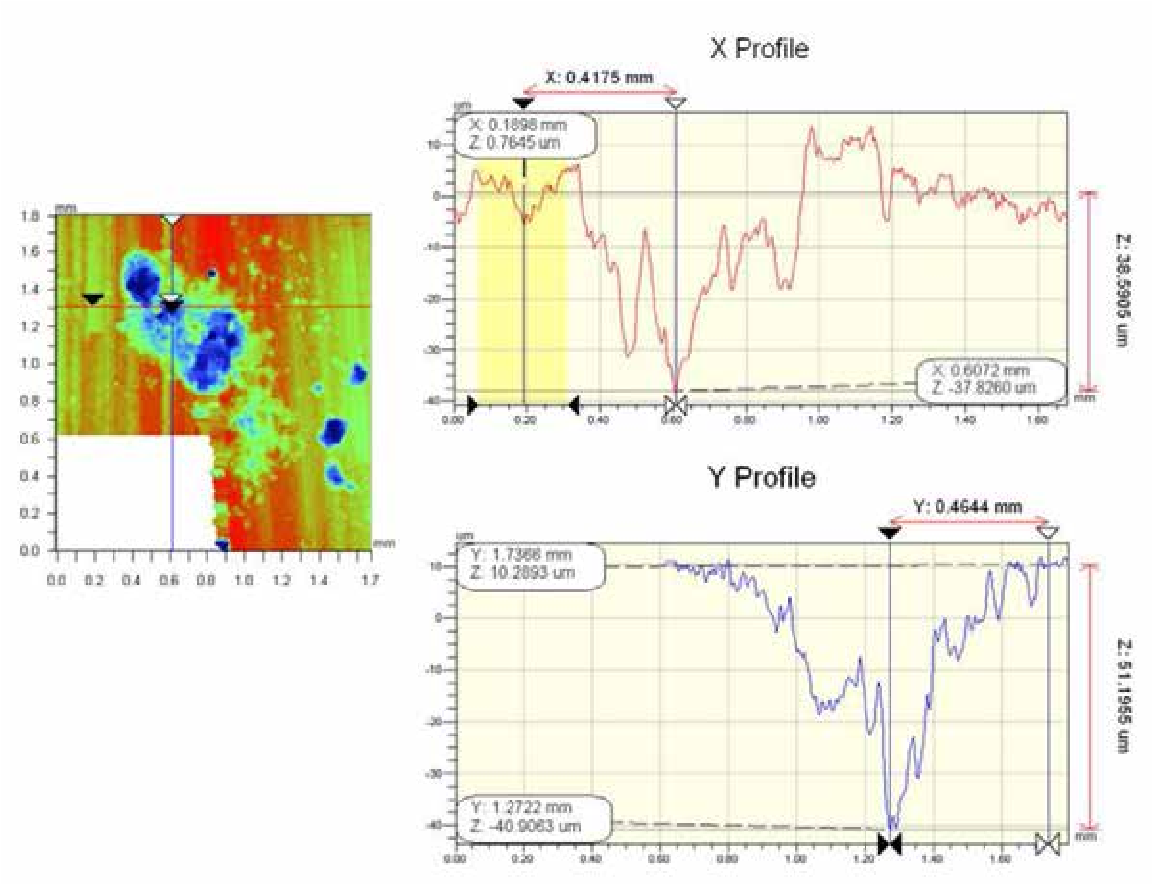


Figure 38 (a)

Profile of a corrosion pit on the shaft fitted to G-CHCN
(Note the scale of the horizontal and vertical axis is different)

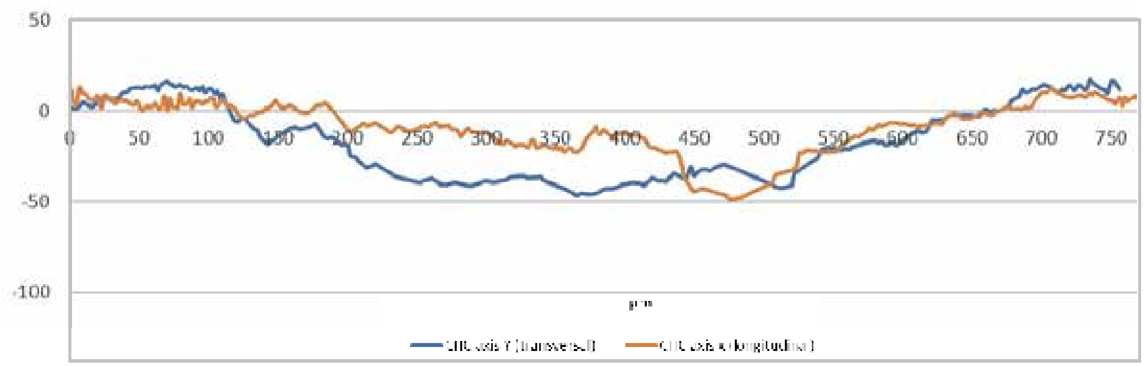


Figure 38 (b)

Profile of corrosion pit on the shaft fitted to G-CHCN
(using the same scale on the horizontal and vertical axis)

1.12.5.6 Striation counting

Striations are normally found on surfaces of high strength steels that have failed in fatigue. Each striation is the result of one load cycle and marks the position of the fatigue crack front at the time the striation was formed. A load cycle is one rotation of the bevel gear vertical shaft. Providing the load on the shaft remains constant, the spacing between striations will increase as the crack grows. However, striations may be difficult to detect, which might affect the estimation of the crack propagation rate obtained from striation counting. For this reason, the manufacturer does not rely on striation counting to assess the crack propagation time.

Striation counting was carried out on the fracture surface of G-REDW by QinetiQ and the helicopter manufacturer. For Crack 'A', the spacing varied from 10.5 striations per µm at the start of the crack to 1 striation per µm at the end of the crack. For Crack 'B' the spacing varied between 6 and 1 striation per µm. From the striations it was estimated by QinetiQ that Crack 'A' was approximately 161 mm long when Crack 'B' started to propagate. The propagation time from the first to last identified striation on the surface of Crack 'A' was estimated by QinetiQ to be 8.5 flying hours.

The manufacturer carried out an analysis of the striations on the fracture surface of G-CHCN and determined that for the cracks on the fracture surface, the spacing varied between 10 to 1.8 striations per µm. Based on the spacing of the striations, the crack propagation rate would have been similar to the crack on G-REDW.

1.12.5.7 Beachmark counting

The fracture surfaces of the shafts on G-REDW and G-CHCN both displayed beachmarks, which can be formed when an event such as an engine start or significant change in torque has taken place (Figure 39). Beachmarks may be difficult to identify and can be interpreted in a number of ways.

On G-REDW, the first beachmark was identified at 4 mm from the initiation point. Using the flight profile (obtained from the CVFDR data) that G-REDW typically flew whilst operating in the North Sea, it was estimated that the time to failure from this mark was between 20 and 31 engine[33] hours. From the recorded data this corresponded to approximately 15 and 21 flying[34] hours; it also equated to 2.16 to 3 million cycles of the shaft when operating under load.

[33] Engine hours is based on the first engine start to the last engine shut down, which corresponds to the MGB operating time.

[34] The flying hours used in HUMS were established from the operation of the air / ground switch. In the AAIB calculation the flying hours were established using recorded data from the radio altimeter.

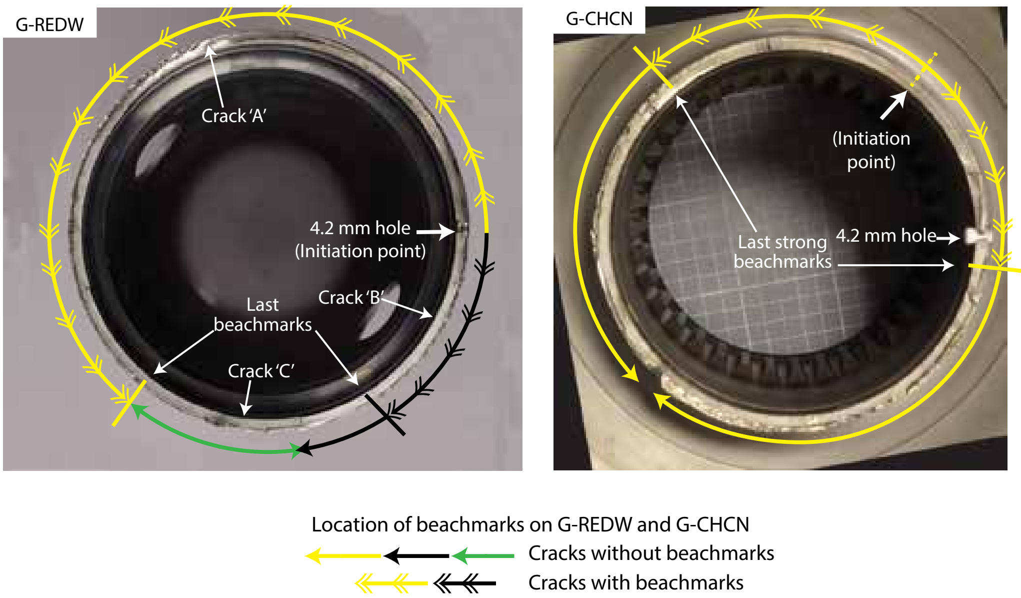


Figure 39

Fracture surfaces on shafts fitted to G-REDW and G-CHCN

The first beachmark on the fracture surface on G-CHCN was identified at 2 mm either side of the initiation point. From the flight profile that G-CHCN normally flew whilst operating in the North Sea, it is estimated that the time to failure from these marks was between 19 and 28 engine hours. From the recorded data this corresponds to approximately 14 and 20 flying hours. It also equates to 2.16 to 3 million cycles of the shaft when operating under load.

1.12.5.8 Time required for cracks to initiate and grow to first beachmark

It was not possible to determine how long it took for the cracks on G-REDW and G-CHCN to initiate and grow to the first beachmark. The growth of the crack from the last identified beachmarks to the final rupture, which was approximately 1% of the cross-sectional area, would have occurred during the accident flights. During the accident flight the cracks on the shaft fitted to G-REDW were estimated by QinetiQ to have grown through an arc of approximately 70º. However, the helicopter manufacturer identified an additional beachmark indicating that it grew through an arc of 30º. On G-CHCN both QinetiQ and the manufacturer agreed that the crack grew through an arc of approximately 215º.

1.12.6 Emergency lubrication system

An examination of the emergency lubrication system was performed and the following observations were made for both G-REDW and G-CHCN:

- There was evidence of glycol (consistent with Hydrosafe 620), oil, and water in the area of the MGB.
- There was no evidence of a rupture of the MGB casing, although Hydrosafe 620 and oil were present in the vicinity of the vent in the MGB casing.
- The fluid recovered from the MGB sumps contained oil and Hydrosafe 620 .
- The Hydrosafe 620 in the reservoirs was recovered. The amounts were consistent with the tanks being full prior to the system being activated and with normal consumption for the period of activation. (The activation period was derived from the CVFDR).
- The installation of the pipework and spray nozzles on the MGBs were satisfactory with no evidence of restrictions.
- The fittings between the engine and bleed-air system, including the restrictor, were correctly installed.
- Although there was signs of water ingress following the ditching, the emergency lubrication system electrical wiring was satisfactory.

1.13 Medical and pathological information

There were no reported serious injuries to the crew or passengers of either G-REDW or G-CHCN.

On G-REDW, whilst in the raft, one passenger suffered from seasickness immediately after he boarded the liferaft, before he was able to take a seasickness tablet. He subsequently required hospital treatment for dehydration. The remaining passengers and crew took seasickness tablets once they boarded the raft.

Only the occupants in one of the two liferafts on G-CHCN took the seasickness tablets stored in the liferaft's rescue pack. This action was instigated by a medic who was amongst the passengers in the liferaft. He commented that the tablets were hard to find in the rescue pack. None of the occupants of this liferaft reported feeling seasick. However, some of the occupants of the other liferaft reported feeling seasick by the time they were picked up by the fast rescue craft, although none were physically sick.

1.14 Fire

Not applicable.

1.15 Survival aspects

1.15.1 Helicopter evacuation G-REDW

Passenger Briefing

The crew of G-REDW alerted the passengers to the possibility of the need to ditch shortly after the loss of MGB oil pressure, giving the passengers approximately 7 minutes to prepare. During this time, the passengers carried out their personal pre-ditching drills, which they recalled from their safety training. They fitted their survival hoods, prepared their rebreathers[35] and located their nearest exit.

Passenger evacuation

Once the helicopter had ditched, the crew briefed the passengers to remain seated until the rotors had stopped. After which, the commander ordered the passengers to evacuate. Both pilots operated the liferaft inflation handles on their respective sides of the cockpit.

Two passengers, on their respective sides of the cabin, jettisoned the main cabin doors, by operating the corresponding internal jettison handles, and the doors fell away from the fuselage cleanly. One of the passengers seated on the left side of the cabin operated the left external liferaft deployment handle.

The left liferaft started to deploy; however, the passengers waiting to vacate by this door stated that they thought that it was slow to inflate.

On the right side of the helicopter, as the right cabin door fell away, the passengers observed that the right liferaft was already starting to deploy. They also stated that the liferaft seemed to be deploying slowly. One of the passengers climbed out of the cabin and onto the sponson. He started to pull the liferaft material out of the recess; this increased the rate of inflation. After the liferaft was fully inflated, he remained on the sponson and restrained the liferaft against the side of the fuselage while the remainder of the passengers boarded. He then boarded the liferaft.

[35] Devices intended to offer the wearer the ability to breathe underwater for a limited period of time to aid escape.

The passengers waiting to leave by the left cabin door became aware that the right liferaft had inflated. These passengers then decided to evacuate through the right cabin door and board the right liferaft. The left liferaft was therefore not used; however, it was later seen to be fully inflated.

Crew evacuation

Following the ditching, both pilots fully jettisoned their respective cockpit doors. The commander vacated the helicopter through his doorway, climbed along the right side of the fuselage, and boarded the right liferaft.

The co-pilot initially vacated the cockpit onto the left side of the fuselage. He saw that the left liferaft had not yet deployed and no-one had exited through that side of the helicopter. He climbed back through the cockpit and exited the helicopter through the right cockpit opening and boarded the right liferaft.

Post-evacuation

Once everyone had boarded the right liferaft, the commander accounted for all persons and the short mooring line was cut. The helicopter and liferaft then drifted apart. When the limit of the long mooring line was reached, the liferaft was upwind and slightly to the right of the nose of the helicopter. The sea swell was causing the helicopter to pitch up and down, with a corresponding vertical motion on the rotor blade tips. One of the blade tips was directly above the liferaft and the vertical motion caused the blade tip to move rapidly and frequently to within a few feet of the liferaft. The co-pilot attempted to restrain the blade, but it made little difference. The proximity of the blade tip caused the occupants of the liferaft concern for their safety as the liferaft was being restrained in a position under the blade tip by the long mooring line. They therefore cut the line, allowing the liferaft to drift clear of the helicopter.

None of the occupants inflated their lifejackets at any time.

1.15.2 Helicopter evacuation G-CHCN

Passenger briefing

The passengers became aware of the emergency on hearing the cabin alarm sounding, followed by a message over the passenger address from the flight crew informing them to prepare to ditch.

Preparation for ditching

In preparation for the ditching, the passengers donned their gloves and neoprene hoods stored in their immersion suits and opened their rebreather mouthpiece covers. Some passengers reported finding it difficult to locate or physically open the mouthpiece cover. They also reported the loss of manual dexterity once they donned the immersion gloves and that this may have contributed to their difficulty in opening their rebreather covers. They considered it would have also made it difficult to operate the cabin window jettison mechanism had they needed to do so.

One passenger was inexperienced as an offshore passenger, identified by him wearing a green armband. Whilst preparing for the ditching, he had placed the rebreather mouthpiece in his mouth; something he was instructed against by some fellow passengers. All passengers were prepared for an evacuation by the time the helicopter ditched.

Crew and passenger evacuation

After ditching, with the engines shut down, the commander cautiously applied the rotor brake to stop the rotors. The helicopter slowly yawed on the water surface through about 90 degrees; the crew deployed the helicopter's sea anchor and the two liferafts using the liferaft inflation handles in the cockpit. The commander reported that the helicopter's sea anchor was effective in keeping the nose of the helicopter into the swell.

The co-pilot climbed into the cabin to oversee the passenger evacuation and slid open the left cabin door. This allowed water to enter the cabin to a depth of about 20 cm. He reported that he opened, rather than jettisoned, this door as he was concerned that it might fall onto, and damage, the liferaft. In opening the door, it now blocked one of the cabin windows, so it could not have been used as an exit had the helicopter subsequently capsized or sunk.

Some passengers remarked on the position of the cabin door jettison handles. These were not adjacent to the doors, so they were concerned that this may have affected the efficiency of their operation had they been required.

Once the co-pilot had exited the cockpit and to assist with an escape should the helicopter capsize, the commander pulled the door jettison handle which released the pins from the cockpit door hinges. The cockpit doors remained latched and stayed in position[36].

When the cabin door was opened by the co-pilot he saw that the left liferaft was fully inflated, but it was being constrained on the sponson by tangled mooring and rescue pack lines. About a quarter of the raft was resting on the sponson, so the co-pilot placed one foot inside the liferaft compartment and pushed the liferaft off the sponson. He pulled the rescue pack out of the water and passed it around the mooring lines to untangle them. Seven passengers and the co-pilot then boarded the left liferaft.

36 By design with the doors latched, they remain closed and need to be physically pushed to jettison them fully from the helicopter.

Meanwhile, the remaining passengers opened the right cabin door. The right liferaft had inflated successfully and was manoeuvred to the side of the helicopter to allow boarding from the cabin door. The remaining ten passengers then boarded the right liferaft.

The commander exited the cockpit, climbed into the cabin, and after ensuring everyone had evacuated the helicopter, boarded the right liferaft via the right cabin door.

Post-evacuation

The sea anchors on both liferafts were deployed, and each short mooring line was cut. The liferafts then drifted down the sides and toward the rear of the helicopter to the full extent of their respective long mooring lines.

The right liferaft was now positioned close to the tail rotor, causing concern among the occupants that the tail rotor blades would hit them or puncture the liferaft. They therefore, cut the long mooring line and the right liferaft drifted clear of the helicopter.

The occupants of the left liferaft also cut their long mooring line, as they had concerns that the helicopter might capsize or catch fire.

The two liferafts joined together, before the occupants were rescued by the Fast Rescue Craft (FRC).

Safety knife

The co-pilot commented that had it not been possible to untangle the lines from the left liferaft, he would have had to have to cut them. He was not provided with a safety knife or line cutter and may have had difficulty in accessing the safety knife attached to the inside of the liferaft.

The commander, who boarded the right liferaft, had a safety knife attached to his flying suit. This had been provided to him by the operator when he was employed on SAR flying duties. The knife attached to the liferaft had come out of its housing and fallen overboard. Therefore, the commander used his own knife to cut the mooring lines. The occupants later recovered the liferaft knife using its attachment lanyard.

1.15.3 Immersion suits

Both pilots on G-REDW were wearing Mulitfabs 411 air crew immersions suits. The suits were designed with an entry zip running centrally up the front and through the split latex neck seal. Both pilots stated that, when the zip is fully closed, the zip fastener running through the neck seal made the suit uncomfortable to wear for extended periods. Consequently, they tended to fly with the zip partly lowered to just below the neck seal.

Before the helicopter ditched, the co-pilot was able to return the entry zip on his suit to the fully closed position. The commander did not close his zip and it remained partly open for the entire event. After the co-pilot got into the liferaft, he found that the neck seal on his immersion suit restricted his ability to move his head so he pulled the zip down to below the neck seal.

At the time of the accident, the operator of G-REDW was in the process of changing the type of immersion suits used by pilots to an orange and black, closed neck seal design. The operator has now issued the suit to all its flight crews.

Neither pilot on G-CHCN was wearing an immersion suit, but both remained largely dry during the evacuation and rescue. There was no requirement, under the operator's standard operating procedures or applicable regulations, to wear one under the prevailing conditions. Both reported that immersion suits were uncomfortable to wear, despite the current model being more lightweight than previous versions, and that a balance had to be struck between the protection they afforded and the risks they presented due to discomfort and heat stress.

All the passengers on both helicopters were wearing immersion suits. Five of the passengers on G-CHCN reported that their feet became wet during the evacuation and rescue. All but one of the immersion suits were recovered after the accident and tested. This revealed leaks in the sock area of five suits, all of which were minor and within the EASA permitted maximum[37]. Further investigation established that three of the leaking suits were from the same manufacturing batch and each had a small slit in a similar position in the tape used to make the seam waterproof over the toes of the sock. As a result of this finding, the supplier added a further layer of tape in this area to all of its suits to provide increased resistance to damage.

1.15.4 Emergency Breathing System (EBS)

The passengers on both helicopters were wearing Lifejacket Air Pocket Plus (LAP Plus) rebreathers.

An EBS is not required under the EASA regulations, nor is there a national or internationally accepted formal standard for their design or manufacture. EASA Rule Making Task 0120 (RMT.0120) was initiated in October 2012 to review rules and advisory material associated with helicopter ditching and survivability,

[37] As specified in EASA ETSO 2C503.

including EBS. The CAA also published CAP 1034 in May 2013, which provided proposed technical standards for EBS for use on helicopters.

After the accident, the manufacturer of the EBS used was passed information on the difficulty experienced by some of the passengers on G-CHCN in identifying and opening the rebreather mouthpiece cover.

The EBS manufacturer was already in the process of developing a new rebreather which would introduce a modified means of accessing the mouthpiece. However, final development and manufacturing were initially planned for 2013, but was awaiting the outcome of the EASA RMT.0120.

As an interim measure, whilst awaiting the completion of the RMT, the manufacturer will upgrade the existing rebreathers, to include a modified means of locating and opening the mouthpiece cover. The change will also include a retaining strap to hold the mouthpiece in place prior to its use when the mouthpiece cover is opened.

1.15.5 Crash position indicators

1.15.5.1 Operation of CPI on G-REDW

The CPI on G-REDW did not deploy and the beacon remained attached to the helicopter. The crew did not manually activate the CPI beacon prior to the emergency evacuation. Therefore, no distress signal was transmitted from the helicopter following the ditching. Photographs taken whilst the helicopter was floating, approximately 8 hours after ditching (Figure 1), show that the water level was above the level at which the water-activated switch was mounted in the cabin.

1.15.5.2 G-REDW CPI operational procedures

The operator's Emergency Procedures checklists (Appendix A), valid at the time of the G-REDW accident, did not contain any reference to the CPI. Following the accident the operator amended the '*Power-On Ditching*' and '*Autorotative Landing or Ditching*' checklists to include the following item after touchdown:

'ACTIVATE …………………. ELT (Check deployed)'

1.15.5.3 Operation of CPI on G-CHCN

The CPI on G-CHCN was manually selected to TRANSMIT by the flight crew during the final preparations for the ditching. At 1424 hrs a 'Detect-only' alert was received by the Aeronautical Rescue Coordination Centre (ARCC) at RAF Kinloss, from a GEO satellite signal. This alert did not provide any positional information, but did contain the 15-digit hexadecimal code unique to G-CHCN. At 1432 hrs an unresolved position alert was received, and at 1453 hrs a further LEO satellite alert was received, confirming the position of G-CHCN.

The CPI beacon remained attached to the helicopter and continued to transmit until it was recovered to land. Photographs taken approximately 24 hours after the ditching (Figure 2), and water damage within the cabin, indicated that the water level had been above that of the water-activated switch.

1.15.5.4 G-CHCN CPI operational procedures

The operator's Emergency Procedures checklists (Appendix A) valid at the time of the G-CHCN accident contained the following item relating to the CPI, in the *'Emergency Evacuation On Water'* checklist:

'CPI …………………….. Confirm deployed'

1.15.5.5 Manual activation of the CPI

Early in the investigation it was determined that once manually selected to TRANSMIT, the CPI will not deploy automatically, either by means of the g-switch or the water-activated switch, unless a system reset, by pressing the TEST / RESET button on the cockpit control panel, is carried out. This system logic effectively renders the water-activated switch, and thus the BRU, redundant following a manual selection of the TRANSMIT function. The helicopter manufacturer was unaware of this feature and no relevant information was included in the EC225 LP Flight Manual. Furthermore, this information was not included in the Type 15-503 CPI Operating Manual published by the CPI manufacturer.

1.15.5.6 CPI examinations

Continuity and insulation resistance tests of the CPI system wiring on both helicopters revealed no abnormalities. A review of the CVFDR data confirmed that the accelerations during both ditching events were insufficient to trigger the g-switches.

1.15.5.7 G-REDW CPI examination

The G-REDW CPI system components, with the exception of the cockpit control panel and the aircraft identification unit, were removed from the helicopter and taken to the CPI manufacturer for examination and testing; they were found to be fully functional. There was no evidence of water ingress in the SIU or the BRU. The BRU was in an undeployed state, indicating that no deploy signal had been received from the SIU. There was no activation code stored in the system memory.

With the components connected to a test bench, submersion of the water-activated switch resulted in activation of the BRU and subsequent transmission of the distress signal. In summary, no defects were identified with the tested components which would have prevented the CPI from deploying automatically during the accident.

1.15.5.8 G-CHCN CPI examination

Although the G-CHCN CPI beacon correctly transmitted distress signals, following the manual activation by the flight crew, all the CPI system components were removed for testing. This was to determine whether there were any issues which could have prevented automatic deployment of the CPI, had this function not been inhibited by the system logic.

The cockpit control unit contained seawater and had suffered extensive internal deterioration due to corrosion rendering it incapable of operating during the testing. The activation code stored in the CPI system memory confirmed the manual TRANSMIT selection of the CPI during the accident.

A visual examination of the water-activated switch showed minor corrosion on one of the contacts, but no evidence of salt water deposits which may have indicated complete immersion in sea water. However, the external electrical connector was corroded, indicating that water had reached at least that level.

The CPI components were installed on the test bench and functioned correctly, leading to successful operation of the BRU and deployment of the CPI beacon.

When the water-activated switch was first immersed in water, it did not activate the beacon deploy signal. On subsequent repeated immersion attempts, the beacon deploy signal was correctly activated but the delay in activation ranged from 15 to 26 seconds. It was subsequently noted that the test bench had a faulty output wire which caused intermittent connections. After the replacement of the output wire, the water-activated switch was subjected to a further twenty immersions in water. The beacon deploy signal activated each time, with the delay being within the specified 5 to 10 seconds. It was not possible to determine whether the initial results had been influenced by the faulty wire on the test bench.

It was concluded that there were no defects with the G-CHCN CPI components that would have prevented the automatic deployment of the CPI beacon, had TRANSMIT not been selected manually.

1.15.5.9 CPI activation after previous EC225 LP accident (G-REDU)

On 18 February 2009 an EC225 LP (G-REDU) struck the surface of the sea during a night visual approach to an oil and gas platform in the North Sea. The AAIB determined that the failure of the CPI to deploy contributed to the delay in locating and rescuing the survivors (AAIB Report 1/2011). The investigation further determined that the CPI on G-REDU should, under the circumstances of the accident, have released automatically and commenced broadcasting on the distress frequencies. The reason for the failure of the CPI to deploy was not fully determined; however, a number of possibilities were considered in the report. As a result of the findings of the investigation, Safety Recommendation 2011-071 was made on 14 September 2011:

> **Safety Recommendation 2011-071**
>
> It is recommended that the European Aviation Safety Agency reviews the location and design of the components and installation features of Automatically Deployable Emergency Locator Transmitters and Crash Position Indicator units, when required to be fitted to offshore helicopters, to ensure the reliability of operation of such units during and after water impacts.

The EASA responded to Safety Recommendation 2011-071 as follows:

> *'A rulemaking task was initiated in May 2012 (Reference: RMT.0120 …), which aims to undertake a broad review of helicopter ditching, water impact events and subsequent occupant survivability. A determination will be made on how certification rules and guidance material can best be developed to further enhance helicopter safety. The installation and functioning of all types of Emergency Locator Transmitters following water impact events is an integral part of this task. Both future and retroactive certification requirement are being considered.'*

The EASA formed a working group to support this rulemaking task, with the first meeting taking place in January 2013. The work is planned to conclude in September 2015. The terms of reference for RMT.0120 include the following item, specific to ELTs:

> *'Based on accident and incident data, identify issues related to ELT/ PLB*[38] *installation and functioning that have resulted in poor in-service experience. (This task is linked to Rulemaking Task RMT.0274, which will consider broader issues relating to ELT installation and functioning and aims to provide consistent regulation across all CSs [Certification Specifications]).'*

[38] Personal Locator Beacon.

RMT.0274 has not yet been initiated by the EASA, and is not likely to commence until 2017, but its aims include:

> *'Incorporate, in the aircraft Certification Specifications, provisions to enforce installation requirements as provided in ED-62A standard. The objective is to ensure that the signal between the ELT unit and the antenna is not disrupted after a crash …'*

1.15.5.10 CPI system modification following G-REDU accident

The Type 15-503-134-1 CPI installation on G-REDU, G-REDW and G-CHCN included the 503-21 standard of BRU. Following the accident to G-REDU, the CPI manufacturer developed a new standard of BRU (503-21-1) incorporating an integral water-activated switch, in addition to the separate cabin-mounted water-activated switch. The BRU integral water-activated switch is independent of the aircraft wiring, and will act to deploy the CPI automatically if the BRU, mounted behind the CPI, becomes submerged. Thus automatic deployment of the CPI may occur, even if TRANSMIT has previously been selected.

BRU 503-21-1 is only compatible with SIU 503-24 with modification state -3 and above. It is designed to increase the likelihood that the CPI beacon will deploy without dependency on the SIU, for example if the SIU is damaged, if the CPI wiring is damaged, or if none of the other SIU triggers have been activated. The BRU will remain functional for up to 15 minutes after power is removed from the SIU, after which an automatic 'power down' switches the BRU to OFF.

1.15.5.11 ELT Minimum Operational Performance Specifications

In November 2013 the EUROCAE Council approved the creation of Working Group (WG) 98, *Aircraft Emergency Locator Transmitters,* with the objective of improving ED-62A MOPS. This activity has been largely prompted by the need to develop MOPS for second generation ELTs based on more capable Medium Earth Orbit (MEO) satellite constellations and capable of activating in-flight, when it is detected that a crash situation is imminent. The scope of WG-98 is still being defined, but the initial Terms of Reference state that the areas to be addressed include the following:

- Create a new class of automatically activated next generation ELTs prior to impact.
- Definition of ELT technical requirements.
- Definition of the criteria for in-flight activation.

1.15.5.12 Other accidents

On 23 August 2013 an AS332 L2 Super Puma, registration G-WNSB, crashed in the North Sea during approach to Sumburgh Airport. Four of the occupants were fatally injured. The investigation into this accident is ongoing. It is known that the CPI deployed and transmitted following the accident. G-WNSB was initially equipped with a Type 113 CPI at manufacture, but this was retrospectively modified to a Type 15-503-134 standard CPI system, which included a beacon deployment control (BDC) unit adjacent to the BRU. The BDC provides and controls the power supplies for automatic activation and release of the beacon in the event of the CPI system wiring being severed. A review of the stored memory in the aircraft identification unit from G-WNSB confirmed that the CPI beacon was deployed automatically when wiring to the CPI was disrupted, most likely when the tailboom separated from the fuselage. The BDC is not included in the Type 15-503-134-1 standard CPI system installed on G-REDU, G-REDW and G-CHCN, nor fitted at manufacture on any other EC225 LP helicopter.

1.15.5.13 CAA ADELT research

The UK CAA has been actively monitoring issues related to the performance of ADELTs for several years. They have completed a research study into the likely causes of ADELT performance problems based on a review of Mandatory Occurrence Reports and aircraft accident reports relating to North Sea helicopter accidents and incidents over a 26 year period. The findings of this research have been published in Civil Aviation Publication (CAP) 1144 *'ADELT Review Report',* dated 27 February 2014.

The research identified that the majority of issues associated with non-deployment of ADELTs were related to equipment selection and installation. It concluded that ADELT functionality is influenced by, among other things, the location of the ADELT, its dedicated power supplies and the ADELT system sensors in the aircraft. In particular the research found that the functionality of an ADELT could be compromised if the system wiring was disrupted during the accident, such as could occur when the tail boom separates from the rest of the fuselage.

CAP 1144 contains a number of recommendations aimed at optimisation of ADELT installations and designs to maximise the likelihood of an ADELT deploying and transmitting correctly. In particular, it recommends that the EASA develop guidance material to assist designers of future ADELT / CPI installations; develop specific design requirements for ADELTS; and re-evaluate current ADELT installations. The full text of the recommendations is included in Appendix E.

1.15.6 Liferafts

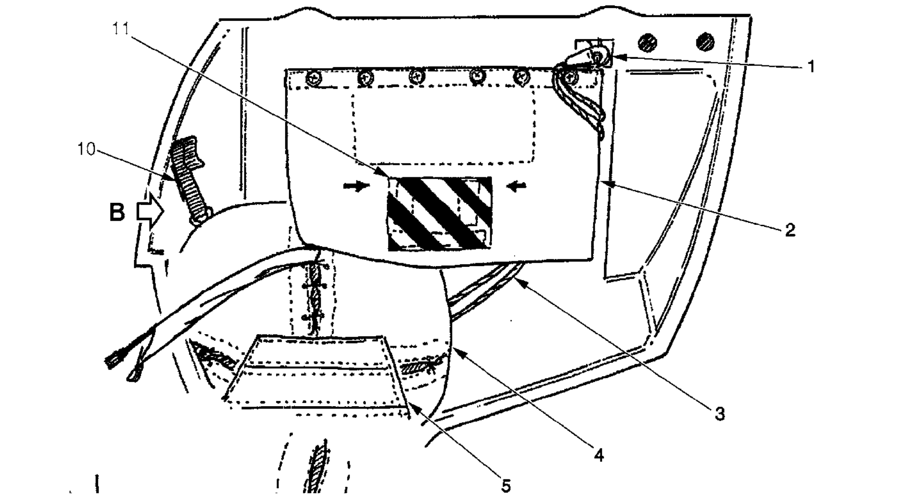
The liferafts from G-REDW and G-CHCN were examined and no defects were found. Photographs and witness accounts provided evidence that all four rafts were serviceable and had fully inflated.

On G-REDW, there were reports that the liferafts appeared to be slow to deploy. There were no other reported difficulties with the liferaft deployments.

On G-CHCN's left liferaft the mooring lines and rescue pack lines became entangled. To investigate this issue, a number of liferaft installations on the operator's fleet of helicopters were examined, a liferaft installation was observed and the liferaft installation and packing instructions were reviewed.

1.15.6.1 Liferaft installation instructions

The EC225 LP aircraft maintenance manual (AMM) contains a procedure in workcard 25-66-01-061 (version 2012.06.14) for installing the liferaft, inflation bottle and rescue pack. The procedure calls for the liferaft to be installed first and then it states to '*Attach mooring lines (3) to spring-loaded hook (1) and bind them to upper flap (2) using Commercial Adhesive tape.*' These items are shown in Figure 3 of the procedure, reproduced in Figure 40 of this report. There is a spring-loaded hook secured to the end of the mooring lines and the intention of this instruction is for this hook to be attached to the bracket fixed to the sponson at position (1) in Figure 40. The figure does not show how to tape the mooring lines to the upper flap and the lines are shown to pass in front of the flap, when they should pass behind it as shown in Figure 41. This tape is removed after the inflation bottle is installed. The rescue pack is installed last. Figure 40 also shows the mooring lines (3) routing behind and below the liferaft which is different from the routing of the lines in an installation test carried out by the helicopter manufacturer (Figure 41).



1 spring-loaded hook 2 upper flap 3 mooring lines 4 liferaft
5 lower flap 10 recovery strap 11 handle pouch

Figure 40 *(Courtesy of Eurocopter)*

Extract from Figure 3 of
workcard 25-66-01-061- liferaft installation procedure

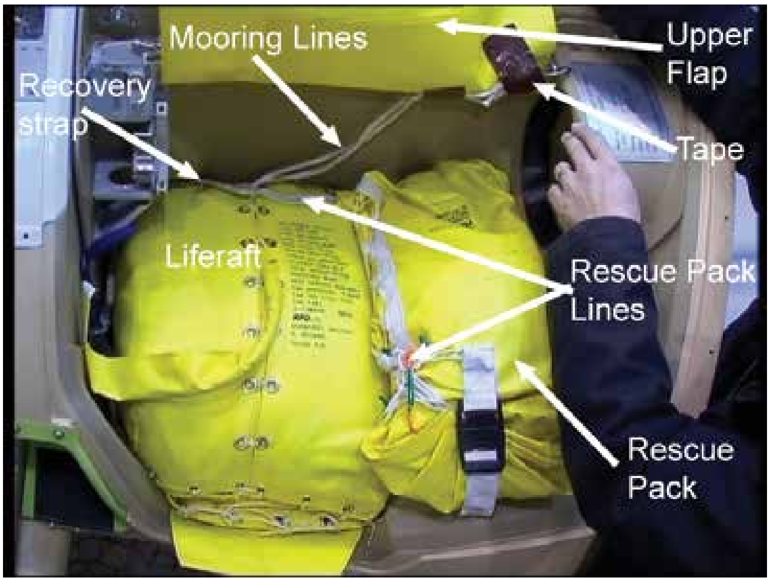
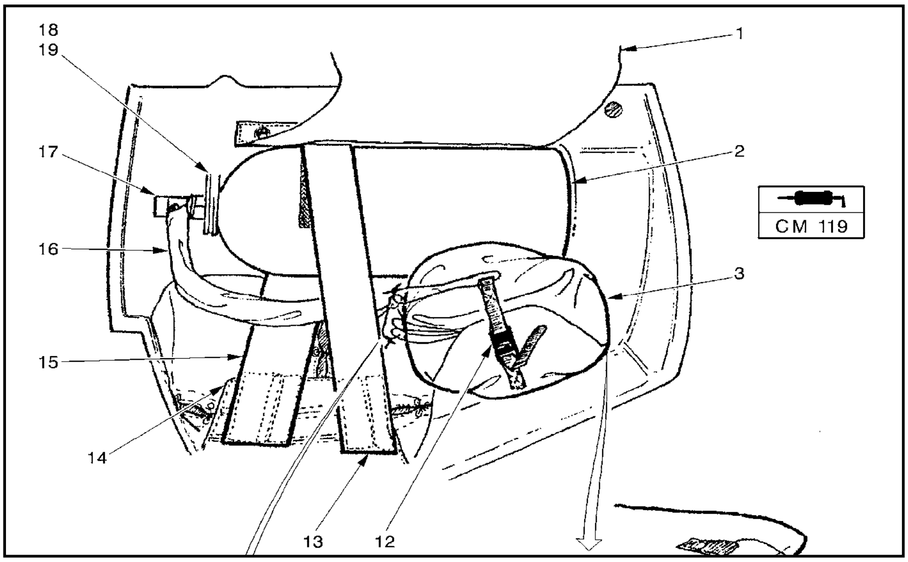


Figure 41

Image from the helicopter manufacturer's installation test showing
the routing of the mooring lines and rescue pack lines

The liferaft installation procedure contains a Caution which states that the mooring lines, recovery strap and rescue pack lines '*MUST NOT CROSS EACH OTHER*', but the actual routing of these lines is not shown. An additional installation diagram, Figure 42 (Figure 2 in the workcard), shows the inflation bottle, rescue pack and liferaft installed, although the liferaft is not easily visible in the lower left part of the image, and the mooring lines are not shown.



(*Courtesy of Eurocopter)*

Figure 42

Extract from Figure 2 of
workcard 25-66-01-061- liferaft installation procedure

Although the instructions state clearly that the mooring lines and rescue pack lines must not cross, whether this is achievable or not depends upon how the lines exit the liferaft pack. In the liferaft installation shown in Figure 43 it was not possible to route the lines without crossing them because the mooring lines exited the liferaft pack forward of the rescue pack lines. The liferaft packing instructions do not specify if one set of lines needs to be forward of the other – for more detail on packing see section 1.15.6.3.

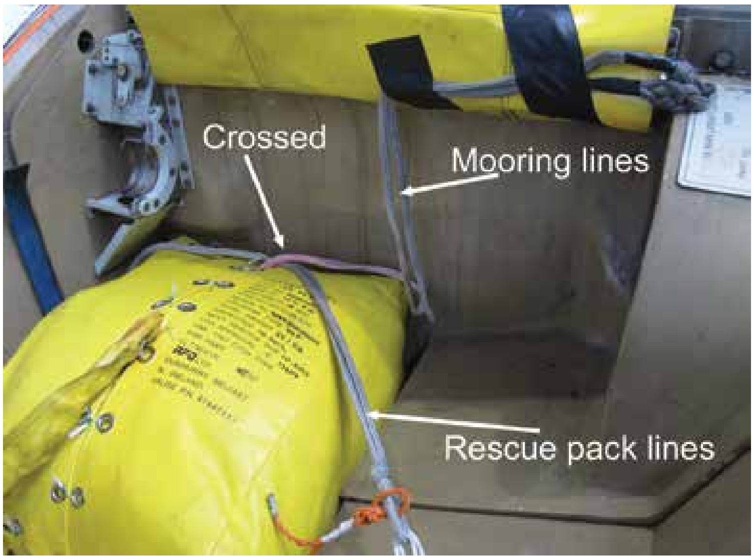


Figure 43

Liferaft installation showing crossed mooring and rescue pack lines

1.15.6.2 Liferaft installation examinations

Six liferaft installations were examined at the operator of G-CHCN's maintenance facility on a combination of EC225 LP and AS332 L2 helicopters which have the same type of liferaft installation.

On four of the installations the mooring lines and rescue pack lines were routed correctly, that is they were not crossed and the mooring lines were routed behind the inflation bottle as shown in Figure 44. However, since the exact routing of the rescue pack lines is not specified in the manual, it is not possible to be certain that they were routed as intended by the helicopter manufacturer.

Figure 44

Mooring lines shown correctly routed behind the inflation bottle

On the remaining two installations, both fitted to EC225 LP, G-WNSO, there were problems with the installation of the lines. This helicopter was undergoing maintenance and both liferafts had been replaced after servicing. The rescue packs had been left open, as the PLBs[39] had not yet been fitted.

In the left compartment of G-WNSO the mooring lines were routed in front of the inflation bottle instead of behind it and the rescue pack lines were twisted around the mooring lines (Figure 45). The engineer who installed this liferaft was interviewed several months later and could not recall fitting the liferaft. He was aware that the mooring lines should be routed behind the bottle. He stated that sometimes it was difficult to install the bottle with the mooring lines behind it, which may have been the reason why he routed the mooring lines in front of the bottle. He could not explain why the mooring and rescue pack lines were left in a twisted state.

39 PLB is the personal locating beacon which are serviced separately from the liferafts.

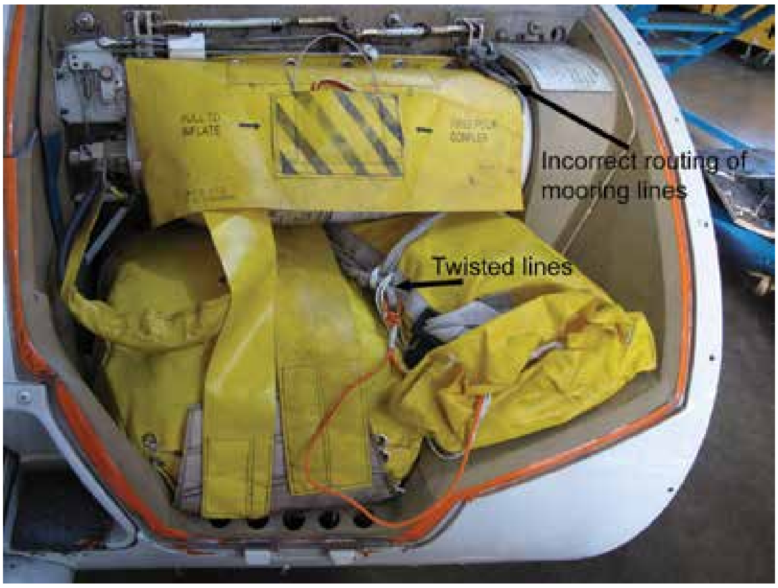


Figure 45

G-WNSO left compartment - incorrect routing of mooring lines in front of the bottle and rescue pack lines twisted around mooring lines

In the right liferaft compartment of G-WNSO, the mooring lines were not connected to the sponson, despite the inflation bottle having already been installed. Also, the rescue pack was not orientated correctly. The engineer who had installed this liferaft stated that sometimes he installed the rescue pack sideways because the PLB made it difficult to fit the rescue pack normally. He could not explain why he had installed the bottle without first attaching the mooring lines to the sponson.

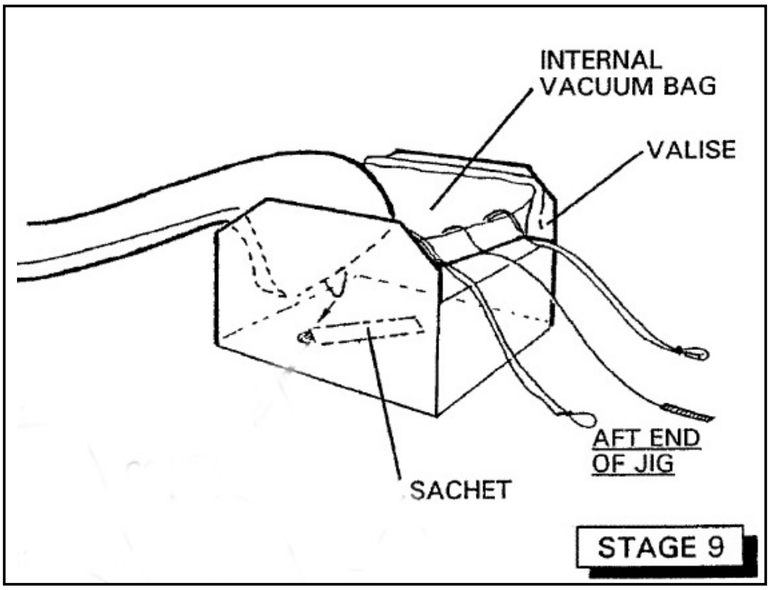
Neither of the engineers involved with fitting the liferafts to G-WNSO were involved with fitting the liferafts to G-CHCN. The engineer who installed the left liferaft on G-CHCN had worked on Super Pumas for the previous eight years and he estimated that he had installed about 20 liferafts in that time. He could not recall fitting the liferaft to G-CHCN but he stated that he knew the mooring lines needed to be routed behind the bottle and he did not think he had ever routed them in front. He was licensed on the AS332 L2 but not on the EC225 LP, so although the liferaft installations are the same, after he fitted the liferaft the work was certified by another engineer who was licensed on type. He stated that the engineer who certified the work should have identified any installation errors so he thought that an error was very unlikely.

In November 2013 the operator of G-CHCN reported to the AAIB that during a liferaft change on a Super Puma based in Australia, it was discovered that the mooring lines had been routed in front of the inflation bottle and that a joining cord had been used between the spring-loaded hook and the attachment bracket. This prompted the operator to carry out a worldwide check of liferaft installations on all of its in-service Super Puma helicopters and no anomalies were identified.

1.15.6.3 Liferaft packing instructions

The liferafts are required to be serviced annually. An EASA Part 145 Approved Maintenance Organisation (AMO) serviced the left liferaft on G-CHCN on 20 February 2012. The liferafts are required to be packed in accordance with the Component Maintenance Manual (CMM) for the Type 18R MK 3 liferaft (issue 2 from June 1998 was current at the time).

The instructions in the CMM contain diagrams for how to fold the liferaft, but do not depict the rescue pack line, mooring lines or the retrieval line. The first diagram to show these lines is CMM Figure 712, reproduced here in Figure 46, where the liferaft is already packed in the valise.

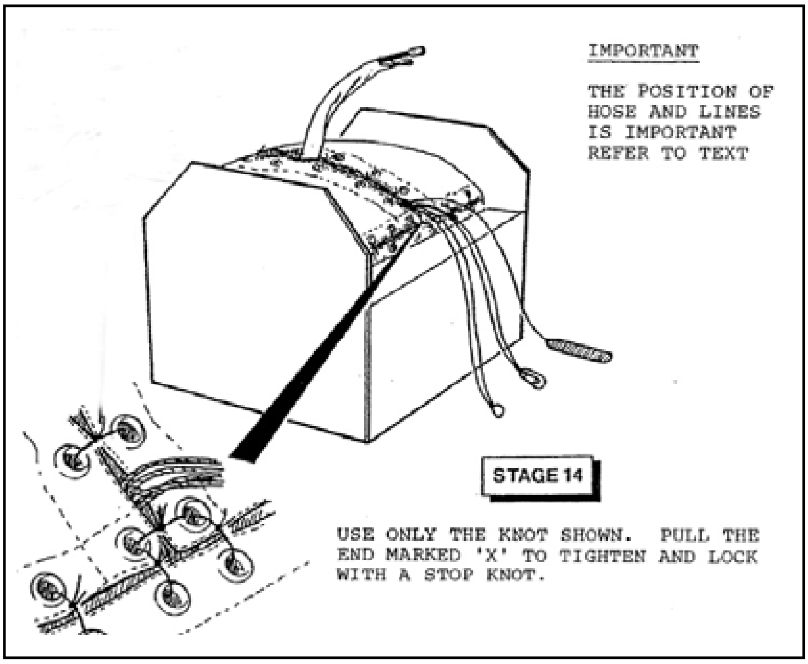


(*Courtesy of Survitec)*

Figure 46

Figure 712 from liferaft CMM showing mooring, rescue pack and retrieval lines

The retrieval line, which has a strap on the end, can be identified in the centre, but the other two lines are difficult to identify. The liferaft manufacturer stated that the lines furthest away in the diagram are the mooring lines and the ones closest are the rescue pack lines. In a later diagram, Figure 47 (CMM Figure 714), the lines appear in a different order but there is no explanation in the text for this.



(*Courtesy of Survitec)*

Figure 47

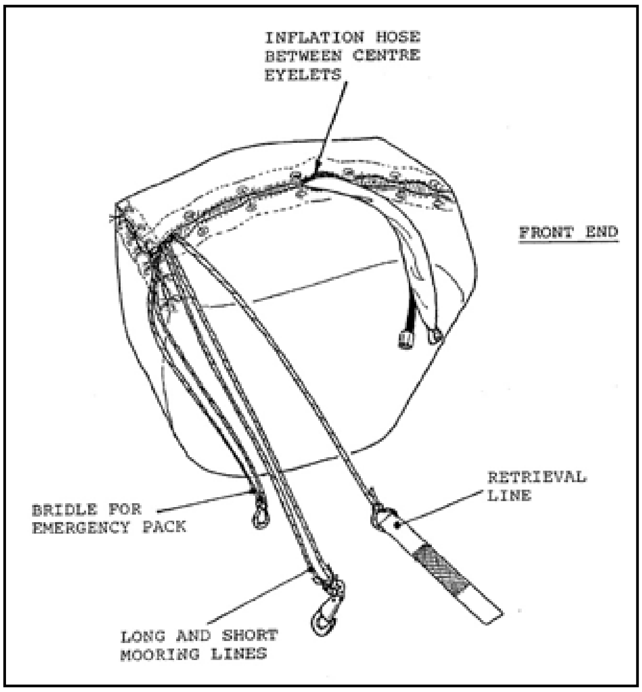
Figure 714 from liferaft CMM – lines shown in different order from Figure 712

The final diagram in the packing instructions, Figure 48 (CMM Figure 715), shows the rescue pack line[40] emerging at the back of the valise and the mooring lines forward of these. This orientation will result in these lines crossing when the liferaft is installed in the sponson because the rescue pack line is routed forwards while the mooring lines are routed rearwards. The instructions in the text relating to the lines state (note: the 'bridle' is the rescue pack line):

> *Fig. 712. 'Make sure that the bridles and operating lines are hung over the side of the jig as detailed.'*
>
> *Fig. 713 and Fig 714: 'Lift the lines and the hose out of the valise. Position the bridle and lines between the first and second pair of eyelets at the aft end of the valise join.'*

There are no further instructions detailing how to pack the lines and there are no notes of caution about not crossing the lines.

40 In the CMM the rescue pack is referred to as the 'emergency pack'.



(*Courtesy of Survitec*)

Figure 48

Figure 715 from liferaft CMM – final orientation of lines

The Part 145 AMO that serviced the left liferaft on G-CHCN was asked if they were aware of any packing issues. A search of their database revealed four reports of incorrectly packed liferafts between May 2008 and September 2013. These were discovered when the liferafts were sent to them for their annual service. In all four cases the mooring, rescue pack and retrieval lines exited the valise at its forward end rather than its aft end (Figure 49). This would have resulted in incorrect inflation of the liferaft, causing the lines to twist, and would also have posed a risk of puncture. In two of the cases the liferafts had been packed by organisations in Brazil and Australia that were not registered with the liferaft manufacturer and therefore would not have had access to official copies of the CMM or updates.

In the aviation industry an AMO, such as an EASA approved Part-145 maintenance organisation, can service a liferaft providing they have an up-to-date copy of the CMM for the liferaft. A Part-145 organisation also has to demonstrate that personnel are suitably trained and competent to perform the required maintenance tasks. This training can be provided by the equipment manufacturer or by the Part-145 organisation.

Figure 49

Incorrectly packed liferaft with the lines exiting at the front instead of the rear – discovered during servicing

In the maritime industry, liferafts need to be serviced by an approved servicing station, and the liferaft equipment manufacturers are responsible for ensuring that each servicing station accredited by them has qualified persons whom they have adequately trained and certificated[41].

In the aviation industry the liferaft manufacturer does not accredit the AMO. However, the Part-145 organisation that serviced G-CHCN's left liferaft stated that some of their staff had been trained by the liferaft manufacturer and that these staff had then trained new staff.

1.15.6.4 Liferaft regulations and certification

The Type 18R MK3 liferaft is a variant of the Type 18R MK1 liferaft that was certified in 1986. The Type 18R MK3 liferaft was certified in 1992 and was designed to be stowed in the sponson of the AS332. The differences between the MK1 and MK3 liferaft included relocation of mooring lines, knife and sea anchor. There were 27 deployment tests as part of the MK 3 approval, 24 involving sponson containers or mock containers, and 3 using the actual helicopter. Of these 27 tests, two were carried out with the container partially submerged in water; the rest were carried out on land. The helicopter manufacturer also carried out one inflation test with the liferaft fitted to a helicopter on the ground. The certification requirements at the time (in JAR-29) did not contain any test requirements for remotely inflatable liferafts in sponsons, and neither do the current CS-29 requirements. CS-29 concerning liferafts states the following:

41 International Maritime Organisation (IMO) Resolution A.761(18) adopted on 4 November 1993, '*Recommendation on Conditions for the Approval of Servicing Stations for Inflatable Liferafts*.'

> *'CS 29.1411 Safety Equipment, General*
>
> *(d) Liferafts. Liferafts must be stowed near exits through which the rafts can be launched during an unplanned ditching. Rafts automatically or remotely released outside the rotorcraft must be attached to the rotorcraft by the static line prescribed in CS 29.1415.'*
>
> *'CS 29.1415 Ditching equipment*
>
> *(b) Each liferaft and each life preserver must be approved. In addition:*
>
> *(1) Provide not less than two rafts, of an approximately equal rated capacity and buoyancy, to accommodate the occupants of the rotorcraft; and*
>
> *(2) Each raft must have a trailing line, and must have a static line designed to hold the raft near the rotorcraft but to release it if the rotorcraft becomes totally submerged.'*

Apart from specifying that the liferaft must be securely attached to the helicopter and automatically release if the helicopter submerges, there are no additional requirements relating to its deployment mechanism or reliability. There are, however, detailed requirements for the performance of the liferaft as a stand-alone item in terms of inflation performance and floatation performance. These were detailed in CAA Specification No. 2 '*Inflatable Liferafts*' (Issue 2, November 1985) at the time of the MK1 approval, and are detailed in ETSO[42] 2C505 – the current requirements for liferafts. The requirements in the current ETSO are almost identical to the requirements in the CAA Specification No. 2 and among these are:

- The liferaft shall be packed into a valise or container which in turn will be stowed and restrained on board the aircraft.
- The valise or container shall include suitable lifting handles so the packed liferaft can be moved within the aircraft.
- Inflation initiation shall be within the capability of one person, either in or out of water.
- A painter line shall be provided that is not less than 6 m and not greater than 20 m (ETSO requirement, CAA Specification No. 2 lists a lower limit of 6 m but does not list an upper limit).

[42] European Technical Standard Order (ETSO) 2C505, Subject: '*Helicopter Liferafts for Operations to or from Helidecks Located in a Hostile Sea Area*'.

- The raft shall inflate and be suitable for boarding within 30 seconds of the start of inflation when soaked at temperatures between -30ºC and +65ºC (+70ºC in CAA Specification. No. 2).

- '*The method of packing the liferaft into its valise or container shall be such that the liferaft will* successfully *deploy in the correct attitude for boarding with a probability of not less than 0.90 under the conditions prescribed in paragraph 16*.' Paragraph 16 states that the liferaft shall be capable of withstanding sea and wind conditions of at least Sea State 6 and 60 km/h (40 mph) respectively.

- Tests shall include inflations in both calm and disturbed water ('*e.g. in a swimming pool and in choppy sea or simulated choppy sea conditions*').

None of the test requirements relate specifically to the inflation of a liferaft installed in aircraft structure, such as a sponson. The requirements in Paragraph 16 of the ETSO relate solely to the liferaft's post-inflation performance and not its deployment from a sponson. As a consequence there is no requirement to carry out an inflation test of the liferaft fitted to a sponson. There is also no requirement to consider the possible different aircraft attitudes when inflation might be required.

In addition to CS-29 and ETSO 2C505 there are operational requirements for liferafts in JAR-OPS 3. JAR-OPS 3.830 states that a helicopter carrying more than 11 persons and operating more than 10 minutes flying time from land, must carry a minimum of two liferafts sufficient to accommodate all persons on board. The Acceptable Means of Compliance (AMC) relating to 3.830, states that 50% of the liferafts '*should be jettisonable by the crew while seated at their normal station, where necessary by remote control*'. In the case of the EC225 LP both liferafts meet this requirement. The AMC also details the survival kit that should be provided in the liferaft. Of the items listed in the AMC, the police whistle and sea water desalting kit were not provided on the Type 18R MK3 raft survival kit. Although JAR-OPS 3.830(a)(5) states that only life saving equipment '*as appropriate to the flight to be undertaken*' is required to be included in the liferaft.

1.15.6.5 Liferaft deployment following previous EC225 LP accident (G-REDU)

On 18 February 2009 an EC225 LP, registration G-REDU, operating in the North Sea, struck the sea on approach to an offshore installation (AAIB Report 1/2011). The report states:

> *'As the left liferaft was inflating, it seemed to the passengers to be restricted by various lanyards, so some of them assisted with its deployment. The right liferaft external deployment handles on the side of the fuselage were pulled but the liferaft did not deploy as expected, so the passengers manually removed the liferaft from its housing in the right sponson allowing it to inflate.'*

1.15.6.6 Liferaft inflation time

Some of the passengers on G-REDW commented that the liferafts were slow to deploy. The passengers on G-CHCN did not make such comments, but in the case of G-CHCN both liferafts were deployed by the pilots before the doors were opened. In the case of G-REDW the passengers quickly jettisoned the doors before the liferaft inflation had initiated, so they had to wait for them to inflate before they could board. ETSO 2C505 requires that the liferaft inflates and is suitable for boarding[43] within 30 seconds at -30ºC, and the same requirement existed in CAA Specification No. 2. The 18R MK3 raft did not meet this requirement as its inflation time at -30ºC was 44 seconds. The Declaration of Design and Performance (DDP) for the MK3 liferaft[44] included this fact as a '*Departure from Specification*'. The DDP also stated that the inflation time at 20ºC was 23 seconds[45]. The actual inflation time in the case of G-REDW is not known and the perception of time can change in stressful situations, but given that the air temperature was about 9ºC, the liferafts should have inflated in about 30 seconds.

1.15.6.7 Length of long mooring line

Following the ditching of G-REDW, in which all occupants boarded the right liferaft, the short mooring line was cut and the raft was allowed to drift away from the helicopter to the limit of the long mooring line which was reached when the liferaft was slightly to the right of the nose of the helicopter. One of the main rotor blade tips was directly above the liferaft and the sea swell was such that the blade moved rapidly up and down. The motion of the blade tip and its proximity caused concern to the survivors so they decided to cut the long mooring line.

43 The liferaft manufacturer described 'suitable for boarding' to mean that the primary buoyancy chambers are rounded out to shape but not yet necessarily at the working pressure of the chamber.

44 DDP No. 127 Issue 5, CAA Ref. E13607.

45 The inflation time is lower at higher temperatures due to the higher pressure of the gas in the inflation bottle.

Following the ditching of G-CHCN, the occupants of the right liferaft were concerned about their proximity to the tail rotor and decided to cut the long mooring line. The long mooring line is intended to keep the liferaft attached to the helicopter at a safe distance in order to aid location, and it automatically releases if the helicopter sinks. In the case of the Type 18R MK3 liferaft the long mooring line is 12 m long. The distance from the liferaft attachment point in the sponson to the forward most position where a main rotor blade tip could reach is about 10 m (Figure 50). However, any slack in the line due to wave motion could result in part of the liferaft being beneath the main rotor blade tip.

Neither the helicopter nor the liferaft manufacturer could explain why 12 m had been chosen for the length of the long mooring line. At the time of the liferaft's approval the CAA Specification No. 2 only required that it be longer than 6 m. Currently, ETSO 2C505 requires that it be between 6 and 20 m and the AMC to JAR-Ops 3.830 states it should be 20 m long.

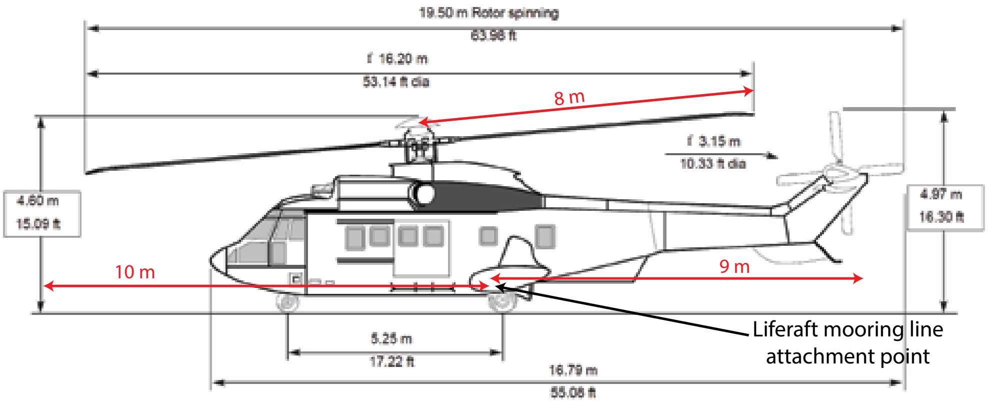


Figure 50

Dimensions of EC225 LP highlighting liferaft mooring line attachment point

1.16 Tests and research

1.16.1 Oil Sampling

As the emergency lubrication system had operated on G-REDW and G-CHCN, the oil in their MGBs had been contaminated with glycol and water. The oil also contained a significant amount of iron particles that were generated as the upper part of the shaft continued to rotate after it failed.

In November 2012, as part of the investigation, oil samples were taken from the MGBs of six EC225 LP helicopters based at Aberdeen. The results of these tests are shown in Appendix H.

1.16.2 Survey of shafts for evidence of red deposit (containing iron oxide)

In May 2013 the manufacturer asked its operators and repair stations to undertake a visual survey of the inside of the bevel gear vertical shafts fitted to EC225 LP and EC725[46] helicopters to inspect for the presence of a red deposit of the type found on the inside of the shaft fitted to G-CHCN.

Residue was identified on 20 of the 28 shafts and a few 'corrosion like' marks were present on 15 of these shafts. Where the corrosion was present, it was found in the same areas as seen on G-CHCN. The amount of deposit and corrosion found on these shafts was significantly less than that seen on G-CHCN.

The results of the small survey suggested that there is a relationship between the quantity of the red deposit, the amount of corrosion and the shaft life. However, none of the shafts inspected had reached their second overhaul due at 4,000 hours.

1.16.3 Corrosion

1.16.3.1 Active corrosion

Corrosion is the gradual destruction of a material as a result of a chemical reaction within its environment.

Corrosion fatigue is a failure mode that occurs under the concurrent action of corrosion and cyclic loading. With corrosion fatigue the material no longer has a fatigue limit below which cracking would not occur. A high ratio of inter-granular to trans-granular cracking in the fracture surface can be an indication of corrosion fatigue. A description of the fatigue limit is given at Appendix G.

The term 'Active Corrosion' was used by the manufacturer during this investigation to describe the first stage of corrosion fatigue that may have occurred at the start of the crack, without leaving any physical evidence. The early stage is considered to be crack initiation and early crack propagation, in the presence of high humidity. The manufacturer was also of the opinion that 'Active Corrosion' would cause a reduction in the fatigue limit.

1.16.3.2 Previous testing on the effect of corrosion pits and fatigue life

During the 1990's the manufacturer undertook a study into the effect of corrosion pits on the damage tolerance of light alloy steel and a low alloy steel 30NCD16 in a high strength condition. The knowledge gained from this study was used to support the fatigue substantiation for a number of products used in several models of helicopter rotor systems.

46 The EC725 is the military variant of the EC225 LP.

The manufacturer's assessment was that the corrosion behaviour of 32CDV13 steel used in the EC225 shaft was similar to the 30NCD16 steel used in the AS332 shaft. The effect of a corrosion pit on the fatigue limit of coupons made from 30NCD16 steel was established by first using a salt spray to produce corrosion pits, to a depth of 0.18 mm. Fatigue tests were then carried out which established that the fatigue limit of these coupons was ± 300 MPa. This was a reduction (knock down) of 1.75 on the fatigue limit of coupons without corrosion pits of ± 525 MPa.

Care should be taken when considering the effect of corrosion pits on the fatigue life. A heavily corroded surface can generate a lower stress concentration factor than isolated pits such as were found on the shafts fitted to G-REDW and G-CHCN.

1.16.3.3 Susceptibility of 16NCD13 and 32CDV13 steel to corrosion

During this investigation the helicopter manufacturer carried out two laboratory tests on samples of 32CDV13 and 16NCD13 steel to determine if either was more susceptible to corrosion. In the first test salt spray was applied to the samples for up to 18 hours. In the second test the samples were placed in a climatic chamber at a temperature of 70°C and 80% relative humidity for a period of 8 hours. From the tests the manufacturer concluded that there was no significant difference between 32CDV13 and 16NCD13 steel in their susceptibility to corrosion. However, these were accelerated corrosion tests in a severe environment and it is not known if this would be the case in a mild corrosive environment in service.

1.16.3.4 Corrosion on the internal surface of the rotor mast

After the EC225 LP entered service, corrosion pits were discovered inside the main rotor mast with the most severe occurrences being found in helicopters operating in Asia. The MGB can vent through five 3 mm holes positioned in the top of the main rotor mast. It was, therefore, concluded that the corrosion in the mast occurred either as a result of:

- moist air being drawn into the mast as the gearbox cools after engine shutdown, or
- the condensation of moisture in the air as it vented out of the gearbox through the mast during operation.

As a result of this finding, the manufacturer modified the mast to incorporate a new surface finish. In addition they introduced, during the 1,200 hour or two year inspection, a check to detect and rectify any corrosion. The inside of the mast on G-CHCN had this new surface finish applied and after the accident there was no visual sign of corrosion in this area.

The manufacturer also considered whether corrosion might occur in the MGB. However, their assessment was that, unlike the inside of the mast, the components within the gearbox were protected from corrosion by the oil mist. This assessment was supported by the fact that there had been no reports of corrosion on the components within the MGBs on any of the Super Puma variants.

1.16.3.5 Effect of corrosion pits on the fatigue limit

As part of this investigation, the manufacturer carried out fatigue tests on a number of bevel gear vertical shafts manufactured from 32CDV13 steel to establish the fatigue limit and the effect of corrosion pits and humidity on this limit. During the tests the bevel wheel was clamped and an alternating load with a mean of zero (R=-1) was applied at the bottom of the shaft in order to subject it to a bending force (Figure 51).

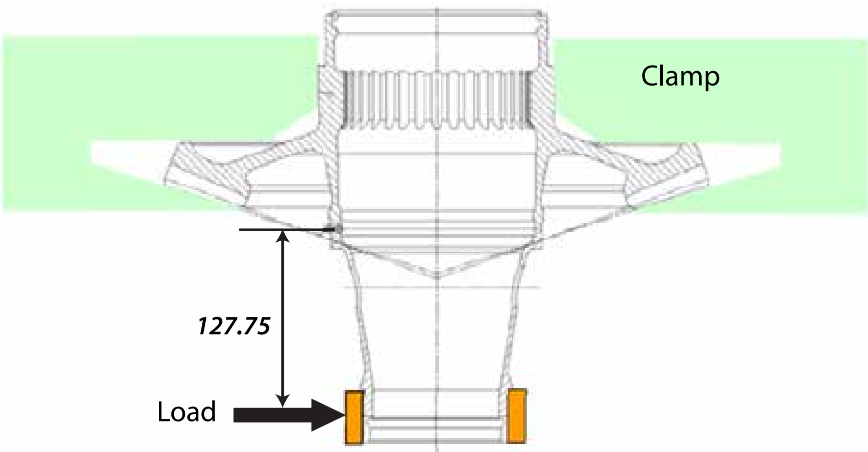


Figure 51

Clamping arrangement for the fatigue bending tests

The fatigue bending tests were not fully representative of the actual conditions in flight. The time for the corrosion to form was much shorter on the test shafts and it was not possible to simulate the accumulation of the red deposit (iron oxide) accurately or the effect of the oil and the centrifugal force. The frequency that the load was applied also differed. During the tests the load was applied between 2 Hz and 8 Hz, whereas in flight the shaft rotates at 40 Hz.

Details of the significant bending fatigue tests, and the method by which the fatigue limit was calculated, are in Appendix H and the results are summarised in Table 1. Miner's Rule[47] was used to establish the fatigue damage at each load level during the test. The fatigue limit was then calculated from the Wöhler curve equation:

$$\frac{S}{S_{\inf}} = 1 + \frac{A}{N^{\alpha}}$$

Where:

S = Applied cyclic stress amplitude
S_{inf} = Stress amplitude at the fatigue limit
N = Number of cycles to initiate a fatigue crack
A and α are properties of the material and are geometry dependent

For the tests where crack initiation did not start at a corrosion pit, the previously measured values of A = 1 and α = 0.0323 for 32CDV13 steel were used to establish the fatigue limit. Where the crack initiated from a corrosion pit, the manufacturer used a computer programme to fit the test results with the S-N curve established from the testing of corroded coupons made from 30NCD16 steel (section 1.16.3.2), which the manufacturer considered to be valid for 32CDV13. This curve-fitting exercise generated values of A= 0.137 and α = 0.768.

[47] Miner's rule is a method of approximating the cumulative fatigue damage that occurs at each level of load during the test cycle.

Shaft	Area	Humidity	Crack	Calculated Fatigue Limit	Comments
M787	Inner radius	Low	From 34 µm pit at inner radius.	± 358 MPa	Pre-corroded with salt spray.
M225	Inner radius	Low	From 60 µm pit at inner radius.	± 300 MPa	Pre-corroded with salt spray.
M175	Inner radius	Low	From 70 µm pit at inner radius.	± 282 MPa	Pre-corroded with salt spray. Profile of corrosion pit similar to G-CHCN.
M662 hole 1	Inner radius	High	From 30 µm pit at inner radius.	± 249 MPa	Hole 1 pre-corroded in salt water. No evidence of corrosion on surface of crack.
M662 hole 2	Inner radius	High	From 40 µm pit at inner radius.	± 224 MPa	Drip of water onto hole 2 at 85ºC. Evidence of corrosion fatigue on surface of crack.
M422	Inner radius	Low	From 20 µm pit at 29 mm hole.	± 453 MPa	No exposure to salt.
M422	4.2 mm hole	Low	From 5 µm pit at 4.2 mm hole.	± 629 MPa	4.2 mm hole exposure to salt atmosphere in climatic chamber prior to test.
M041	4.2 mm hole	High	From 40 µm pit at 4.2 mm hole.	± 292 MPa	Pre-corroded with salt spray. Wetted PTFE plug.
M391	29 mm hole	Low	From 23 µm pit at 29 mm hole.	± 470 MPa	No pits or cracks at 4.2 mm hole or inner radius. A fatigue limit of ±670 MPa was calculated for the weld.

Table 1

Summary of significant results from manufacturer's shaft bending fatigue tests

1.16.3.6 Effect of corrosion pit depth on fatigue limit

Data points from the bending fatigue tests carried out on shafts M787, M225, M175 and M662 are plotted in Figure 52 as a function of the calculated fatigue limit versus the corrosion pit depth. These data points are all for cracks that initiated from corrosion pits in the inner radius. Points M787, M225 and M175 were considered to be low humidity points and the data points from shaft M662 were considered to be high humidity points caused by water being dropped onto the surface at a temperature of 85ºC. The G-CHCN failure is plotted for comparison using the stress level produced at Takeoff Power (TOP).

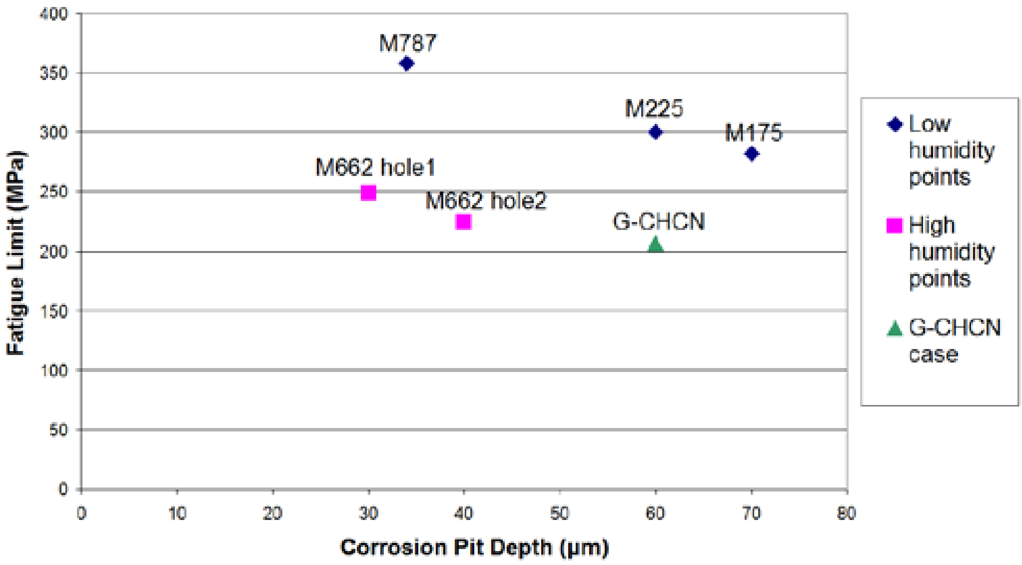


Figure 52

Effect of corrosion pit depth and humidity on the manufacturer's calculated fatigue limit for the inner radius

1.16.4 QinetiQ coupon testing

As part of the investigation, QinetiQ was tasked with investigating whether it was possible for cracks to propagate from small defects at representative operational stresses and to compare the fatigue performance of coupons[48] made from parent and electron beam welded material. A total of 14 coupons were produced from Class 2 32CDV13 steel. It should be noted that this test programme was not designed to be an exhaustive investigation into crack initiation and propagation; moreover, the number of coupons tested was not considered to be statistically significant to establish the full fatigue characteristics of the material.

[48] Coupons are samples of material that are manufactured to a prescribed size and shape.

The test programme compared the behaviour of parent and weld material control specimens containing a 4 mm diameter hole with similar specimens containing different features emanating from the hole. These features included short sharp fatigue cracks to represent a potential worst case stress raiser that could form during manufacture or service, and blunt scratches to represent machining marks or handling damage. The short fatigue cracks were introduced by drilling a 1 mm diameter hole and then subjecting the specimen to a three-point bend fatigue loading to initiate a crack. Once initiated, the cracks were grown in four-point bending with the length of the crack monitored by optical microscope and SEM inspections. Once the short fatigue crack had reached the required length, a 4 mm diameter non-countersunk hole was drilled and reamed, so the crack was located in the weld fusion line. The length and depth of the short crack was measured using a SEM to ensure it was in the target range of 50 to 100µm. The blunt notches were cut with a knife on either side of a 4 mm hole and were typically 0.01 to 0.02 mm long. Initiation of cracks from these blunt notches and their propagation was carried out using a four-point bending machine (Figure 53).

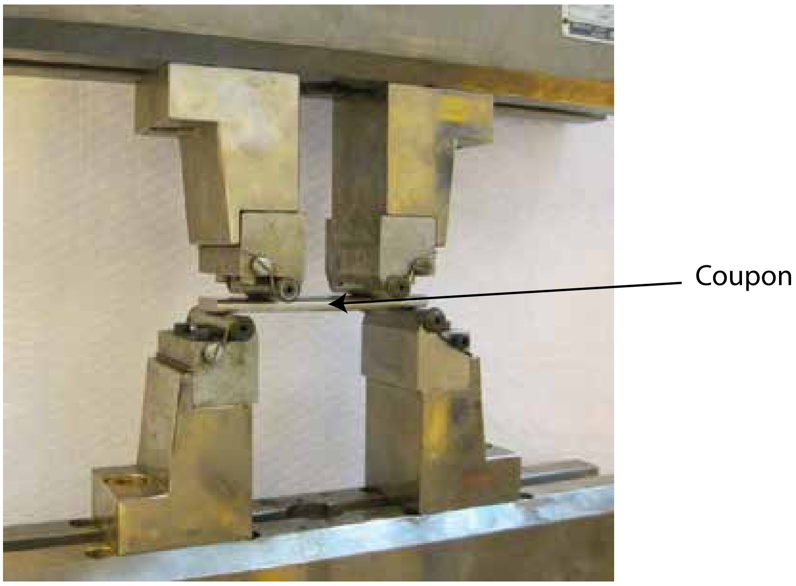


Figure 53

Four point bending machine

The tests indicated that on the electron beam welded coupons:

- The fatigue limit of the welded material was conservatively estimated to be ±600 MPa, when no defect or crack was present.

- Fatigue cracks can initiate from small features (burrs, notches) at low stress amplitude values below the estimated fatigue limit of ±600 MPa.

- Short fatigue cracks can propagate at stress amplitude values below the estimated fatigue limit of ±600 MPa and of similar magnitude to the stresses in the component in service.

- The weld microstructure influences crack growth direction. The weld centre-line and fusion line, even after a post-weld heat treatment, are particularly susceptible locations depending on the direction of welding.

On the coupons made from the parent material, the tests indicated that:

- Fatigue cracks can initiate from small features only at high stress amplitude values above the estimated fatigue limit of ±600 MPa.

- Short fatigue cracks can propagate at stress amplitude values below the estimated fatigue limit of ±600 MPa and of similar magnitude to the stress in the shaft in service.

In summary, the tests indicated that the weld region may be more susceptible to fatigue crack initiation than the parent material and that crack growth in some locations in the weld region can be very rapid after the crack has initiated. QinetiQ concluded that small surface defects in the shaft must be avoided as they could lead to the initiation of fatigue cracks, and the presence of the weld in the shaft should be considered during any component analysis.

1.16.5 Helicopter manufacturer's Finite Element Model (FEM)

1.16.5.1 Original FEM

A FEM was developed to support the fatigue substantiation submitted during the original certification process for the EC225 shaft. The model established that the stresses in the area of the weld were predominantly generated by loads from the bevel gear (99%); the torque generated by the oil pumps contributed less than 1% of the loads. The model included an allowance for the internal clearance of the bearings. Flexing of the gearbox casing was not taken into consideration.

The FEM model identified three critical areas on the part: the upper diameter, the 4.2 mm hole in the weld and the 29 mm lubrication holes. The model assumed that the 4.2 mm hole was aligned with one of the 29 mm holes.

Using the following data, the model established that the maximum, worst case, alternating stress[49] in the 4.2 mm hole was ±83 MPa. The maximum alternating stress in the 29 mm hole was ±71 MPa.

Standard smooth specimen fatigue limit for 32CDV13 steel.	± 600 MPa
Fatigue limit used -allowing for Ra of 3.2	± 480 MPa
Torque – high cycle fatigue	9,175 ± 917 Nm
Torque – low cycle fatigue	5,043 ± 504 Nm

1.16.5.2 FEM revised during investigation

Following the accident involving G-REDW, the manufacturer reviewed the FEM model and corrected the boundary conditions on the upper roller bearing that supports the bevel gear vertical shaft. The effect of these changes was to reduce the stiffness of the bearing, which increased the amount the shaft could flex.

There were also other refinements of the FEM that produced a more accurate prediction of the stress in the area of the weld. These included: reduction of the size of the mesh used in the computer model and the effect of the relative position of the 4.2 mm and 29 mm holes. The maximum stress in the 4.2 mm hole occurs when the relative position between the 4.2 mm hole and its nearest 29 mm lubrication hole is 40º.

The FEM model was used to examine the effect of shaft misalignment on the stresses in the area of the weld and it was established that there was negligible effect on the stress up to a misalignment of 0.5 mm. Beyond 0.5 mm the manufacturer would expect to see damage to the bearings that support the shaft. No such damage was seen on the bearings fitted to either G-REDW or G-CHCN.

The model predicted that the maximum stress at the 4.2 mm hole would be ±314 MPa at TOPtran[50]. This was 3.8 times higher than the original FEM prediction. The relative position between the holes on G-REDW was 38º and therefore the stress was effectively the same as the maximum stress predicted by the model.

[49] The different stresses are defined in Appendix G.
[50] TOPtran is the maximum power, which is defined in Appendix D.

1.16.6 Dynamic tests

A dynamic test was carried out by fitting an EC225 shaft into an EC225 LP MGB which was then run in the manufacturer's test cell (Figure 54).

Figure 54

Shaft undergoing dynamic test.

The gearbox was mounted in a similar manner as on the helicopter. However, while the strut (lift) bars were at the same angle, they were slightly longer. The mounting adaptor used at the bottom of the gearbox was also slightly stiffer than the 'barbeque plate[51]' used on the helicopter. The engines were represented by two electrically driven input shafts; no power was taken off the gearbox for either the tail rotor or auxiliary components. Consequently, for the same representative engine output power, the load on the bevel gear vertical shaft would be slightly higher than on the helicopter. The manufacturer stated that this was taken into account by reducing the engine torque during the test on the rig to achieve the same values as experienced on the helicopter. No account was taken for flexing of the rotor mast.

The maximum stress at TOPtran[52], calculated from the strain gauge data, was ±298 MPa, which was within 5% of that predicted by the refined FEM model. This was 3.6 times greater than that calculated in the initial FEM used in the fatigue substantiation for the certification of the MGB.

51 The barbeque plate is a flexible mounting plate that reacts the torque loads produced by the MGB.
52 TOPtran is the transient takeoff power and is explained in Appendix D.

1.16.7 Ground vibration testing

The investigation considered whether the bevel gear vertical shaft experienced unexpectedly high loads from a resonant frequency. Tests were carried out by striking an EC225 shaft, instrumented with accelerometers, with a tool to determine the natural frequencies of the shaft. The tests were repeated on an AS332 shaft and there were no significant differences. An FEM model was also used to calculate possible natural frequencies that might excite high stress in the areas of the weld and inner radius – this revealed potential high oscillations at 1,897 Hz and 2,061 Hz. This then led to a series of ground rig tests using an instrumented MGB mounted to a rotating bench. Tests were carried out with both a new MGB and the MGB from G-REDW. Accelerometers were mounted on the inside and outside of the MGB to measure the vibration levels, and an array of strain gauges was installed on the bevel shaft to measure the strain from which the stresses could be calculated (Figure 55).

The tests were carried out by varying the rpm between 250 and 280 rpm in small increments and also varying the torque up to TOPtran. The rig was also used to vary the mast bending moment. However, none of the ground tests revealed any significant dynamic responses.

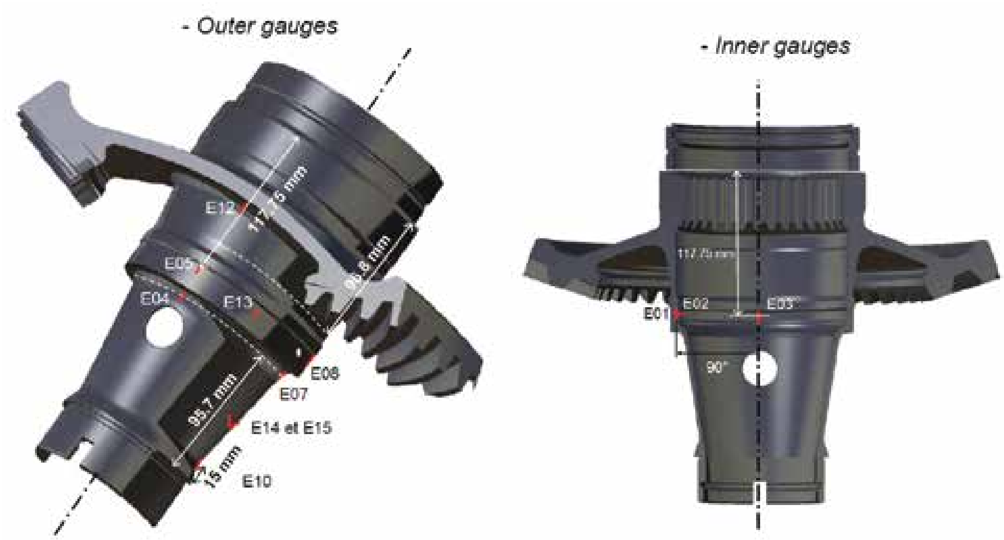


Figure 55

Strain gauge locations on shaft M8375

1.16.8 Flight test vibration and loads

The vibration tests carried out on the ground rig were repeated in flight to see if the aerodynamic and mast bending loads changed the results. The helicopter was modified to enable the rpm to be varied in small increments. The following test conditions were examined while varying the rpm: engine start-ups, ground-taxi while varying cyclic and collective position, hover, climbs, cruise at varying speeds up to V_{NE}, manoeuvrability including left, right and rearwards flight at up to 25 and 30 kt, 45° banked turns, single-engine flight and autorotations. Due to the weather conditions it was not possible to perform taxi tests in strong crosswinds, so landings on slopes were performed to simulate the large sideways cyclic deflections that would be required in strong crosswinds. None of the flight tests revealed any significant dynamic responses. The maximum loads were achieved at maximum torque.

The measurements obtained from the strain gauges on the shaft were used to refine the FEM model. Stress can be calculated from the strain gauge data by multiplying the strain by the Young's Modulus, which is a property of the material. The Young's Modulus of the weld is different from the parent material; this was taken into account in the calculations. As some of the strain gauges overlapped both the weld and parent materials the accuracy of the stress calculation would have been reduced. Also, in areas where there was a high stress gradient, the strain gauges only provided the average stress across the length of the gauge which was 4 mm. Therefore, FEM analysis was still required to determine the maximum stress in these areas.

A comparison of the FEM predicted stresses, and the ground (rotating) rig test stresses and the flight test stresses are shown in Table 2. The numbers E01 to E13 represent the different strain gauge positions and the blank boxes indicate that the respective strain gauge malfunctioned, producing no data.

Strain gauge	**FEM** (MPa)	**Ground Test 1**[1] (MPa)	**Ground Test 2**[2] (MPa)	**Flight Test**[3] (MPa)
E01	99	97	90	
E02	37	32	29	27
E03	80	72		
E04	48	50	48	53
E07	118	118	113	121
E08	45	32	29	
E10	29	23		29
E12	59		55	58
E13	45			37

1 Eurocopter rotating test on MGB: No.1. Test date July 2012. Reference DEL 1273.
2 Eurocopter rotating test on MGB: No.2. Test date November 2012. Reference DEL 1294.
3 Eurocopter flight test on MGB: No.2. Test date November 2012. Reference DEV 2327.

Table 2
Stress comparison between FEM, ground rig test and flight test

1.16.9 Crack propagation

The manufacturer carried out dynamic testing of seven shafts to determine the rate that a circumferential crack would propagate at a power setting of MCP and MCP-15%. Decreasing the engine power by 15% should decrease the rate the crack grows by 33%. (Appendix G). Cracks were initiated at the 4 mm hole, weld and inner radius and the shafts were then subjected to the engine power spectrum similar to that experienced on G-REDW and G-CHCN. At the end of the tests the manufacturer carried out beachmark counting to determine the rate of crack propagation. A similar test was carried out, in flight, on a shaft fitted to an EC225 LP. As the tests were carried out on shafts and not coupons, the results included the effect of residual stress[53] and the position of the 29 mm lubrication hole, on the local stress at the 4.2 mm hole and inner radius. The results of the test are at Figure 56.

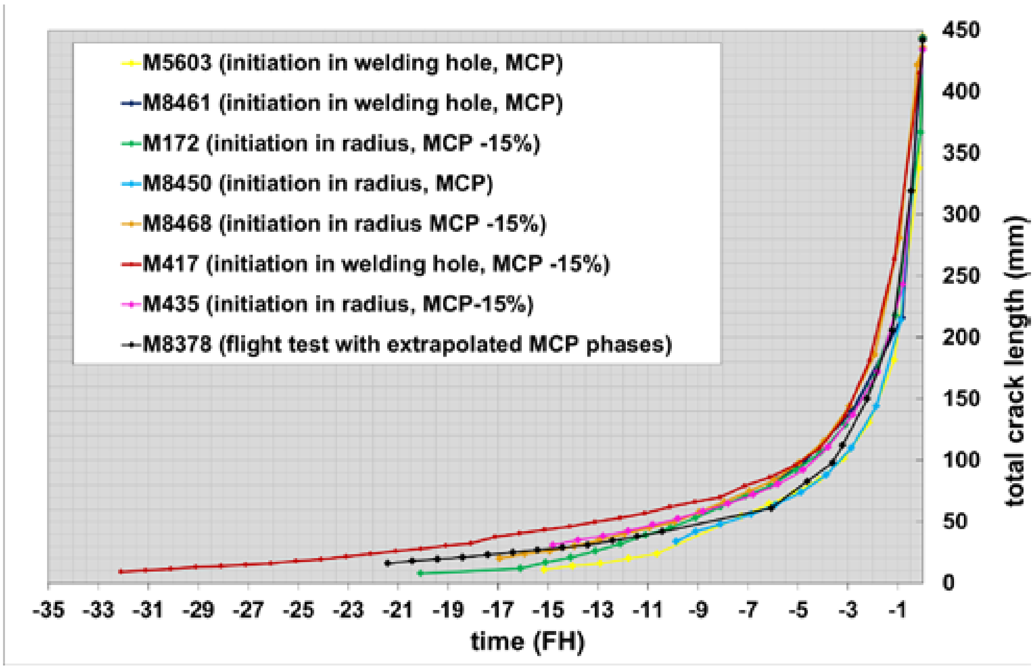


Figure 56

Results of manufacturer's crack propagation tests

These tests also explored the response of the HUMS MOD-45 indicator as the crack grew in length. The results of these tests are at Figure 57 and show that HUMS indicator MOD-45 exceeded the red threshold of 0.2 g rms when the crack length was between 87 mm and 100 mm.

[53] Refer to Appendix F.

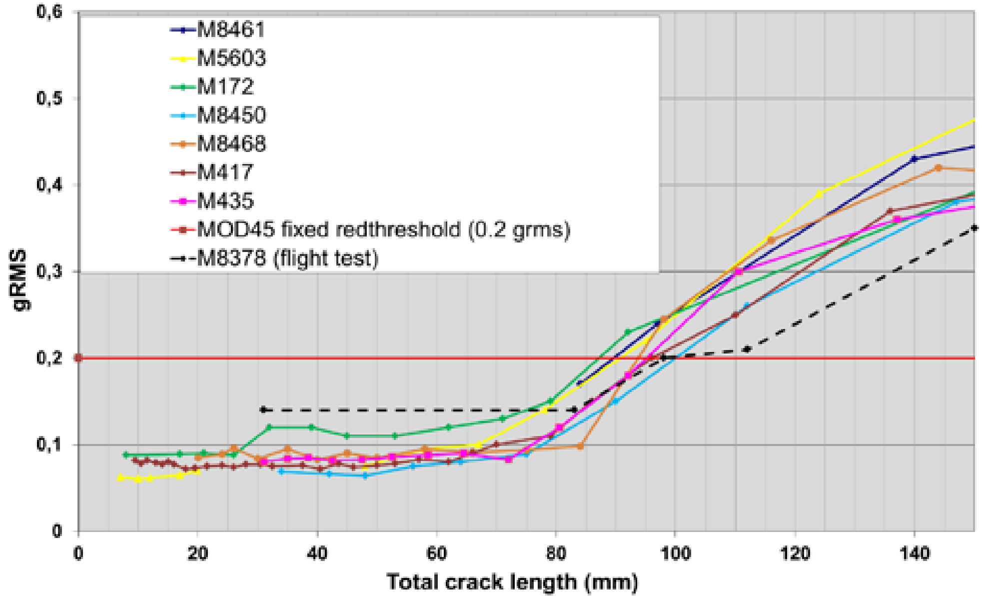


Figure 57

Results of manufacturer's crack propagation tests

1.16.10 Fatigue crack growth analysis

As part of this investigation, Cranfield University was contracted[54] by the AAIB to use a 'fracture mechanic model' to determine if the combination of service loading, material properties and corrosion pits could explain the failure of the bevel gear vertical shafts fitted to G-REDW and G-CHCN. A definition of the engine power settings and explanation of fracture mechanics is at Appendix D and G.

The method employed was to calculate constant amplitude Stress-Life (S-N) curves to grow a fatigue crack from a range of starting defect crack sizes, and included the geometry of the corrosion pits on G-REDW and G-CHCN, in the location where the fatigue cracks initiated. A number of different defect depths and shapes were used as starting cracks. The smallest of these were representative of the actual pits which initiated the shaft failure cracks. The S-N curves were then compared with the stress levels at TOPtran and MCP to see if the cracks would grow. A number of scenarios of the model were run using a combination of the following data:

- Long crack growth data alone. No residual stresses.
- A combination of available short and long crack growth data. No residual stresses.
- Long crack growth data with residual stresses.

54 Fatigue investigation of corrosion pitted gearbox shafts: Review of investigations and calculations relating to the failures of EC225 Gearbox shafts on G-REDW and G-CHCN. September 2013, Professor P E Irving, Cranfield University.G-REDWG-CHCN.

Crack growth predictions were performed using the 'AGROW' software programme developed by the US Airforce. Because the spectrum of service stress cycle amplitudes was not available to use as an input to the programme, a constant amplitude fatigue crack growth analysis was carried out instead. The model used the geometry of the corrosion pits and stresses / flight loads provided by the manufacturer, who also supplied long crack data for 32CDV13 steel; there was no short crack data available for this material.

As there were no published solutions for the calculation of the stress intensity of long cracks propagating circumferentially around a tube subject to bending with a small torsional component, the solutions for cracks in tension and bending in a flat plate were used instead. The accuracy of this approach was considered to be acceptable for the early part of the failure process. All the different scenarios were terminated at a through crack length of 20 mm; thus the later part of the service failure, where a crack longer than 20 mm propagated around the circumference of the shaft, was not modelled. However, the beachmark analysis showed that from a crack length of 20 mm it took less than 20 flying hours before the shaft failed.

The results of the model showed that by using the long crack data only, a crack would grow from a defect 1 mm deep only if the power setting was at TOPtran. However, when the long crack data was combined with either the short crack data, or residual stress data, cracks would grow from a 60 μm deep defect located in the 4.2 mm hole (G-REDW) at a power setting of TOPtran and from the inner radius (G-CHCN) at a power setting of MCP.

The following diagrams show the results of the modelling. Figures 58 and 59 shows the results using long and short crack data; Figures 60 and 61 shows the results using long crack data and residual stress. The maximum tensile stress on the vertical axis is the remote stress away from a feature such as the 4.2 mm hole. The effect of the stress concentration from these features is included in the S-N curve. If the S-N curve goes below the horizontal line representing the stress at TOPtran and MCP power settings, then a crack would grow from a 60 μm defect.

The remote stress at TOPtran and MCP power settings for the G-REDW and G-CHCN conditions are as follows:

	4.2 mm hole (G-REDW)	Inner radius (G-CHCN)
MCP	98 MPa	185 MPa
TOPtran	120 MPa	227 MPa

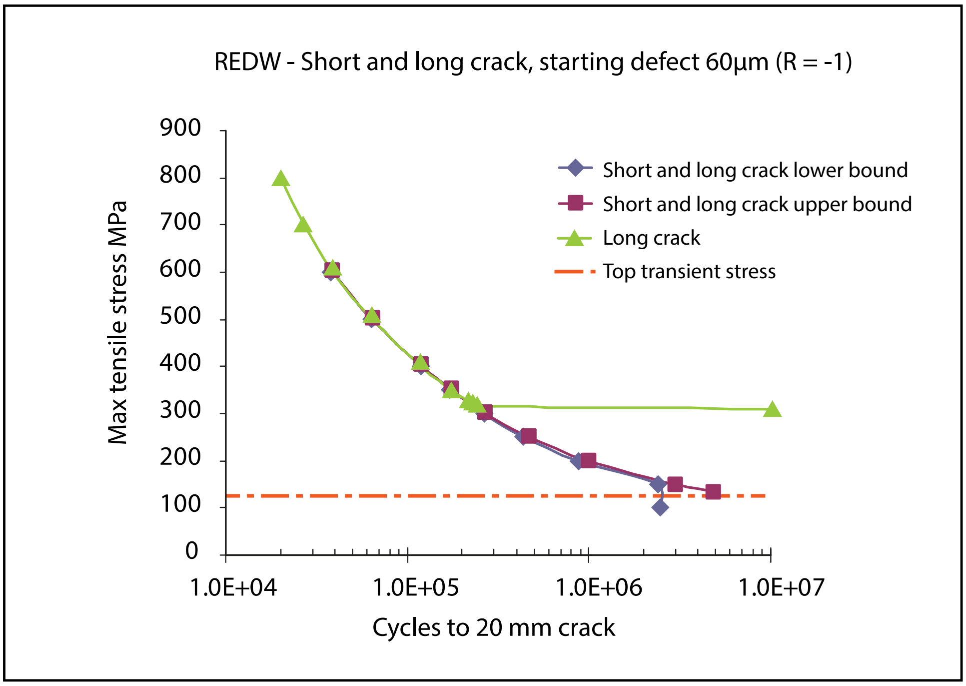


Figure 58

Results of Cranfield model using short and long crack data for the G-REDW failure

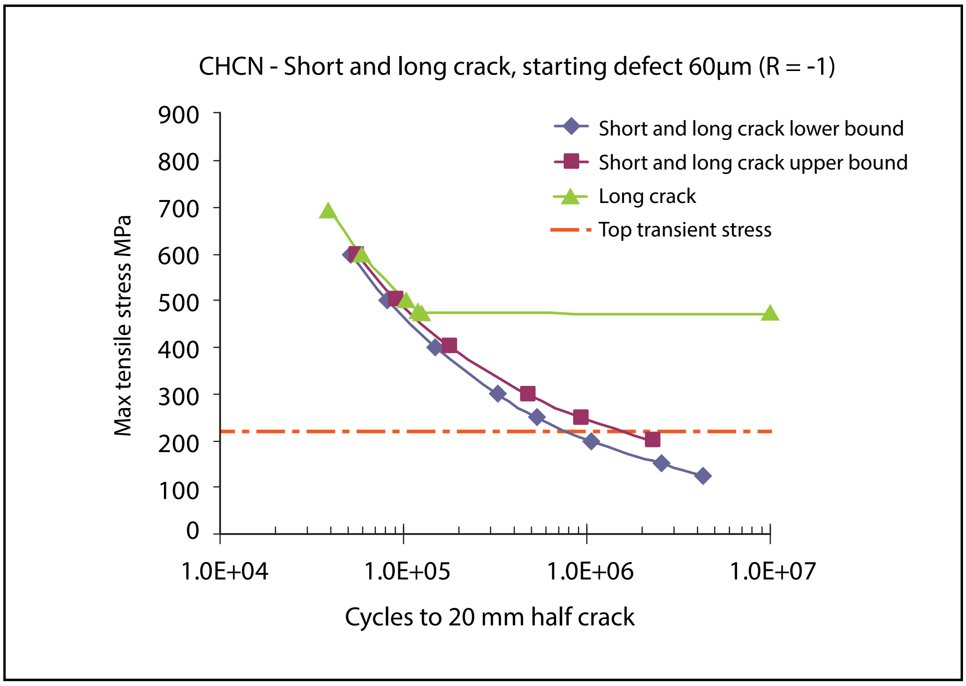


Figure 59

Results of Cranfield model using short and long crack data for the G-CHCN failure

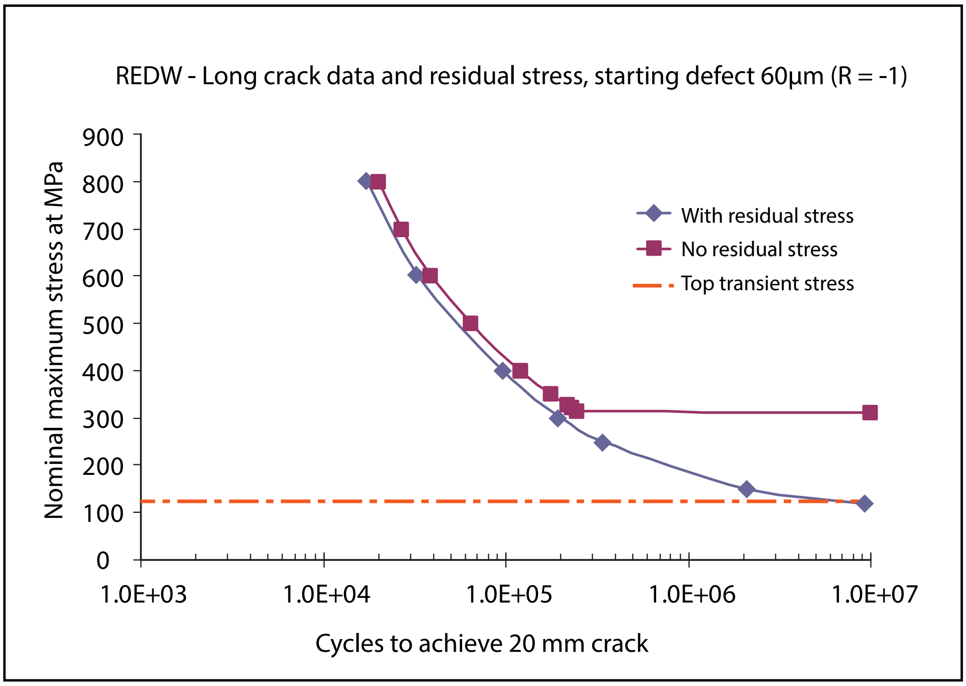


Figure 60

Results of Cranfield model using long crack data and residual stress for the G-REDW failure

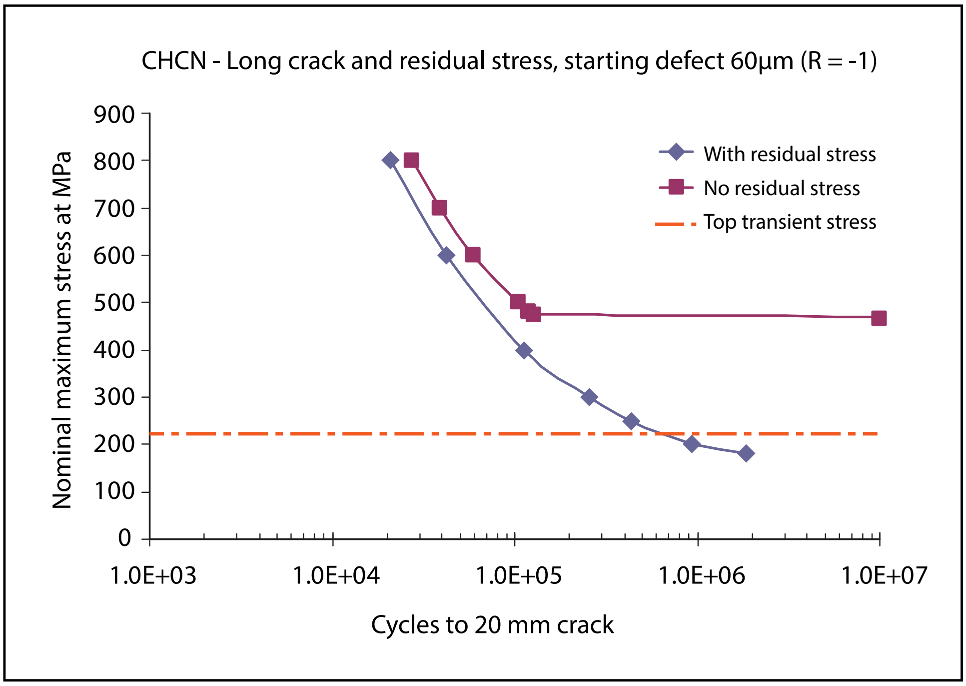


Figure 61

Results of Cranfield model using long crack data and residual stress for the G-CHCN failure

1.16.11 Emergency lubrication system

The components of the emergency lubrication systems on both helicopters were inspected in detail and tested. Tests were also carried out to measure the bleed-air pressure on several engines with different modification states.

1.16.11.1 Bleed-air and Hydrosafe 620 pressure switches

The two pressure switches from both helicopters were tested. All four switches conformed to their respective acceptance tests, with activation thresholds (p_{on}) in the range of 0.61 to 0.68 bar (relative to ambient).

1.16.11.2 Emergency Lubrication System - Hydrosafe 620

Both Hydrosafe 620 pumps were tested and operated to specification. Thus there was evidence that the pumps were operating normally from the time the system was activated until the helicopter ditched.

Bench tests were carried out on a MGB with a failed bevel gear vertical shaft. The Hydrosafe 620 and bleed-air supplies were activated and temperatures were measured at the Hydrosafe 620 pressure switch housing and MGB casing. It was found that after about 10 minutes the Hydrosafe 620 pressure had started to decrease to around 0.7 bar relative. This value is higher than the threshold for the pressure switches fitted to the accident helicopter, but lower than the maximum specification for these components.

1.16.11.3 Printed Circuit Board (PCB)

The PCBs, which control and monitor the emergency lubrication system, were functionally tested and operated in accordance with the factory inspection test. The 30-second time delay for the PCBs from G-REDW and G-CHCN, during which a failure warning is inhibited, were measured and were consistent with the period of time between the crew's activation of the system and the illumination of the MGB EMLUB caption, derived from the CVFDR.

1.16.11.4 Emergency Lubrication System - Engine tests

The engine and helicopter manufacturers tested the bleed-air output from several Turbomeca Makila 2A and Makila 2A1 engines. These included bench tests of the engines from G-REDW and G-CHCN, ground tests on in-service helicopters, and flight tests by the helicopter manufacturer. These tests revealed that the bleed-air pressure depends on:

- the altitude,
- power setting,
- engine modification state.

Figure 62 is an indicative graph. For a given pressure altitude and power setting, it was found that the bleed-air pressure (red band) is always lower than that used in the design and certification of the system (blue line).

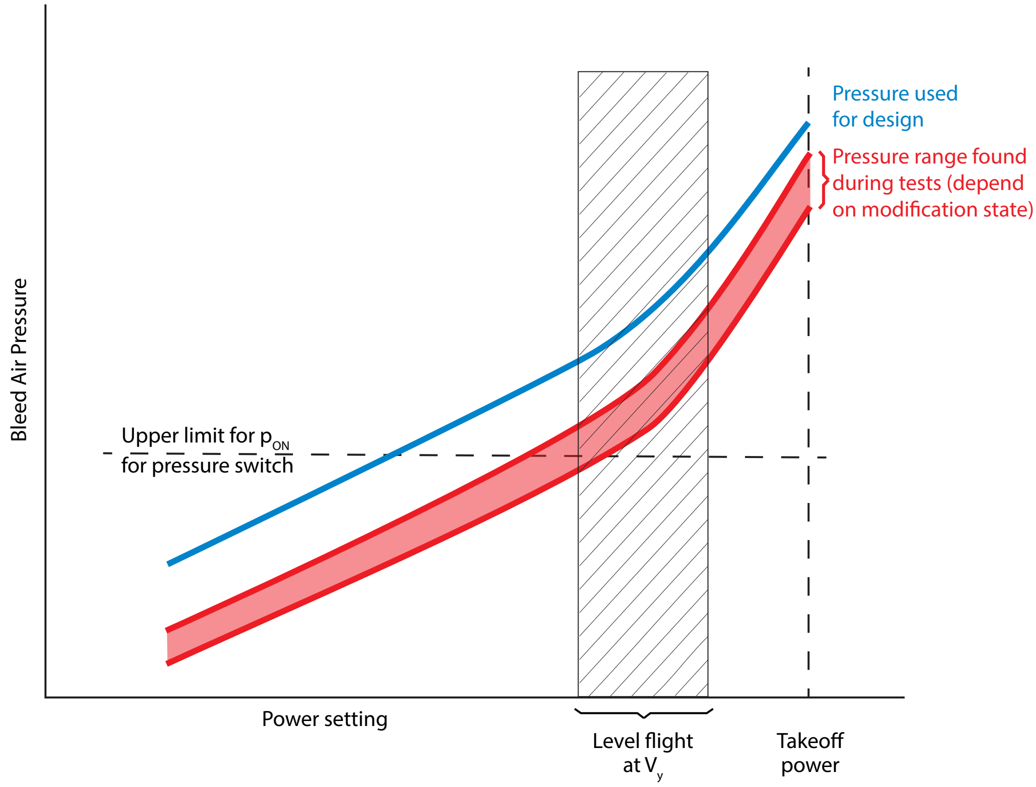


Figure 62

Variation of bleed-air pressure with engine power setting

1.16.11.5 Bleed-air system

The components of the bleed-air systems from the accident helicopter were tested along with similar tests carried out on new components, in particular to understand the pressure losses in the system. This was carried out on a ground rig, with and without the Hydrosafe 620 supply operating. From these and the engine tests, it was concluded that a bleed-air pressure switch with a p_{on} at the top end of the specified tolerance (1 bar) could generate an MGB EMLUB caption, even if all the parts of the emergency lubrication system were operating within their design specifications.

1.16.11.6 Emergency Lubrication System wiring

The pressure switches have three output pins which are electrically connected to the PCB. The helicopter manufacturer's original specification for the pressure switches was that the wire from Pin 3 was to be common. However, the selected supplier for the pressure switches delivered the switches with Pin No 1 as common; this change was accepted by the helicopter manufacturer and the wiring on the helicopter was changed accordingly. The original specification for the pressure switches was not changed to reflect the different pin positions.

Owing to obsolescence, the helicopter manufacturer issued a specification for new pressure switches, requiring them to be interchangeable with the original switches. It included the original specification for the internal wiring of the pins. Therefore, the replacement pressure switches were developed with Pin 3 as the common pin. The helicopter wiring, however, remained configured for a common Pin 1. The new pressure switches, manufactured by a different supplier, were fitted to helicopters as part of a modification in 2010 (MOD 0752520). The wiring and internal schematic for the switches before and after MOD 0752520 is shown in Figure 63. The schematic is valid for both the Hydrosafe 620 and bleed-air switches.

The terms 'p_{on}' 'p_{off}' and 'Common' in Figure 63 reflect the helicopter wiring identification. The figure illustrates that changing the internal architecture to Pin 3 as 'common' on the replacement switch meant it now fed the p_{off} signal wiring on the helicopter.

This means that the MGB EMLUB caption, for helicopters with MOD 0752520, will illuminate after a 30-second delay following activation of the emergency lubrication system, if there is:

- A pressure above the switch threshold (which will result in an erroneous signal being detected by the PCB).
- A pressure below the switch threshold which will result in detection of a low pressure condition.
- An erroneous signal to the PCB for other reasons.

In summary, the MGB EMLUB caption will illuminate for any of the three possible states - high pressure, low pressure or an erroneous signal - when the system is activated. These are the only three possible states for the system and hence for all helicopters with MOD 0752520 the MGB EMLUB caption will illuminate after the 30-second delay following system activation. Both G-REDW and G-CHCN had MOD 0752520 embodied.

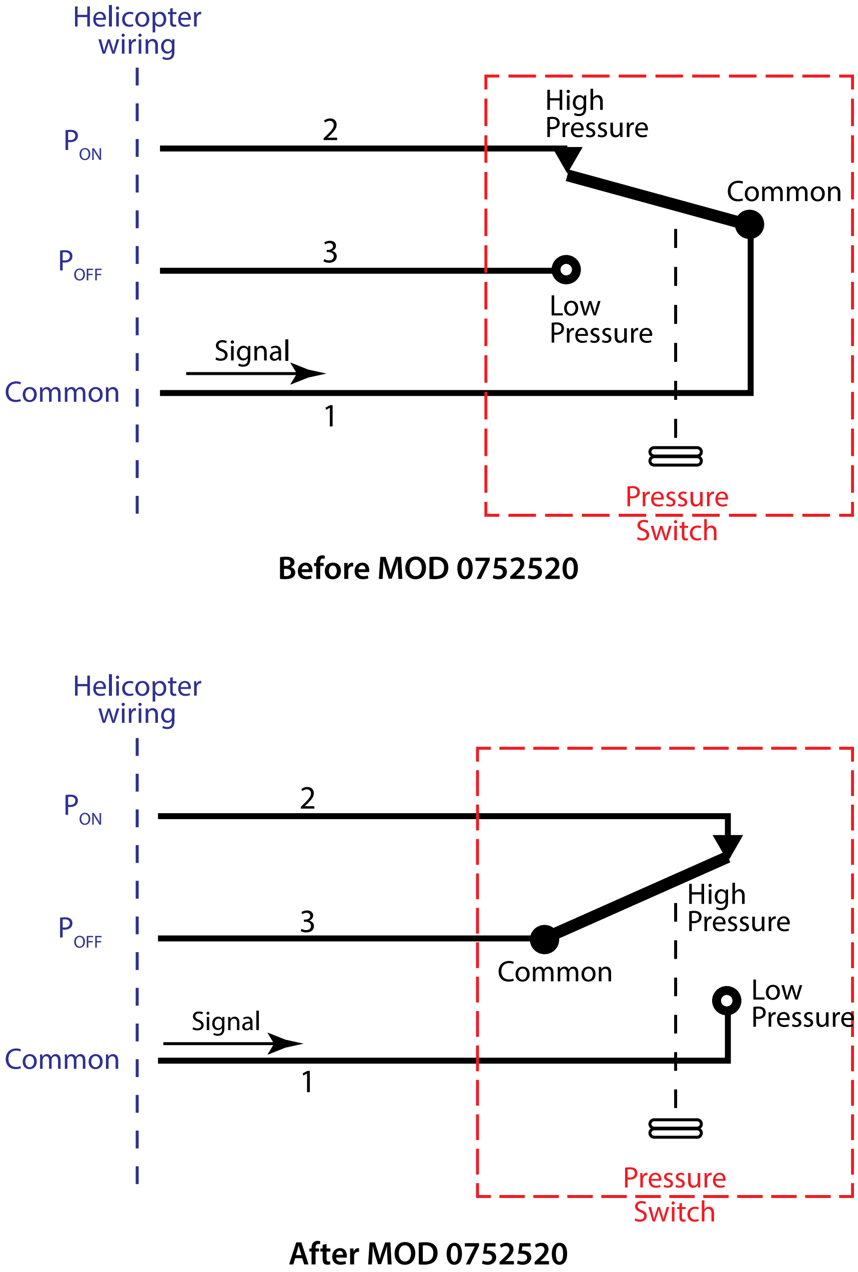


Figure 63

Schematic of wiring and pressure switches pre and post MOD 0752520

1.17 Organisational and management information

Not applicable to this investigation.

1.18 Additional information

1.18.1 Case hardening

Case hardening is the process of hardening the surface of a metal. Carburising was used to surface harden the bevel gear part of the AS332 shaft. However, the requirement for the EC225 LP MGB to be capable of running dry for 30 minutes meant that the bevel gear on the EC225 LP needed to be able to operate at a higher temperature than on the AS332 L2. This higher temperature would have compromised the effect of the carburisation and therefore it was necessary to use a surface hardening process called nitriding.

The change from carburising to nitriding required a material change. Therefore the 16NCD13 steel used in the AS332 shaft was replaced with 32CDV13 steel in the EC225 shaft.

The change in surface hardening was not a factor in the accidents involving G-REDW and G-CHCN because the part of the shafts where the fatigue cracks occurred was not subjected to either the carburising or nitriding processes.

1.18.2 Certification requirements

The EASA Type-Certificate Data Sheet, No. R.002, states that the EC225 LP was certified to JAR 29 change 1 and that the helicopter was *'designed as a derivative product of the former type certified AS332L2'*. The technical investigation for the compliance was undertaken by the Direction générale de l'aviation civile (DGAC).

The original certification basis for the fatigue tolerance evaluation of the AS332 L2 was to Federal Aviation Requirements (FAR) 29.571, at Amendment 24. For the EC225 LP, the certification basis was FAR 29.571, at Amendment 28 and JAR 29.571, Issue 1. Both these documents introduced flaw tolerance requirements for Primary Structural Elements (PSE).

The manufacturer requested,[55] through the DGAC, that for PSE on the EC225 LP that had not changed significantly from the AS332 L2, that they be permitted to use information derived from the original AS332 L2 certification tests. For those PSE that had changed, which included the bevel gear vertical shaft, a request was made to certify these items against FAR 29.571, at Amendment 28.

The manufacturer[56] subsequently requested, through the DGAC, that they be permitted to use Notice of Proposed Rule Making (NPRM) 29.571[57]. This

[55] Request made through Certification Review Item (CRI), No 03.
[56] Eurocopter request made using CRI No 04.
[57] The text of the relevant sections of the NPRM are at Appendix I.

proposed a clarification and amendment to the flaw tolerance evaluation in FAR 29.571. The DGAC authorised the manufacturer to use this NPRM, which has since been incorporated[58] into FAR 29.571 and CS 29.571.

1.18.3 Fatigue life of AS332 L2 and EC225 LP bevel gear vertical shafts

Both the AS332 L2 and EC225 LP bevel gear vertical shafts were designed on the basis that the operating stresses are always below the parent material safe fatigue limit. Therefore, if the surface was adequate, the shaft should have an infinite fatigue life.

As the shaft rotates, each point on its circumference is subjected to an alternating stress with the largest tensile stress occurring on the inner surface of the shaft. The R ratio[59] is approximately -0.9.

1.18.4 Minimum required safety factor

The Failure Mode Effect Causal Analysis (FMECA) for the MGB identified the failure of the bevel gear vertical shaft as being ‘catastrophic’[60] to ‘hazardous’[61] (the FMECA refers to the shaft as a web). The manufacturer’s procedures[62] classified the shaft as a ‘Critical Part[63]’, which for a Safe Life Fatigue required it to be designed with a safety factor of three by analysis, or by test using a safety factor[64] obtained from a proprietary formula[65]. This formula takes into account statistical factors such as the scatter of the fatigue limit and the number of tests carried out. In comparison, a ‘non-Critical Part’ with a failure classed as ‘hazardous’ requires a safety factor of two by test, or three by analysis.

The regulator does not specify the minimum safety factor, but instead requires the manufacturer to demonstrate that ‘catastrophic’ or ‘hazardous’ failure is avoided. This approach requires the regulator to review the methodology used by the manufacturer and establish how they assess and account for all the various parameters that make up the fatigue life substantiation.

58 The NPRM was incorporated into FAR 29.57 by Amendment 55 in December 2011 and into CS 29.571 by amendment 3.
59 R Ratio is explained in Appendix G.
60 ‘Catastrophic’ is defined as a failure condition which would result in multiple fatalities, usually with the loss of the aircraft.
61 ‘Hazardous’ is defined as a failure condition that would reduce the capability of the aircraft to the extent that there would be a large reduction in safety factors or functional capability, and serious or fatal injury to a relatively small number of occupants other than the flight crew.
62 Detailed in Eurocopter Document CAL 08 023.
63 A Critical Part is a part the failure of which could have a catastrophic effect and for which critical characteristics have been identified, which must be controlled to ensure the required level of integrity.
64 Safety factor in this context is the ratio of the fatigue limit to the maximum tensile stress.
65 Eurocopter document CAL 08022.

1.18.5 Safety factor for the AS332 bevel gear vertical shaft

The AS332 shaft was qualified by the fatigue testing of four shafts in rotative bending[66]. During the tests three of the four shafts experienced fatigue cracks in the weld which allowed the manufacturer to establish the mean fatigue limit, in terms of the bending moment, to be ± 8,535 Nm.

As the fatigue tests were undertaken by testing, the required safety factor was based on the manufacturer's proprietary formula, which established that the safety factor for an infinite fatigue life had to be greater than 1.85. This required the shaft to be able to withstand a maximum bending moment of ± 4,613 Nm (8,535 Nm / 1.85) without suffering from a fatigue crack. The bending moment at TOPtran on the AS332 L2 was established to be ± 2,845 Nm; therefore the actual safety factor would have been 3 ((8,535 Nm / 2,845 Nm) = 3).

While the analysis of the shaft identified the 4.2 mm hole in the weld as a Critical Area, the tests did not, and did not need to, identify the stress in the shaft because the substantiation was demonstrated by the loads and not the stress levels. As the safety factor was demonstrated by testing it included any effect of residual stress in the shaft. The testing and analysis did not consider the effect of corrosion on the fatigue life, as it was expected that the oil rich environment inside the shaft would prevent corrosion.

As part of this investigation the manufacturer produced a FEM which established that the maximum stress in the 4.2 mm hole at TOPtran was ± 314 MPa for the EC225 LP shaft and ± 342 MPa for the AS332 L2 shaft.

1.18.6 Certification of the EC225 bevel gear vertical shaft

The fatigue analysis of the EC225 shaft was carried out from a combination of experience gained from the testing of the AS332 shaft, a FEM and rotative bending testing of welded coupons.

There was no documentation to show that the manufacturer considered the effect of residual stresses. The manufacturer stated that residual stresses were considered and they believed that there would have been a compressive residual stress on the surface and no significant tensile residual stress within the component.

As the stress levels were similar on both the AS332 and EC225 shafts, the effect of residual stress was not considered to be significant during the design of the EC225 shaft. However, the increased thickness of the EC225 shaft, in the area of the weld, meant that the residual stresses in this shaft could be different from those in the AS332 shaft.

[66] The fatigue substantiation is detailed in document 332A.05.3286 issue C.

The manufacturer's experience was that corrosion would not occur in the oil mist environment within the MGB and therefore no allowance for corrosion was made when considering the fatigue life of the EC225 shaft.

1.18.7 Stress levels in the EC225 LP bevel gear vertical shaft

1.18.7.1 Introduction

The bevel gear vertical shaft is subject to rotating bending and the stresses vary through the wall thickness with the largest stresses occurring on the inner surface. On G-REDW and G-CHCN the fatigue cracks originated near the inner surface of the shaft.

The stresses presented in this report all refer to the values on the inner surface of the shaft.

1.18.7.2 Original EC225 certification submission

In considering High Cycle Fatigue, the manufacturer identified in the original[67] fatigue substantiation document three critical areas on the shaft fitted to the EC225 LP. One of these areas was the 4.2 mm hole in the weld that was identified as Critical Area 2. The inner radius was not identified as a critical area.

In establishing the maximum stress, the original FEM used the TOPtran power setting multiplied by a factor of 1.1 to account for fluctuations in torque. The TOPtran torque of 9,170 Nm ± 10% was established from flight test data collected during the certification process. This value is similar to the 9,807 Nm ± 5%, generated by the manufacturer's flight load department. The FEM established that the maximum and minimum stress for Critical Area 2 was 82.2 MPa and -68.5 MPa, which gave an alternating[68] stress (σ_a) of ±75.4 MPa. Allowing for 10% torque fluctuation the maximum alternating stress was ±83 MPa. A description of the calculation of the maximum stress is given at Appendix G.

1.18.7.3 Maximum stress obtained from testing and analysis

Following the accidents involving G-REDW and G-CHCN, the manufacturer obtained data from flight tests, dynamic rig tests and the FEM, which allowed them to refine further the maximum stresses in the shaft.

It was established that the maximum stress in the weld occurred at the 4.2 mm hole, in the centre of the melt. However, the fatigue crack on G-REDW initiated close to the fusion line where the FEM showed that maximum stress at TOP power setting was 56 MPa lower than the stress at the centre of the weld.

67 EC 225 LP Fatigue Substantiation, Document 332A 05 3205.01, Issue A.

68 $\sigma_{max} = (\sigma_{alt\ max} – \sigma_{alt\ min})/2$

The maximum stresses obtained during this investigation are listed in Table 3 and include a 10% factor to account for torque fluctuations. The mean stresses were assessed as being close to zero.

Area	Power setting	Stress
Maximum stress at 4.2 mm hole (inner countersink).	TOPtran	± 315 MPa
	TOP	± 286 MPa
	MCP	± 256 MPa
Location of crack initiation on inner countersink, G-REDW failure condition.	TOPtran	± 252 MPa
	TOP	± 230 MPa
	MCP	± 206 MPa
Maximum stress at inner radius, G-CHCN failure condition. (Values taken from Eurocopter Document 332A056031, Issue A.)	TOPtran	± 227 MPa
	TOP	± 207 MPa
	MCP	± 185 MPa

Table 3

Stresses in 4 mm hole and inner radius

1.18.7.4 Residual stress

During this investigation the helicopter manufacturer contracted two separate companies to measure the residual stresses in a sample of bevel gear vertical shafts using x-ray diffraction. An explanation as to how the residual stresses are generated is at Appendix G. There was variability in the residual stress values measured because of the different size probes and the method used by each company. The stress measurements showed that although the residual stresses on the inner surface of the shaft in the melt and inner radius were compressive, they reached a maximum tensile stress within 100 μm of the surface. A summary of the residual stresses is at Table 4.

Area	Depth (μm)	Stress (MPa)	Comment
Melted zone	Surface	700 (Compressive)	
	60	350 (Tensile)	Depth of corrosion pit G-REDW
	100	550 (Tensile)	Maximum tensile stress
Inner radius	Surface	500 (Compressive)	
	60	350 (Tensile)	Maximum tensile stress / depth of corrosion pit G-CHCN

Table 4

Summary of residual stresses

A FEM model, which was also developed to estimate the residual stresses, showed that there are high stress gradients around the melted zone in the weld and low stress gradients close to the inner radius where the crack initiated on G-CHCN.

It was not possible to measure the residual stress on the countersink where the crack on G-REDW initiated. However, the manufacturer and Cranfield University both assessed that the residual stress in this area was probably around 200 MPa in tension.

1.18.7.5 Effect of residual stress on fatigue limit

While coupon testing is used to establish the fatigue limit of a material, it may not be representative of the fatigue limit of a component where the manufacturing process may introduce different residual stresses to those generated during the production of the coupon. This difference was seen during this investigation in the results of residual stress measurements undertaken by the helicopter manufacturer on coupons and components. For this reason, it may be necessary, when using fatigue data obtained from the testing of coupons, to include residual stresses in the calculation of the maximum stress. Fatigue tests undertaken on components will include the effect of residual stress; therefore where residual stresses are present, the fatigue limit from these tests will be different to that obtained from the coupons. In using the fatigue limit obtained from component testing it is not necessary to include residual stress in the calculation of the maximum stress as it has already been accounted for in the fatigue limit.

1.18.7.6 Effect of corrosion pits on the local stress

The effect of a corrosion pit is to reduce the fatigue limit of the material. Corrosion pits can be considered to be notches and have at their tip a stress concentration factor K_t, whose value is dependent on the geometry of the pit or notch. The stress at the tip of the pit or notch is defined as $K_t\sigma$ where σ is the remote stress. In analysing the fatigue failure of a material containing a notch, the fatigue limit could be reduced by a factor equal to K_t. However, K_t may be regarded as a theoretical worst case as the experimentally measured reduction of the fatigue limit is always less than K_t. The experimentally measured reduction in fatigue limit from a notch, or corrosion pit, is termed K_f. The difference between K_f and K_t is dependent on notch depth, root radius and the strength of the material.

Cranfield University[69] showed, assuming that the corrosion pit geometry can be represented as an idealised 'V' shaped notch, that K_t is strongly dependent on the depth and radius of the corrosion pit. This relationship is seen at Figure 64[70]

[69] Cranfield University. Fatigue lives of corrosion pitted gearbox shafts: Review of investigations and calculations relating to the failures of EC225 gearbox shafts in G-REDW and G-CHCN. P E Irving. September 2013.

[70] Jozelich AM (2009) "Investigation of the transformation of defects into cracks". Cranfield University MSc thesis 2009; Cranfield University..

for three different tip radii, ρ, and the pit depth, d. These relationships are independent of material type and assume that the local stress at the root is less than the yield stress.

A similar corrosion pit, close to the initiation of the crack on G-REDW, had a depth of 44 μm and a tip radius of up to 5 μm.

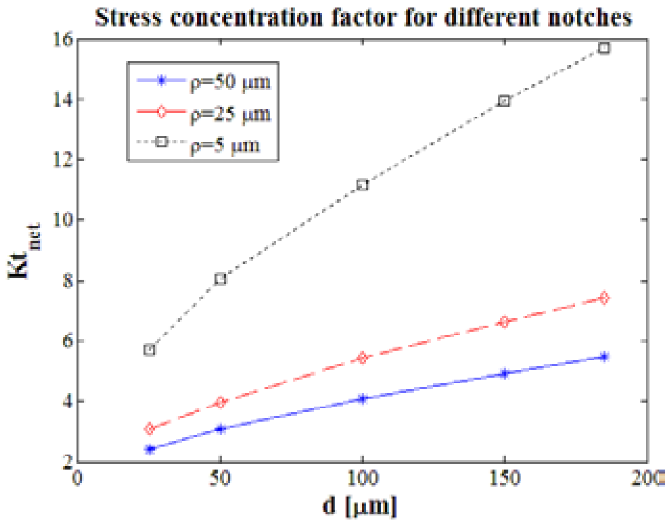


Figure 64

Stress concentration factor K_t related to corrosion pit root radius (ρ) and depth (d) provided by Cranfield University

Experimental data obtained by the manufacturer on the actual reduction in fatigue limit (K_f) as a result of a corrosion pit is shown in Figure 65. From Figure 65 it can be seen that a corrosion pit 60 μm (0.06 mm) deep would result in a K_f of 1.3.

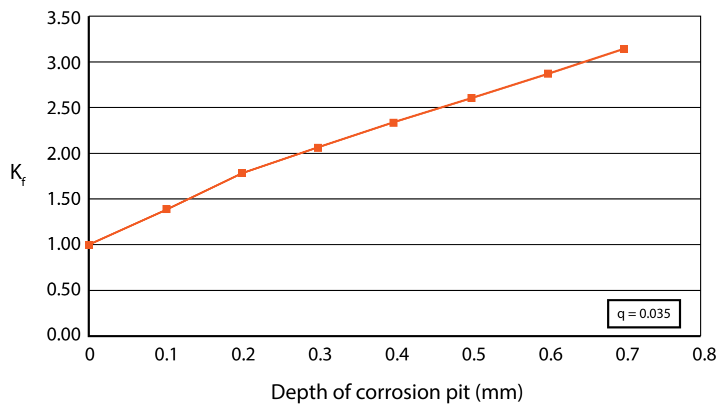


Figure 65

Experimental stress concentration factor K_f related to depth of corrosion pit

1.18.7.6.1 Effect of corrosion pits on the fatigue limit of the shaft

A number of tests, using coupons and bevel gear vertical shafts, were carried out by the manufacturer to establish the effect of a corrosion pit and humidity on the calculated fatigue limit. The results of these tests are summarised in Table 1 and Figure 52.

- On shaft M391, which had no corrosion pits, the estimated fatigue limit at the 4.2 mm hole was demonstrated as being greater than ±655 MPa, with a surface roughness of 1.6 μm, and at the inner radius as being greater than ± 480MPa, with a surface roughness of 3.2 μm.
- The test on shaft M175 demonstrated that 70μm corrosion pits, with low humidity, reduced the estimated fatigue limit of the inner radius to ±282 MPa. Tests on the same shaft demonstrated that the fatigue limit at the 4.2 mm hole was greater than ±422 MPa.
- Tests on shafts M041 and M662 demonstrated that corrosion pits, with high humidity, reduced the estimated fatigue limit of the 4.2 mm hole to ± 292 MPa (40 μm corrosion pit) and the inner radius to ±249 MPa (30 μm corrosion pit).

The tests were carried out on new bevel gear vertical shafts and the quoted fatigue limits included the effect of residual stress (Appendix H). From these tests, the manufacturer concluded that the fatigue behaviour of the bevel gear vertical shafts, with artificial corrosion pits, is dependent on the level of humidity.

1.18.7.7 Material properties

The original fatigue certification of the EC225 LP shaft did not include the properties of the weld and assumed that the parent material and region of the weld had the same Young's Modulus[71]. Metallurgy and coupon tests undertaken during this investigation have shown that the region of the weld has higher strength and hardness, and different crack propagation properties, from the parent material. The following material properties for the parent material were used in the original fatigue analysis of the 4.2 mm hole in the weld, Critical Area 2.

Ultimate Tensile Strength (UTS)	1,180 MPa
Yield strength	950 MPa
Standard fatigue limit	± 600 MPa

[71] Footnote removed August 2014.

The standard fatigue limit of ± 600 MPa was the nominal fatigue limit demonstrated on welded coupons of 32CDV13 steel with a R_a of 1.6 μm (the specified roughness of the hole). The specified surface roughness of the remainder of the shaft is 3.2 μm and the manufacturer calculated, by applying a factor of 0.8 to the fatigue limit of ± 600 MPa, that the fatigue limit for the area of the shaft around the weld was ± 480 MPa.

During this investigation the manufacturer established the following material properties at the inner radius and 4.2 mm hole.

Area	Yield strength	UTS
Inner radius	900 MPa	1,180 MPa
4.2 mm hole	1,440 MPa	>1,440 MPa

1.18.7.8 Safety factors

The safety factor of the shaft can be calculated by determining the distance between the maximum stress and the Gerber line. This can be done graphically or by solving the following Gerber equation; in this equation the residual stress is added to the mean stress σ_m. A description of the safety factor and Gerber line is at Appendix G.

$$\frac{n \cdot \sigma_a}{\sigma_e} + \left(\frac{n \cdot \sigma_m}{\sigma_{UTS}}\right)^2 = 1$$

For the situation when the mean and residual stress is zero, the safety factor (n) is equal to the fatigue limit, $\sigma_{e,}$ divided by the alternating stress σ_a.

The manufacturer determined, through their internal processes, that the bevel gear vertical shaft required, by analysis, a minimum safety factor of 3. To ensure that the EC225 shaft met this requirement, they divided the fatigue limit of the material in the area of the weld by a factor of 3 and verified the maximum dynamic stress (±83 MPa) in Critical Area 2 (4.2 mm hole) was less than this value.

The calculation recorded in the original fatigue substantiation document (Issue A) used a fatigue limit of ±560 MPa, rather than ±600 MPa[72] which was the fatigue limit established by the testing of coupons. The value of ±560 MPa was multiplied by a factor of 0.8 to adjust the fatigue limit for the surface roughness R_a of 3.2 μm. This gave a fatigue limit of ±448 MPa. However, the design drawing specified the surface finish of the 4.2 mm hole as 1.6 μm, which meant that the fatigue limit used for the substantiation was conservative. The calculation recorded in the substantiation document established that in order to

72 The use of 560 MPa instead of 600 MPa was an oversight; however, it resulted in a conservative value.

meet the minimum factor of safety, the maximum dynamic stress would need to be less than ±149.3 MPa (obtained by dividing 448 MPa by 3).

The AAIB used the Gerber equation to calculate the safety factor for a number of different conditions. The calculated safety factors took into account the underestimation of the maximum alternating stress, the inclusion of the residual stress, and the effect of the corrosion pit on the fatigue limit.

The data used in the Gerber equation and the results of the calculations are presented in Table 5 and the supporting notes.

	Area	**Fatigue limit** (MPa)	**UTS** (MPa)	**Residual stress** (MPa)	**Alt Stress** (MPa)	**Mean stress** (MPa)	**Safety factor**
i	Critical Area 2, data from substantiation document Issue A	448	1,180	0	83	6.8	5.4
ii	Critical Area 2 after review of FEM	592	1,180	0	313	17	1.9
iii	Critical Area 2, post FEM and including residual stress	592	1,180	200	313	17	1.7
iv	Critical Area 2 including residual stress and corrosion using K_f=1.3	455	1,180	200	313	17	1.4
v	Critical Area 2, residual stress included and active corrosion. (M041)	292 (estimated)	1,440	–	313	17	0.9
vi	Critical Area 2, including residual stress. Data from substantiation document Issue E	655	1,440	–	313	17	2.1
vii	Inner radius, residual stress included	480	1,100	350	227	10	1.6
viii	Inner radius, residual stress included and active corrosion (M662)	249 (estimated)	1,100	–	227	–	1.1
ix	Inner radius, including residual stress and K_f = 1.3	369	1,100	350	227	10	1.3
x	Critical Area 3 (inner radius), residual stress included. Data from substantiation document Issue E	480	1,100	–	227	10	2.1

Table 5

Summary of fatigue safety factors for the bevel gear vertical shaft

These supporting notes describe each condition in Table 5 for which the safety factor was calculated:

i. Critical Area 2. Data was obtained from the EC225 LP fatigue substantiation documentation, Issue A. Residual stress was not included. The fatigue limit of ± 448 MPa and UTS of 1,180 MPa was for the parent material with a surface roughness of 3.2 µm. The material properties of the weld were not used in the original substantiation.

ii. Critical Area 2. Data was obtained from the revised FEM and included the effect of the revised shaft boundary conditions and the relationship between the 4.2 mm and 29 mm holes. The fatigue limit of ± 592 MPa was the lowest observed during the testing[73] of welded coupons, with a surface roughness of 1.6 µm, carried out by the manufacturer in the 1980's. These results were obtained using a different electron beam weld machine to that currently used. The UTS of 1,180 MPa was obtained from the original fatigue substantiation document.

iii. Critical Area 2. Data was obtained from the revised FEM. Residual stress was included. Fatigue limit of ±592 MPa was for a surface roughness of 1.6 µm.

iv. Critical Area 2. Data was obtained from the revised FEM. Residual stress was included. A K_f of 1.3 was used for the stress concentration factor of a corrosion pit (section 1.18.7.5.2). This K_f was applied to the fatigue limit at Condition iii, to give an adjusted fatigue limit of ±455 MPa.

v. Critical Area 2. An estimated fatigue limit of ±292 MPa was obtained from the testing of shaft M041. Residual stress was included.

vi. Critical Area 2. Data obtained from the revised FEM. Residual stress was included. Estimated fatigue limit of ±655 MPa was obtained from Issue E of the fatigue substantiation document for the bevel gear vertical shaft. This fatigue limit was established, by the manufacturer, from the testing of shafts M391 and M422.

vii. Inner radius. Data obtained from the revised FEM. Residual stress was included. Fatigue limit of ±480 MPa for the parent material was calculated for a surface roughness of 3.2 µm (section 1.18.7.6).

viii. Inner radius. Estimated fatigue limit of 249 MPa was obtained from the dynamic testing of shaft M662. Residual stress was included. The test was carried out in a humid environment and represents the effect of active corrosion.

[73] Eurocopter document 332A056031, Issue A, page 38.

ix. Inner radius. Data obtained from the revised FEM. Residual stress was included. A K_f of 1.3 was used for the stress concentration factor of a corrosion pit (section 1.18.7.5.2). This K_f was applied to the fatigue limit at Condition vii, to give an adjusted fatigue limit of ±369 MPa.

x. Inner radius. This feature has been identified as Critical Area 3 in Issue E of the fatigue substantiation document. The estimated fatigue limit of ±480 MPa was established by the manufacturer from coupon testing and the dynamic testing of shaft M391 and M422. These tests demonstrated that the estimated fatigue limit for the parent material with a surface roughness of 3.2µm, and corrosion, was greater than ±482 and ±453 MPa.

1.18.8 Corrosion protection of the shaft during manufacturing

The practice within the production[74] facility was to degrease the component at the start of each operation and dip it in a bath of oil at the end of each operation. To ensure that oil flowed into the 4.2 mm hole in the weld it was necessary to turn the shaft over and lay it on its side and rotate it in the oil bath. This activity was not recorded on the worksheets and it was possible that it could have been missed, or not carried out thoroughly enough to ensure that oil flowed into the hole.

The visual and dimensional inspection is the last opportunity for corrosion to be detected prior to the fitting of the PTFE plug. However, if the hole and countersinks pass the inspection prior to rework being carried out on another detail, then there is no requirement to re-examine these areas subsequently for corrosion. For the shaft fitted to G-REDW (M385) the holes and countersinks were visually examined on 3 December 2010. As rework was required on other parts of the shaft, the PTFE plug was not fitted until 7 March 2011.

The PTFE plug is fitted when the component is in a degreased condition. Once fitted, it could prevent oil from flowing into any gap that might exist between the insert and the hole / countersink when it is later dipped into the bath of oil.

1.18.9 Change in angle of the countersink in the 4.2 mm hole

The design initially called for a 100º±1º countersink to be formed in the inner and outer surface of the hole in the weld on the EC225 shaft. However, in order to standardise production tooling it was decided to change these countersinks to 90º±1º, the same angle as the countersinks in the hole in the weld on the bevel pinion. The effect on the stress at the hole was determined to be negligible and the change was first made on 14 June 2010 on serial number M330. No change was made to the PTFE plug.

[74] The manufacturing process is described at Appendix C.

Following the accident to G-REDW, the manufacturer established that when the plug is fitted in the 90º±1º countersink a small gap approximately 0.37 mm long and 0.05 mm wide could remain between the insert and the side of the countersink.

1.18.10 Quality of finish of the 4.2 mm hole

Following the accident to G-REDW, eighteen EC225 bevel gear vertical shafts, between serial number M308 and M559, were re-examined by the helicopter manufacturer. All the parts were subject to a visual inspection. In addition, thirteen of the parts were subject to a detailed dimensional inspection; the angle of the inner countersink on six of these thirteen parts was not measured.

From the inspection, it was concluded that with the exception of the angle of the countersink, and the condition of the hole and countersink, the parts conformed to the design definition.

Where data was available it was established that the angle of the outer countersink varied between -4º and +3º outside the design specification. On the inner countersink the angle was found to be between -6º and +14º outside the design limits. The visual inspection also revealed that there was some variability in the angle of the countersink around the hole; there was also evidence of some scoring and tooling marks in the bore of the holes.

1.18.11 Regulatory oversight

European Commission Regulation No 1702/2003, dated 24 September 2003, provides the implementing rules for the airworthiness of aircraft. Part 21 (J) relates to the Design Organisation Approval (DOA) and Part 21 (G) relates to the Production Organisation Approval (POA).

For the EC225 LP, the EASA holds Part 21 (J) responsibility for the regulatory oversight of the DOA holder and the DGAC is responsible for the Part 21 (G) oversight of the POA holder.

Part 21A.139 requires a POA holder to demonstrate that it has established, and is able to maintain, a quality system that will ensure that each part conforms to the applicable design data and is in a condition for safe operation. The regulation requires the quality system to include inspections of parts and internal quality audits.

1.18.12 Inspection of the EC225 bevel gear vertical shafts during overhaul

1.18.12.1 Maintenance procedure

The EC225 shafts were overhauled every 2,000 hours in accordance with the Overhaul Manual, Work Card 63.26.37.820. The work card refers to two other manuals: General Notes, 60.00.30.800, and Surface Defects, 60.00.41.800.

The General Notes include instructions on how to carry out the inspection and the following sections are significant for the inspection of the bevel gear vertical shaft for corrosion:

> *'Para 1.2.2 Visual examination*
>
> – *Check for absence of corrosion.*
>
> – *Check bores ….for…*
> *…foreign matter, cracks, scoring, fretting, elongation, overheating, distortions and other deteriorations.'*

Although the overhaul procedure required all the bores to be inspected, a visual inspection of the inside of the shaft was carried out using mirrors and a light sources with the PTFE plug still fitted in the 4.2 mm hole. The inspection was carried out after the shaft had been cleaned and degreased. Following the failure of the shaft fitted to G-REDW, the manufacturer amended the overhaul procedure to require the PTFE plug to be removed and the 4.2 mm hole to be visually inspected.

The General Notes also include the following guidance on the classification of the stress and acceptable level of corrosion.

> *'Para 1.4.1 Classification of stressed area*
>
> *1 Highly stressed areas.*
> *2 Stressed areas.*
> *3 Lightly stressed areas.*

> *Para 1.4.2 Corrosion on steel and aluminium alloys Establish from EC if this applies to the shaft.*
>
> *Area 1: No corrosion is permissible after reworking the part to the repair size or sanding.*
>
> *Area 2: Residual corrosion pits are permissible providing that they are not more than 0.1 mm (.004 in) deep. No corrosion is permissible after sanding, if the repair size is reached.*
>
> *Area 3: A slight corrosion is permissible.*
> *Corrosion marks are authorized, providing that they are not more than 0.1 mm (.004 in) deep.'*

Work Card 63.26.37.820 for the EC225 shaft did not define the classification of the stress in the area of the weld. Consequently, if the repair centre identified any corrosion they would have had to contact the DOA holder for advice. The manufacturer has stated that no such requests had been made.

The manufacturer has advised that they will introduce an amendment to Work Card 63.26.37.820 to include the classification of the stress in the shaft and the acceptable level of corrosion.

1.18.12.2 Feedback from repair centres to the DOA holder

The manufacturer had a system in place where repair centres could report defects or anomalies via a discrepancy or occurrence report. On 15 November 2006 the manufacturer wrote to its repair stations reminding them of the need to inform them of any anomaly within 48 hours of finding it. On 19 June 2009, the manufacturer issued Information Notice 2046-I-00, which highlighted the importance of feeding back any anomalies thorough the occurrence reporting system. Attached to the letter was a copy of the EASA AMC 20-8 'Acceptable Means of Compliance for Airworthiness of Products, Parts and Appliances'. In relation to corrosion the AMC stated the need to report:

> *'III. AIRCRAFT MAINTENANCE AND REPAIR*
>
> *D. Any damage or deterioration (i.e. fractures, cracks, corrosion, delamination, disbonding etc) resulting from any cause (such as flutter, loss of stiffness or structural failure) to:*
>
> *(3) the engine, propeller or rotorcraft rotor system.'*

1.18.12.3 Occurrences of corrosion found during overhaul

There are two maintenance organisations that overhaul the EC225 shafts, one of which is the helicopter manufacturer. A review of their records identified only one documented occurrence of corrosion having been found on the inside of a shaft. The corrosion was found in April 2013 on a shaft (M134) that was on its first overhaul after having flown for 2,196 hours on a helicopter that had operated out of a costal base in Brazil.

1.18.12.4 AAIB review of inspection process

AAIB inspectors examined an EC225 shaft at the manufacturer's overhaul facility to determine if it was possible to detect corrosion on the inside of the shaft in the area of the weld. The AAIB concluded that, with the PTFE plug removed, it was difficult to detect small corrosion pits on the inner countersink of the 4.2 mm hole. While it was slightly easier to detect corrosion pits in the area of the weld, isolated corrosion pits of around 60 μm deep, where the crack initiated on G-CHCN, could be overlooked.

1.18.13 Flight crew checklists

The helicopter manufacturer provides emergency procedures in a section of the approved Rotorcraft Flight Manual (RFM). At the front of the section is an explanation of the terminology relating to the urgency of the failure situation. The operators include a similar section, with more comprehensive advisory material, in their respective Emergency Checklists (Appendix A).

The requirements for air operations on checklist provision are laid down in Annex IV of Regulation (EC) No 216/2008 (the Basic Regulation) Paragraph 1.b, which states for checklists:

> *'A flight must be performed in such a way that the operating procedures specified in the Flight Manual or, where required the Operations Manual, for the preparation and execution of the flight are followed. To facilitate this, a checklist system must be available for use, as applicable, by crew members in all phases of operation of the aircraft under normal, abnormal and emergency conditions and situations. Procedures must be established for any reasonably foreseeable emergency situation.'*

These requirements are further specified in the Regulation on Air Operations (EU) No 965/2012, Part ORO.GEN.110 (h) of Annex III which states:

> *'The operator shall establish a checklist system for each aircraft type to be used by crew members in all phases of flight under normal, abnormal and emergency conditions to ensure that the operating procedures in the operations manual are followed. The design and utilisation of checklists shall observe human factors principles and take into account the latest relevant documentation from the aircraft manufacturer.'*

The helicopter manufacturer provides emergency procedures in its RFM but it does not provide an emergency checklist. Therefore the operators each provide an emergency checklist for their flight crew, derived from the RFM. The manufacturer's specific procedures for '*Total loss of MGB oil or failure of both oil pumps', 'Ditching'* and *'Emergency landing'* and the applicable derived operators' checklists, are reproduced at Appendix A.

The current EASA certification specifications for large helicopters are provided in CS-29 (Large Rotorcraft), Amendment 3, dated 11 December 2012. Within this document are requirements applicable to RFMs. Some of these are:

> *'CS 29.1581 General*
>
> *(a) Furnishing information. A Rotorcraft Flight Manual must be furnished with each rotorcraft, and it must contain the following:*
>
> *(1) Information required by CS 29.1583 to 29.1589.*
>
> *(2) Other information that is necessary for safe operation because of design, operating, or handling characteristics.'*
>
> *and*
>
> *'CS 29.1585 Operating procedures (a) The parts of the manual containing operating procedures must have information concerning any normal and emergency procedures, and other information necessary for safe operation, including the applicable procedures, such as those involving minimum speeds, to be followed if an engine fails.'*

There is no AMC material linked to these specifications.

The EASA Certification Specifications CS-25 (Large Aeroplanes) contains similarly worded requirements. In this case AMC material is provided:

> *'AMC 25.1581 Aeroplane Flight Manual*
>
> *1 PURPOSE*
>
> *The primary purpose of the European Aviation Safety Agency (EASA) approved Aeroplane Flight Manual (AFM) is to provide an authoritative source of information considered to be necessary for safely operating the aeroplane. This Acceptable Means of Compliance (AMC) identifies the information that must be provided in the AFM under the airworthiness regulations and provides guidance as to the form and content of the approved portion of an AFM. Although mandatory terms such as 'shall' or 'must' are used in this AMC, because the AMC method of compliance is not mandatory, these terms apply only to applicants who seek to demonstrate compliance by following the specific procedures described in this AMC.'*

and in the Operating Procedures section:

> *'2) Format. Procedures should be presented either in a narrative or a checklist format, depending upon the intended use of the AFM.*
>
> *(i) Narrative. This format is acceptable if sources of procedures information other than the AFM are intended for flight crew use (e.g. Flight Crew Operating Manual (FCOM)). Procedures presented in this format should be drafted in a manner from which the needed sequence can be easily established.*
>
> *(ii) Checklist. This format should be used if the AFM is intended to be used directly by the flight crew for operating procedures…*
>
> *…(4) Procedures Content.*
>
> *(iii) AFM Used Directly. For those manufacturers and operators that do not produce other sources of procedures information (generally manufacturers and operators of small transports), the AFM is the only source of this information. In this circumstance, the AFM operating procedures information must be comprehensive and include information such as cockpit checklists, systems descriptions and associated procedures.'*

There are equivalent requirements and information in the US FAR, Part 29 Airworthiness Standards: Transport Category Rotorcraft, Part 25 Airworthiness Standards: Transport Category Airplanes, and FAA Advisory Circular No: 25.1581-1 Change: 1.

1.18.14 Sikorsky S-92A accident

On 12 March 2009 a Sikorsky S-92A helicopter suffered a loss of MGB oil and a subsequent malfunction of the main gearbox. The accident was investigated by the Transport Safety Board of Canada (TSB Canada) and reported in Aviation Investigation Report A09A0016. One of the findings of the report was:

> *'The pilots misdiagnosed the emergency due to a lack of understanding of the MGB oil system and an over-reliance on prevalent expectations that a loss of oil would result in an increase in oil temperature. This led the pilots to incorrectly rely on MGB oil temperature as a secondary indication of an impending MGB failure.'*

It was also noted in this report that there was no published descent profile for the crew to follow after the failure.

A safety action taken as a result of the accident was that the operator developed, in consultation with the regulator, a descent profile for a MGB oil pressure loss.

2 Analysis

2.1 Introduction

The loss of the main rotor gearbox (MGB) oil pressure on G-REDW and G-CHCN was the result of a 360º circumferential crack on the bevel gear vertical shaft, in the vicinity of the weld that joined the two sections of the shaft together. As a result of this failure, drive was lost to the main and standby oil pumps that are driven by a pinion on the lower part of the shaft.

Following the loss of oil pressure the crews activated the MGB emergency lubrication systems, which should have allowed the helicopters to continue flying, at a reduced power setting, for a further 30 minutes. This would have been sufficient time to enable G-REDW to return to Aberdeen airport and for G-CHCN to fly to Sumburgh airport. However, the MGB EMLUB captions on both helicopters illuminated, indicating to the crew that their emergency lubrication systems had failed, a situation which required an immediate ditching. Both helicopters ditched successfully in the North Sea. These were the only two occasions that the emergency lubrication system on the EC225 LP had been used in-service. The ditchings took place in relatively benign conditions, during daylight, and in both cases the flotation system worked effectively. The helicopters remained upright; the passengers and crew were able to evacuate onto the liferafts, before being rescued with no serious injuries.

Examination of the MGBs after the accidents revealed that both emergency lubrication systems had operated and there was no visual evidence of heat damage or imminent failure of the gearboxes fitted to G-REDW and G-CHCN. The MGB EMLUB captions had illuminated as a result of the incompatibility between the aircraft wiring and the internal configuration of the pressure switches in both the bleed-air and water/glycol (Hydrosafe 620) supplies.

2.2 Operational aspects

The weather conditions, cruise altitude, airspeed and torque settings were similar for both flights. The flights proceeded uneventfully until the red master emergency warning light, indicating a total loss of MGB oil pressure, illuminated.

2.2.1 G-REDW

The crew of G-REDW were not influenced by any prior knowledge of the failure of a bevel gear vertical shaft so their actions can be taken to meet a realistic expectation of how an unprepared crew will respond.

The commander responded to the warning at once by taking control, reducing speed to V_y and calling for the checklist, thereby completing the first element of the operator's memory actions. The co-pilot did not recollect initially that there were memory actions. However, he realised later, when prompted by the checklist, and carried out the appropriate action once he had crosschecked his selection of the SHOT pushbutton with the commander. This resulted in an elapsed time of 1 min 50 seconds before the emergency lubrication system was activated. In the meantime the commander had broadcast a PAN call and turned through 180º towards the coast.

Once the emergency lubrication system was activated the co-pilot continued with the checklist and advised the commander that the status of the helicopter allowed continued flight for up to 30 minutes. The commander noted this. The indication of a failure of the emergency lubrication system changed the status of the flight to one of 'land immediately'.

While the commander assimilated this new information he continued the descent at a steady rate of 500 fpm. He called for the ditching checklist and during the descent ensured that the helicopter and the passengers were prepared for ditching. When this was completed he continued on course towards the coast, descending to 200 feet with the surface in sight. There are very compelling reasons why a pilot would not want to ditch a helicopter and the commander took the time to consider his decision. Despite the checklist requirement to ditch immediately he was seeking an additional indication that the main gearbox was about to fail; he was influenced by the apparent stabilisation of the oil temperature as well as the proximity of the shore.

There was an almost continuous stream of radio and internal communication during the descent which will inevitably have reduced the commander's capacity for analysis and may have contributed to a delay in his decision to ditch. When both pilots noticed an unusual smell of oil the commander turned the helicopter into wind and carried out a successful ditching.

2.2.2 G-CHCN

The crew of G-CHCN were aware of the circumstances surrounding the accident to G-REDW and this, together with their previous training experience, had a significant influence in their handling of the emergency.

The initial part of the failure management was similar to that for G-REDW and a descent was initiated. The vertical flight profile differed for G-CHCN in that once the emergency lubrication system had indicated as failed the rate of descent was increased significantly until the helicopter was at 500 feet amsl. The rate of descent was then reduced as the helicopter descended close to the surface. The helicopter was then flown for around four minutes, below 250 ft, in sight of the surface towards a nearby vessel.

The commander's decision to delay ditching the helicopter, so that he could hover-taxi close to the ship, was influenced by having read the G-REDW report that explained that the emergency lubrication system had operated correctly. The co-pilot's experience of dealing with the specific emergency in simulator exercises gave him more mental capacity to assist the commander in dealing with the situation.

2.2.3 Procedures and checklists

The activation of the emergency lubrication system on G-REDW took 1 minute 50 seconds. The operator's checklist denoted this as a 'boxed item', indicating that it was to be performed from memory. This suggests that immediacy is required, but the design of the activation system does not facilitate this. The illuminated SHOT push-button, required to activate the emergency lubrication system, is located on the overhead panel, out of normal view. There is no repeater light on a lower instrument panel to attract the crew's attention and on this occasion neither pilot remembered to check if the SHOT push-button was illuminated. In the emergency procedures contained in the RFM, the helicopter manufacturer does not specify whether the action should be performed from memory, nor are they required to do so by regulation.

The elapsed time, from the first failure indications, to the ditching of G-REDW was 8 minutes 55 seconds and for G-CHCN was 7 minutes 6 seconds. A comparison of the flight data shows that the descent profile flown for each helicopter was significantly different; G-REDW's descent was flown with the use of the autopilot upper modes and G-CHCN was flown manually to about 500 feet. No guidance was provided in either the RFM or operator's manuals for a descent strategy detailing the best airspeed, torque and rate of descent. The TSB Canada Aviation Investigation Report A09A0016 details a safety action whereby the operator, in conjunction with the regulator, developed a descent profile for a MGB oil pressure loss. The profile was designed to optimise the descent, to minimise the loads on the MGB and to expedite the landing.

Both helicopters spent several minutes flying close to the surface. It was daylight with good surface conditions. There may be some different interpretation amongst pilots and operators concerning the meaning of *'Land or ditch immediately'.* Indeed, the operator of G-REDW indicated, in a *'note'* within their checklist, that there is a need to *'land immediately'* following the illumination of the MGB EMLUB. There are also different interpretations of 'land immediately' given in the derived operator's manuals, with little guidance from the helicopter manufacturer in the RFM.

A comparison between the helicopter manufacturer's RFM and each operator's emergency checklist showed that there are variations in procedures which could be significant in the management of some failures. For example, in the AFM it is advised that the landing gear may be either up or down for ditching; the G-REDW checklist required the gear to be down and the G-CHCN checklist required gear to be up. None of the differences were considered to have been a factor in the outcome of either of these accidents, but these could affect the outcome of other emergency situations.

The manufacturer provides emergency procedures in their RFM in accordance with certification requirements in CS-29[1]. Emergency checklists are not provided, and are not required to be provided. Conversely, the manufacturers of large fixed wing aircraft are required to provide emergency checklists in their Aircraft Flight Manual (AFM), or other approved material such as an FCOM, where the AFM is not used directly by flight crew.

The aircraft operator is required to provide checklists, so when the aircraft manufacturer does not provide an emergency checklist the operators have to derive their own. The investigation identified that there were variations between the emergency checklists produced by the operators and the emergency procedures in the RFM. When an operator produces an emergency checklist there is a potential for an inadvertent change to an emergency procedure, which could be operationally significant and may not have been intended by the manufacturer.

The following Safety Recommendation is therefore made:

Safety Recommendation 2014-013

It is recommended that the European Aviation Safety Agency provide Acceptable Means of Compliance (AMC) material for Certification Specification (CS) 29.1585, in relation to Rotorcraft Flight Manuals, similar to that provided for Aeroplane Flight Manuals in AMC 25.1581 to include cockpit checklists and systems descriptions and associated procedures.

2.3 Emergency and survival equipment

2.3.1 Crash position indicator

The CPI is a primary radio location aid used in an emergency to alert search and rescue authorities, and assist in the location of the helicopter and survivors. The accidents to G-REDW, G-CHCN and G-REDU are among three survivable off-shore accidents, investigated by the AAIB since the provision of an ADELT has been a mandatory requirement. In all three accidents anomalies were identified with the performance of the ADELT.

[1] CS-29 superseded the JAR 29 requirements applicable at the time of EC225 LP certification.

2.3.1.1 Non-deployment of G-REDW crash position indicator

The CPI on G-REDW did not automatically deploy or transmit following the ditching. The flight crew did not manually activate the CPI, and at the time of the accident the operator's Emergency Procedures contained no requirement for them to do so. The operator has since amended the relevant procedures.

The helicopter remained intact and floating after the ditching and there was no disruption to the CPI wiring. As the accelerations experienced during the ditching were insufficient to trigger the g-switches, and as the crew did not manually activate the beacon, the only remaining means to trigger deployment of the CPI was the water-activated switch.

Photographic evidence shows the water level was above that of the water-activated switch several hours after the helicopter ditched. It has not been possible to determine whether the water had reached this level within the 2-hour life of the SIU battery. The water-activated switch functioned during subsequent testing and no defects were identified which would have prevented the CPI from deploying automatically during the accident.

Given the circumstances of the accident, the CPI should have deployed but its failure to do so did not, in this case, adversely affect the search and rescue effort. It was not possible to determine why the CPI did not deploy automatically. The following were considered as possible contributors:

- the continuity of the wiring in the CPI system when submerged,
- the design of the water-activated switch,
- the location of the water-activated switch relative to the water level,
- the time taken for the water to reach a sufficient level for activation before the system became unpowered.

There is considerable variability between helicopters of the same type or fleet in the installed location of Type 15-503 CPI components. Some installations are more susceptible than others to the possibility of disruption during an accident. The 503-21-1 standard of BRU, with the integral water-activated switch, may serve to increase the likelihood of automatic CPI deployment if the BRU becomes submerged. Additionally, the BDC on the Type 15-503-134 CPI system may increase the likelihood of automatic deployment following disruption to the CPI wiring.

Issues relating to the design, installation and airframe integration of ELTs are among the subjects currently being reviewed by the EASA as part of RMT.0120, which was convened in response to AAIB Safety Recommendation 2011-071, arising from the G-REDU investigation. No further Safety Recommendations on these issues are made.

2.3.1.2 Manual activation of the CPI on G-CHCN

The investigation determined that automatic functionality of the CPI was inhibited on G-CHCN following the manual selection of the TRANSMIT function. This meant that had the floating helicopter subsequently capsized or sank, the CPI would have stayed attached to the helicopter, greatly reducing the possibility of successful detection of the beacon transmission by satellites.

In response to this finding the CPI manufacturer amended the Type 15-503 CPI Operating Manual informing that the CPI system must be reset following a manual TRANSMIT selection, in order to restore full automatic functionality. In addition, the helicopter manufacturer communicated this finding to all operators in Safety Information Notice No. 2567-S-25, dated 18 March 2013 and amended the Flight Manual for all Eurocopter helicopters equipped with a Type 15-503 CPI system.

The AAIB reported these preliminary findings in Special Bulletin S2-2013. The Type 15-503 CPI system is fitted to several other aircraft types and as other ADELT devices may be subject to a similar inhibition of the automatic deployment function following a manual selection to TRANSMIT, the following Safety Recommendations were made on 18 March 2013:

Safety Recommendation 2013-006

It is recommended that the European Aviation Safety Agency requires the manufacturers of aircraft equipped with a Type 15-503 Crash Position Indicator system, or similar Automatically Deployable Emergency Locator Transmitter, to review and amend, if necessary, the respective Flight Manuals to ensure they contain information about any features that could inhibit automatic deployment.

Safety Recommendation 2013-007

It is recommended that the Federal Aviation Administration requires the manufacturers of aircraft equipped with a Type 15-503 Crash Position Indicator system, or similar Automatically Deployable Emergency Locator Transmitter, to review and amend, if necessary, the respective Flight Manuals to ensure they contain information about any features that could inhibit automatic deployment.

In October 2013, the EASA made the following initial response to Safety Recommendation 2013-006:

> *'EASA, in cooperation with the manufacturer, has re-examined the requirements of the Emergency Locator Transmitter EUROCAE ED-62 and studied the system specifications again and it was concluded that the equipment is not 100% compliant to the Minimum Operational Performance Standards (MOPS)…..'*

In demonstrating compliance with ED-62, as part of the ETSO-2C126 approval process, the CPI manufacturer interpreted the ED-62 definition of an ADELT as requiring functionality for manual activation before a crash *or* automatic deployment after a crash. The current ED-62A definition, however, requires manual activation before a crash *and* automatic deployment after a crash. In reviewing these requirements in the course of responding to Safety Recommendation 2013-006, the EASA determined that the ED-62 requirements for manual activation and automatic deployment were in fact parallel requirements. The EASA therefore concluded that the Type 15-503 CPI equipment meets neither the original, nor the current certification requirements and as such, requested that the CPI manufacturer develop a modification to make the equipment compliant with the requirements applicable at the time of certification.

The CPI manufacturer is developing a modification to allow future segregation of the manual transmission and automatic deployment functions. This modification requires a complete redesign of the SIU.

On 17 January 2014 the EASA issued Airworthiness Directive 2014-0019, dated 17 January 2014, applicable to all aircraft equipped with a Type 15-503-134 or Type 15-503-134-1 CPI system. This AD requires:

a. A temporary amendment to the AFM and installation of a placard next to the CPI cockpit control panel which states 'DO NOT USE TRANSMIT OVER WATER', within 30 days of the effective date of the AD.

b. Replacement of the SIU with a modified SIU incorporating automatic CPI deployment following a manual activation, within 24 months of the effective date of the AD.

The action taken by the EASA to mandate the replacement of the SIU with a modified version precludes the need to review and amend the Flight Manuals of aircraft equipped with a Type 15-503 CPI system, as recommended in Safety Recommendation 2013-006. However, the manual TRANSMIT function of the

Type 15-503 CPI system is a safety feature, and limitation on its use over water during the 24-month compliance time of the AD could lead to a reduction in the number of options available to flight crew for activation of the CPI during an emergency ditching or loss of communications scenario. Neither the initial nor final response by the EASA to Safety Recommendation 2013-006 took into account any other types of ADELT which may have similar features. However, the EASA have separately advised the AAIB that the only other similar ADELT system in production does not include any provision for overriding automatic deployment of the beacon. The AAIB have therefore categorised Safety Recommendation 2013-006 as 'Accepted – Closed.'

The FAA rejected recommendation 2013-007; refer to Section 4.3 for the full text of the FAA response.

2.3.1.3 CAA ADELT research

The UK CAA research into ADELT performance identified a number of factors relating to the design, installation and location of ADELTs which adversely affected their functionality. The EASA RMT.0120 is currently reviewing all aspects of helicopter ditching and water impacts, including the functionality of all types of ELTs. In addition ED-62A is in the process of being rewritten. These activities create a significant opportunity to influence the future design specifications and certification requirements for ADELTs, and the findings and recommendations from the CAP 1144 will serve to greatly inform these activities.

2.3.2 Liferafts

During the deployment of the liferafts on G-CHCN, the mooring lines and rescue pack line on the left liferaft became entangled initially preventing it from being used. As it was daylight and the sea state was only moderate, the co-pilot was able to free the liferaft successfully. There were three possible reasons why these lines had become entangled:

- The lines inside the liferaft valise may have been packed incorrectly.
- The lines may have been packed incorrectly when the liferaft was installed in the sponson.
- The liferaft was correctly packed and installed, but due to the design of the installation there may have been variability in the deployment mechanism which might have caused the lines to tangle.

There was evidence that liferafts were sometimes packed incorrectly with the lines exiting the front of the valise instead of the rear. This would cause problems with their deployment. The CMM, however, makes it clear that the lines should exit the rear of the valise.

The instructions in the CMM, however, are not completely clear on how to route the lines when folding and packing the liferaft. Some of the diagrams show the lines in different positions with no explanation. While the mooring lines and rescue pack lines are not supposed to cross when the raft is installed, the final diagram in the packing instructions shows the mooring lines exiting the valise forward of the rescue pack line which would result in them crossing once installed. Therefore the following Safety Recommendation is made:

Safety Recommendation 2014-014

It is recommended that the liferaft manufacturer, Survitec Group Limited, revises the Component Maintenance Manual for the Type 18R MK3 liferaft to include clear instructions and diagrams on how to route the rescue pack lines and mooring lines when packing the liferaft.

The liferaft manufacturer has stated that they will review the CMM and will publish a Service Letter highlighting to liferaft maintenance organisations the importance of the lines exiting the rear of the valise and not the front.

The second possibility is that the lines were routed incorrectly when the liferaft was installed in the sponson. When a sample of liferaft installations were examined at the maintenance organisation that had maintained G-CHCN, two installations were found with incorrectly routed mooring lines and in one case the mooring lines and rescue pack lines were twisted. Therefore, it is possible that a similar installation error may have occurred when the left liferaft on G-CHCN was installed, although the engineer who installed it stated that he did not think he had made such an error and that it had been independently inspected. The Aircraft Maintenance Manual contains instructions and diagrams which are not completely clear as to how the mooring lines and rescue pack lines should be routed, thereby increasing the possibility of the lines being incorrectly installed. Therefore, the following Safety Recommendation is made:

Safety Recommendation 2014-015

It is recommended that Eurocopter revise the Super Puma Aircraft Maintenance Manual Task 25-66-01-061 *'Removal-Installation of the Life Raft Assembly'* to include clear instructions and diagrams on how to route the rescue pack lines and mooring lines when installing the liferaft.

The third possibility is that the liferaft was packed and installed correctly, but the design of the installation may result in variability in the deployment mechanism, causing the lines to tangle. There were similar liferaft deployment problems caused by restricted lines during the accident to EC225 LP, G-REDU, in 2009. As part of the approval process for the liferaft installation 27 tests were completed, but only two tests were carried out with a sponson partially submerged in water. No tests were carried out to simulate rough sea or high wind, as is required for the liferaft approval as a stand-alone item. This is because there are no certification requirements for externally mounted liferafts other than a requirement about how its line should be attached and released from the aircraft.

Therefore the following Safety Recommendations are made:

Safety Recommendation 2014-016

It is recommended that the European Aviation Safety Agency review the installation of the Type 18R MK3 liferaft in the EC225 sponson to ensure that there is a high degree of deployment reliability in foreseeable sea conditions.

Safety Recommendation 2014-017

It is recommended that the European Aviation Safety Agency develop certification requirements for externally mounted liferafts fitted to offshore helicopters which ensure a high degree of deployment reliability in foreseeable sea conditions.

The EASA RMT.0120 working group is aware of this recommendation and is considering proposing certification requirements for externally mounted liferafts that would also take aircraft attitude into account.

Following the G-REDW ditching, the occupants of the right liferaft were concerned about the proximity of the main rotor blade, so cut the long mooring line. Similarly, following the G-CHCN ditching, the occupants of the right liferaft were concerned about the proximity of the tail rotor blades, so cut the long mooring line.

The long mooring line is intended to keep the liferaft attached to the helicopter at a safe distance to aid location and is designed to release automatically if the helicopter sinks. However, the long mooring line on the Type 18R MK3 liferaft is 12 m long and when compared to the dimensions of the EC225 LP, it may not be long enough to ensure that the liferaft is maintained at a safe distance.

Neither the liferaft nor the helicopter manufacturer could explain why 12 m had been chosen for the length of the long mooring line. At the time of the liferaft's approval the CAA Specification No. 2 only required that it be longer than 6 m. The current requirement, ETSO 2C505, requires that it be between 6 and 20 m, whereas AMC to JAR-Ops 3.830 states that the long mooring line should be 20 m long. Therefore, the liferaft was not compliant with the AMC for JAR-Ops 3. The length of the long mooring line should be enough to ensure that the liferaft is able to float at a safe distance from the helicopter and its rotor blades. Therefore the following Safety Recommendation is made:

Safety Recommendation 2014-018

It is recommended that the European Aviation Safety Agency amend the regulatory requirements to require that the long mooring line on liferafts fitted to offshore helicopters is long enough to enable the liferaft to float at a safe distance from the helicopter and its rotor blades.

The EASA RMT.0120 working group is aware of this recommendation and is considering a similar recommendation.

2.4 MGB warnings and indications

2.4.1 General

The MGB warnings and indications were similar on both helicopters and were consistent with the failure of the bevel gear vertical shaft and the simultaneous loss of drive to the main and standby oil pumps.

2.4.2 Analysis of warnings

Just prior to the events there was no evidence of any abnormal MGB oil pressure or temperature indications. It was therefore considered that the oil quantity was sufficient for the normal operation of the MGB.

The sudden loss of the MGB oil pressure and the illumination of the amber cautions MP (main pump low pressure) and SB/P (standby pump low pressure), followed by the red warning light MGB.P (MGB no longer lubricated) were the first indications that the shaft had failed. As the upper portion of the shaft continued to rotate it rubbed against the lower section generating metallic debris that fell into the sump and activated the MGB sump chip warning.

With the oil no longer cooling the bearings, the temperature of the bearings and gears within the MGB would have increased. This heat would have been conducted through the components and gearbox casing to the oil in the sump, which was no longer circulating and being cooled by the oil cooler. The oil temperature sensor, which is located in the gearbox sump and surrounded by oil, transmitted this increasing temperature to the VMS display. The lag between the loss of oil pressure and the rise in the recorded MGB temperature is due to the time taken for the heat at the bearings and gears to be conducted through the gearbox to the oil in the sump.

When the emergency lubrication system was activated, pressurised air and Hydrosafe 620, consisting of water and glycol, entered the MGB. The glycol provided lubrication and the water cooled the gearbox. The increased pressure within the gearbox caused the air and a mixture of some of the water, glycol and oil to exit the MGB through the vent and accumulate on the helicopter decking around the rear of the MGB.

Following the activation of the emergency lubrication system, the MGB oil temperature continued to rise and the MGB EMLUB caption illuminated indicating that the emergency lubrication system had failed. The recorded MGB temperature reached 128ºC on G-REDW and 123ºC on G-CHCN before stabilising at around 120ºC. The cockpit MGB amber oil temperature indicator is triggered when the temperature exceeds 125ºC and therefore would have operated on G-REDW; this warning is not recorded on the CVFDR.

The torque profiles on both helicopters were different following the activation of the emergency lubrication system. On G-REDW the torque remained relatively steady at 40%, whereas on G-CHCN it fluctuated between 40% and 80%. Despite the different torque levels, the temperatures in both gearboxes reduced and stabilised at the same temperatures, indicating that, contrary to the MGB EMLUB warning, the emergency lubrication system was operating correctly.

The delay between the operation of the emergency lubrication system and the drop in the indicated oil temperature can be explained by the time taken for the components in the gearbox to cool down and for the Hydrosafe 620 to remove some of the heat from the oil in the sump. A mixture of oil and glycol found on the decking and sides of the helicopter and the glycol found throughout the gearbox, was further evidence that the emergency lubrication system had operated.

While the recorded MGB oil temperature profiles are similar on both helicopters, it would be inadvisable for crews to use this information to determine if the MGB EMLUB warning is false. With a lubrication system failure, such as loss of oil or failure of both pumps, the oil is no longer being circulated and cooling the MGB. The time lag of the oil temperature measured at the sensor will be different to the temperature of the bearings and gears. If the bearings and gears overheat then they could rapidly fail and cause a catastrophic failure of the gearbox. For this reason crews should act on the warning captions.

2.4.3 Epicyclic chip detector warning

Approximately 3 minutes after the emergency lubrication system was activated, the recorded data showed that on both G-REDW and G-CHCN a metal particle had been detected by the epicyclic chip detector and that the signal had remained active for the remainder of the flight. The epicyclic magnetic chip detector system is not latched, which means that any debris would have had to have remained on the detector for the remainder of the flight.

During the examination of the MGBs no metal particles, of a sufficient size to bridge the gap on the chip detector, were found in the epicyclic module. The signals from the epicyclic chip detectors occurred after the emergency lubrication system had been activated. Consideration was given to the possibility that the conductivity of the Hydrosafe 620 may have been sufficient to generate the signals; however, the properties of the Hydrosafe 620 meant this was unlikely. The emergency lubrication system sprays a relatively large quantity of pressurised fluid and air into the epicyclic module and it is possible that this may have dislodged and washed small quantities of normal wear deposits into the small recess in which the epicyclic chip detector was located. The fine metal particles would initially have been suspended in the fluid, but could have accumulated and bridged the gap on the chip detector. However, the reason for the activation of the epicyclic chip detector warning could not be established.

2.5 Emergency lubrication system

After the failure of the shaft and the loss of gearbox pressure, both crews activated the emergency lubrication system. There is clear evidence within the gearboxes that the system operated, but after 30 seconds they were presented with the MGB EMLUB warning caption.

Early in the investigation it was determined that bleed-air pressure switches at the top end of their specified tolerance can generate an MGB EMLUB caption, even though all the components of the emergency lubrication system are operating within their specifications. In October 2012 the AAIB made Safety Recommendation 2012-034 to the European Aviation Safety Agency:

Safety Recommendation 2012-034

It is recommended that the European Aviation Safety Agency requires Eurocopter to review the design of the main gearbox emergency lubrication system on the EC225 LP Super Puma to ensure that the system will provide the crew with an accurate indication of its status when activated.

Further investigation work was undertaken to evaluate the flow characteristics in both the bleed-air pressure system and the Hydrosafe 620 system throughout the operational envelope. The problem with the wiring of the pressure switches has also been identified. This was caused by an error in the specification issued to the replacement pressure switch manufacturer and resulted in all EC225 LPs, with MOD 0752520 embodied, having a pressure switch configuration that resulted in illumination of the MGB EMLUB caption once the system was activated and after the 30-second delay. This was the reason for the MGB EMLUB caption during the accident flights for G-REDW and G-CHCN.

The helicopter manufacturer has made several modifications that were summarised in their Safety Information Notice No 2606-S-63 on 07 July 2013 and detailed in the Eurocopter ASB No EC225-05A033.

The following were mandated:

- Modification of the wiring to the bleed-air and Hydrosafe 620 pressure switches.
- Replacement of the Hydrosafe 620 pump to ensure that the flow rate variations with temperature are reduced.
- Introduction of bleed-air and Hydrosafe 620 pressure switches with much tighter tolerances.
- Replacement of the emergency lubrication system PCB incorporating a longer delay time to ensure that the system is stabilised once activated.
- Changes to the Maintenance Manual procedures for the Emergency Lubrication System to include: functional electrical tests, tests on the activation pressures for the bleed-air and Hydrosafe 620 pressure switches; more comprehensive testing of the flow rate of the Hydrosafe 620 pump; and more comprehensive testing of the P2.4 valve with engines running.

2.6 Failure of bevel gear vertical shaft

2.6.1 Introduction

Both shafts failed as a result of a fatigue crack that initiated from a corrosion pit approximately 60 µm deep. The crack in the shaft fitted to G-REDW initiated in the inner countersink in the 4.2 mm hole on the fusion line of the weld. The crack in the shaft fitted to G-CHCN initiated on the inner radius in the parent material.

During the initial certification of the EC225 LP, the stress levels in the area of the weld on the bevel gear vertical shaft were considered to have been relatively low. With a calculated safety factor of 5.4, a 60 μm deep corrosion pit should not have caused the failure of the shafts. It was, therefore, necessary to undertake a comprehensive investigation to determine if the material properties and stresses used in the initial design of the EC225 shaft were correct or if there were any features on the EC 225 shafts that might have contributed to the failures.

This section will discuss the various factors that contributed to the failure of the shafts. It includes the results of the two approaches taken during the investigation: the manufacturer's approach that used computer models and testing of components, and a theoretical approach undertaken by Cranfield University. The results of the crack propagation modelling and testing will also be discussed. This work was undertaken to verify that the material properties and stresses in the shaft were fully understood and to explain the trend on the HUMS condition indicators.

This section will address the significant factors in the following order:

- Cause and effect of the corrosion pits.
- Cause and effect of the residual stresses.
- Inaccuracies in the FEM that resulted in the underestimation of the maximum stress.
- The effect of the above on the shaft safety factor.
- Use of the Cranfield University fracture mechanic model to provide independent confirmation that the failure of the shafts could be accounted for by the material properties and stresses.
- The contribution of the differences introduced into the EC225 shaft compared to the AS332 shaft to the crack initiation.
- The use of crack propagation predictions to provide confidence that the material properties and loads on the shaft were understood.

2.6.2 Presence of corrosion on the bevel gear vertical shaft

2.6.2.1 Effect of corrosion pits

While the development of the corrosion pits was different for the shafts fitted to G-REDW and G-CHCN, they were assessed to be a contributory factor in the failure of both shafts.

In the analysis of the effect of the geometry of the corrosion pits, two approaches were taken, one analytical and the second using the results of tests. The analytical approach involved estimating a stress concentration factor for the corrosion pit, which was used as a 'knock-down' factor to reduce the fatigue limit. The second approach involved fatigue bending tests on shafts seeded with corrosion pits.

The second approach, which used test data, also evaluated the effect of the geometry of the corrosion pits in the presence of high humidity. The high humidity could result in corrosion fatigue, which could further reduce the stress level at which a crack could initiate.

The high ratio of inter-granular to trans-granular cracking normally associated with corrosion fatigue was not identified in the fatigue cracks on the shafts fitted to G-REDW and G-CHCN. Tests on EC225 shaft M041 and M662 demonstrated that a significant 'knock-down' on the fatigue limit can occur when humidity and a corrosion pit are present. The manufacturer developed the term 'Active Corrosion' to explain the limited evidence of inter-granular cracking during the early stages of corrosion fatigue.

2.6.2.2 G-REDW development of the corrosion pit

The shaft fitted to G-REDW (M385) had been in storage for a year and then operated for 167 flying hours before it failed approximately two months after being fitted to the gearbox.

The corrosion on this shaft was concentrated in a narrow ring around the inner countersink of the 4.2 mm hole and it was from a pit, located on the weld fusion line, where the crack on G-REDW initiated. Corrosion pits were also found in the bore of the hole. The corrosion pits were very difficult to see with the naked eye and were only initially detected with the use of an SEM. A number of possibilities as to how the corrosion pits formed were considered.

The change of the countersink angle from 100º to 90º meant that there would have been a gap between the PTFE plug and the countersink consistent with the narrow area in which corrosion was found. It is possible that moisture or a corrosive agent might have been trapped in this gap (Figure 66).

However, the geometry of the countersink, which was outside the design specification, should have allowed oil, during the normal operation of the gearbox, to flow into this gap and inhibited the corrosion mechanism. This gap would also have provided a path for other contaminants to enter this area in the countersink during normal operation of the gearbox.

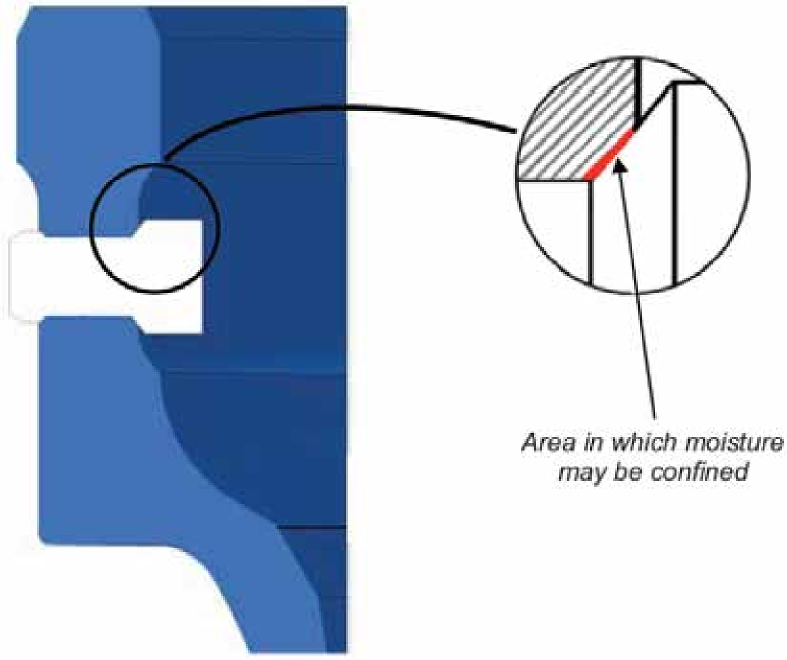


Figure 66

Gap between plug and countersink

Due to the manufacturing tolerances of the PTFE plug and the countersink it is also possible for a small gap to exist in countersinks with an angle of 100º. A similar ring of corrosion to that seen on the shaft fitted to G-REDW was also present in the inner countersink on the shaft fitted to G-CHCN (M122) that had an angle of 100º. On this shaft a slither of PTFE had become trapped between the plug and the countersink which should have allowed oil to enter any gap.

While the investigation discovered that the countersinks had been incorrectly formed on a number of other shafts, the manufacturer advised that apart from these two occurrences, corrosion in this area had never been found on any other AS332 or EC225 shafts. However, the inspections were carried out during the overhaul with the PTFE plug still fitted in the 4.2 mm hole; moreover, during manufacturing and overhaul the area was inspected using a mirror and light source. Consequently, it was unlikely that the inspection would have detected the ring of corrosion seen in the inner countersink on the shafts fitted to G-REDW and G-CHCN. Therefore it was possible that other shafts may have had similar corrosion which was not detected by the inspection methods employed.

The corrosion on the shaft from G-REDW was localised and found nowhere else on the shaft apart from in the area of the 4.2 mm hole. The fact that the narrow ring of corrosion did not extend across the full depth of the countersink suggests that it occurred after the PTFE plug had been fitted into the hole. This operation occurred after the final inspection at the end of the manufacturing process and before the shaft was prepared for storage.

It was not possible to determine when the surface first became contaminated or if the corrosion occurred during manufacture, storage, assembly of the MGB or during operation. Shafts M391 and M422 were both degreased prior to the start of their fatigue tests, at the end of which corrosion pits were found on the 29 mm lubrication holes on both shafts. This demonstrates that it is easy to contaminate the shafts and that unprotected 32CDV13 steel can corrode in ambient conditions.

2.6.2.3 Safety actions taken to prevent corrosion at the 4.2 mm hole

Following the accident involving G-REDW, the manufacturer undertook a number of measures and safety actions to detect damage and prevent corrosion in the area of the 4.2 mm hole in the weld. These actions included:

- New tooling to ensure the countersinks are manufactured to the correct tolerances.
- A final polishing operation during the manufacturing process to remove any corrosion or service blemishes.
- Improved inspection methods.
- Introduction of a sealant to fill the gap between the PTFE plug and countersink.
- Shafts that were subject to the above four processes during the manufacturing process were given serial numbers above M5000.
- Recall of all in-service EC225 shafts that had a countersink of 90º and serial number lower than M5000. The 4.2 mm hole on these shafts was drilled out to ensure the removal of any corrosion and fatigue cracks. The countersinks were then polished and new plugs were installed using sealant to fill the gap. The shafts were reissued, following overhaul and repair, with a new serial number greater than M8000.
- Introduction of a 5 µm acceptance criteria for the allowable depth of scores on shafts with serial numbers in the sequence M5000 and M8000.

2.6.2.4 G-CHCN – development of the corrosion pit

The design of the EC225 shaft was based on the manufacturer's in-service experience of the AS332 shaft and knowledge of the tolerance of other components, manufactured from high strength steels, to corrosion. The manufacturer had no reports of corrosion on the AS332 shaft and stated that the oil mist environment within the gearbox would protect the shafts from corrosion. Testing by the manufacturer, undertaken as part of this investigation, determined that the 32CDV13 steel used in the manufacture of the EC225 shaft was no more susceptible to corrosion than the 16NCD13 steel used in the manufacture of the AS332 shaft. However, the tests were carried out in a severe environment.

The MGB and mast operate at atmospheric pressure and the manufacturer was aware that moisture could enter the gearbox and mast through the vents during normal operation of the helicopter. This was evident by the corrosion which occurred on the inside of some EC225 LP rotor masts that operated in a humid environment and was resolved by the introduction of a new surface finish and decreased inspection intervals of this area. It was established from the results of the random tests of the oil sampled from a number of EC225 LP helicopters operating in the North Sea, that the oil in the MGB can contain water.

The manufacturer had experienced a high scrap rate of the EC225 LP shafts during their first overhaul as a result of wear of the splines on the first stage sun gear. This wear generated a reddish deposit, containing iron oxide, which should have been washed away down the inside of the shaft into the sump by the lubrication oil sprayed through the 29 mm holes in the shaft. The random sampling of oil taken from the MGBs of helicopters operating from Aberdeen identified one helicopter, with a shaft that had operated for 2,207 hours, where the concentration of iron particles in the oil was above the manufacturer's recommended warning threshold. Given that there were no reports of other areas of significant wear within the EC225 LP gearboxes, this finding suggests that the lubrication oil was removing some of the iron oxide debris from the inside of the shaft.

Red deposits were found on the inside of the bevel wheel part of the shaft fitted to G-CHCN (M122). Whilst the deposits covered all of the top part of the inside of the shaft, it was particularly concentrated in three distinct bands where there are features in the shaft. One of these bands was located in the area of the inner radius, a design feature that was not present on the AS332 shaft. On removing the deposit, corrosion pits were found under each of these bands and it was from one of these corrosion pits, located on the inner radius (Figure 67) that the fatigue crack on the shaft fitted to G-CHCN initiated. Apart from these areas, and in the 4.2 mm hole and countersink, corrosion was not found anywhere else on the shaft or on any other components in the MGBs fitted to G-REDW and G-CHCN.

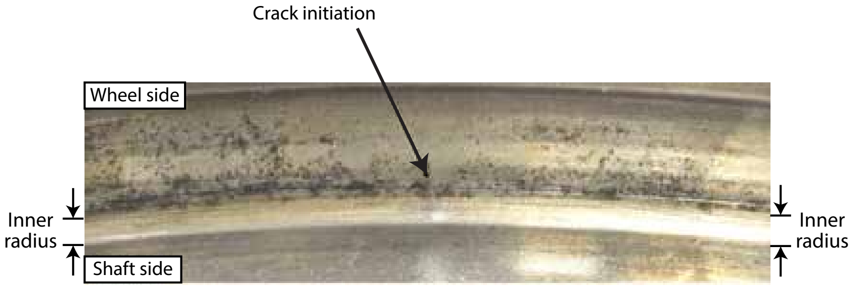


Figure 67

Corrosion pits at the inner radius on G-CHCN

A small survey of EC225 LP shafts, undertaken by the manufacturer as part of this investigation, discovered deposits, which contained iron oxide, in the same areas as on the shaft fitted to G-CHCN. The deposits were concentrated in the same three bands. On a number of shafts, evidence of corrosion pits was discovered under these bands of deposit. However, the deposits and corrosion on the other shafts were not as extensive as on G-CHCN. Corrosion was not found anywhere else on the shafts. This survey suggested that there is a relationship between the shaft life, location where the deposit accumulates and the size and number of corrosion pits.

The investigation considered the mechanism by which the deposit might have caused the corrosion pits. The possibility that a galvanic reaction occurred between the iron oxide particles and shaft material was discounted as being improbable by both the manufacturer and QinetiQ.

The most likely explanation was that the iron oxide was produced at the first stage sun gear splines when the gearbox shaft was rotating. These splines were lubricated by the oil which was sprayed through the 29 mm holes and then flooded up the inside of the shaft. When the shaft stopped rotating, the oil, under gravity, washed some of the iron oxide down the inside of the shaft where it collected in machining marks and recesses in the upper part of the shaft. As the shaft rotated the centrifugal force caused the water, which is heavier than oil, to separate from the oil and become mixed, and trapped, with the iron oxide. This moisture then formed the electrolyte necessary for the corrosion process.

Inspection of the area of the weld, where the corrosion pits occurred, was carried out during the overhaul of the shaft by the use of a mirror and light source. Access was limited and it was possible that evidence of light corrosion in this area might not have been detected during the overhaul. However, the extent and depth of the corrosion under the three bands of deposits was similar and it is probable that the rate of corrosion would have been the same. The band of corrosion located above the first stage sun gear splines was easily accessible and it was unlikely that any visual evidence of corrosion would have been missed during the overhaul (Figure 28, left image). During the overhaul of G-CHCN's shaft, there was no record of any corrosion or excessive wear of the splines. This indicated that the corrosion probably occurred during the 1,800 flying hours and 16 months since it had last been overhauled.

Consideration was given to the possibility that the corrosion on the shaft occurred as a consequence of G-CHCN remaining on and operating from an offshore installation in the North Sea. During this period the helicopter operated short frequent flights, over the sea, between installations and spent more time than usual parked on the installation helideck with the engines shut down. The resulting change in the gearbox temperature and ambient conditions might have increased the amount of moisture that entered the MGB through the vents. Such a scenario might have also occurred in a non-maritime humid environment. However, the result of the survey of other shafts indicated that it was the accumulation of the deposits that was a common factor when corrosion was found on the shafts.

2.6.2.5 Safety actions taken to prevent corrosion inside the bevel gear vertical shaft

As a result of the finding that the accumulation of deposit containing iron oxide was a contributory factor in the formation of the corrosion pits, the manufacturer issued an Alert Service Bulletin, number EC225-05A036, on 7 July 2013. This ASB introduced:

- A periodic internal cleaning of the inside of the EC225 LP bevel gear vertical shaft and the 4.2 mm hole in the weld, every 400 flying hours or two years, whichever came first.
- The introduction of a new plug in the 4.2 mm hole for shafts with serial numbers lower than M5000. This plug is fitted using sealant and can be easily removed for the periodic cleaning of the hole and shaft.
- New oil jets to improve the lubrication and washing of the inside of the shaft with oil.

As part of the redesign of the bevel gear vertical shaft the manufacturer informed the AAIB that in addition to removing features that can trap debris, such as the inner radius, they would also improve the surface finish (R_a) to make it more difficult for deposits to become trapped in machining marks.

2.6.3 Residual stresses

The residual stress changes from a compressive stress on the inner surface of the shaft, to a tensile stress of 350 MPa at a depth of 60 μm in both the weld and inner radius. Such tensile residual stresses only have an effect on crack initiation if there is a defect, such as a corrosion pit, that penetrates to the depth where the tensile residual stress becomes significant. The corrosion pits on G-REDW and G-CHCN were 60 μm deep; therefore, the residual stresses induced by the welding process were a contributory factor in both accidents.

Residual stresses would have been introduced into the shaft during the welding process. To fully relieve these stresses the shaft would have had to undergo a heat treatment, which would have required its temperature to have been raised above 600ºC. However, the two parts of the shaft were welded together after the bevel wheel had been nitrided at 550ºC and raising the shaft temperature above 550ºC would have reduced the effectiveness of the surface hardening process. Consequently the post-weld heat treatment was carried out at 520ºC, which would have relieved some, but not all of the residual stress. Therefore it would have been necessary to consider the residual stress in any fatigue life calculations.

Certification of the AS332 shaft was based on the testing of four shafts which would have included the effect of any residual stresses. As the certification of the EC225 shaft was based on the in-service experience of the AS332 shaft, the manufacturer was only required to carry out coupon testing of the 32CDV13 steel. However, due to the different manufacturing processes the residual stresses in the coupons would have been lower than the residual stress in the shaft.

The following factors might have affected the magnitude and distribution of the residual stresses in the shafts. The AS332 shaft was manufactured from 16NCD13 steel which has a different response to thermal treatments to that of 32CDV13 steel used in the EC225 shaft. Consequently, the different thermal processing during the manufacturer of the shafts would lead to differences in the residual stress after the welding operation. The wall thickness of the two shafts was also different and there was an additional feature on the inside of the EC225 shaft in the area of the weld that was not present on the AS332 shaft.

As a result of the use of different materials, and variations in the dimensions and geometry, the residual stress in the AS332 and EC225 shafts would have been different. However, the certification of the EC225 shaft was based on a FEM that only used the stress levels generated by the torque with no reference to the residual stresses from the manufacturing process.

2.6.4 Stress levels obtained from the manufacturer's FEM and flight tests

The underestimation of the stress levels in the bevel gear vertical shaft obtained from the original FEM was considered to be a contributory factor in both accidents.

Certification of the EC225 shaft was based on a FEM that used the stress levels generated by the torque with no reference to the bending moment or residual stress. However, the AS332 shaft certification was based on the shaft's bending moment; consequently, there was no record in the documentation of the magnitude of the stress levels in the area of the weld.

There were a number of inaccuracies in the FEM for the EC225 shaft. The effect on the local stresses as a result of the relative position of the 4.2 mm and 29 mm holes was not taken into consideration; the shaft bearing constraints were also incorrectly modelled which resulted in an incorrect shaft bending moment. The net effect was an underestimation of the stress in the 4.2 mm hole by a factor of 3.8. The inner radius was not considered to be a critical area at the time of certification.

Since the accidents, the manufacturer has revised the FEM to include the omissions. The stress predictions obtained from the model have been validated by flight and dynamic tests, and the manufacturer has reissued the fatigue substantiation document (Issue E) for the bevel gear vertical shaft.

2.6.5 Safety factors

The safety factor was calculated for the 4.2 mm hole and inner radius for a number of conditions in order to assess the effects of the corrosion pits, residual stress and the underestimation of the alternating stress.

From information contained in the initial fatigue substantiation document (Issue A), produced as part of the certification process for the EC225 shaft, the safety factor for Critical Area 2, the 4.2 mm hole in the weld, was calculated as approximately 5.4. However, a combination of the underestimation of the stress in the original FEM and omission of the residual stress reduced this safety factor to approximately 2.1. A safety factor of 3 was the minimum required by the manufacturer's internal procedures for a critical part where the fatigue life had been established by analysis. Had the manufacturer been aware that the safety factor was less then 3, they would have had to have demonstrated the fatigue life by the testing of a number of shafts.

The effect of the underestimation of the maximum stress and the residual stress can be seen in the following Gerber[2] diagrams where the difference between the maximum stress and the Gerber line represents the safety factor. The stresses and material properties used in calculating the safety factors are presented in Section 1.18.7.7, Table 5.

- The first condition is shown at Figure 68 which represents the understanding during the EC225 LP certification process of the stresses at Critical Area 2: the 4.2 mm hole in the weld. The Gerber line (solid line) is constructed from a fatigue limit of ±448 MPa, based on a surface roughness of 3.2 μm, and the parent material UTS of 1,200 MPa. Point 'a' is the maximum stress, calculated during the original certification, at the 4.2 mm hole and the safety factor is approximately 5.4. The actual safety factor is more conservative as the design drawing specifies the surface roughness of the 4.2 mm hole as 1.6 μm. This would give a fatigue limit of ±560 MPa. This is shown by the dotted Gerber line which uses this fatigue limit and the UTS for the weld material of 1,600 MPa.

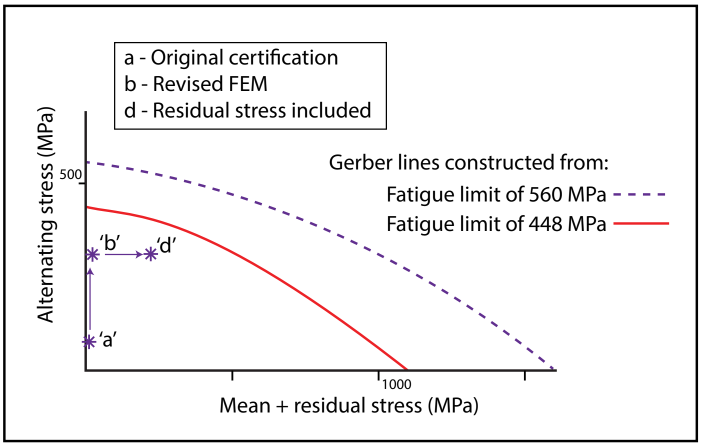


Figure 68

Safety factor at Critical Area 2, 4.2 mm hole in the weld

- The second condition represents the stress after the revision of the FEM following the accidents to G-REDW and G-CHCN. As can be seen at Figure 68, the maximum stress at the 4.2 mm hole has moved from point 'a' to 'b'. The maximum stress at the inner radius is plotted as point 'c' on Figure 69, where the Gerber line is constructed from the parent material fatigue limit of ±480 MPa obtained for a surface roughness of 3.2 μm. The safety factors are approximately 1.9 for the 4.2 mm hole, using the dashed Gerber Line, and 2.1 for the inner radius.

[2] A description of the Gerber diagram and how to establish the safety factor is given in Appendix G.

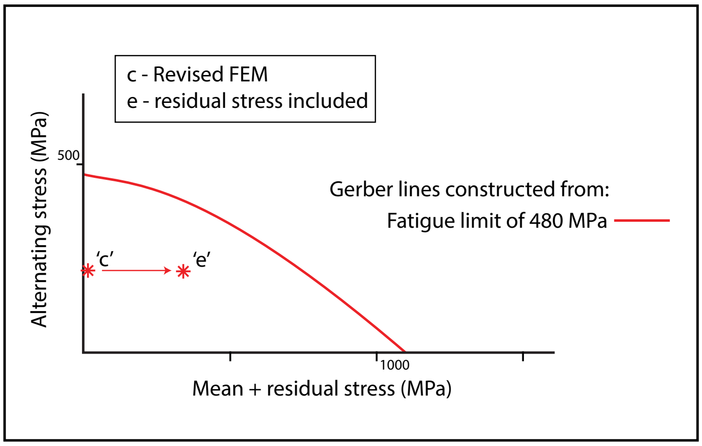


Figure 69

Safety factor at inner radius

- The third condition shows the effect of including the residual stress. The Gerber lines are unchanged, but as can be seen at Figure 68, the maximum stress at the 4.2 mm hole has moved from point 'b' to point 'd'. On Figure 69 it can be seen that the maximum stress at the' inner radius' has moved from point 'c' to point 'e'. The safety factors are now approximately 1.7 for the 4.2 mm hole, using the dashed Gerber Line, and 1.6 for the inner radius.

While the revised FEM model, and addition of the residual stress, has moved the maximum stress closer to the Gerber line, there is still a significant safety factor at the 4.2 mm hole (G-REDW) and the inner radius (G-CHCN).

The following analysis looks at two methods of considering the effect of a corrosion pit on the safety factor. The first method used the manufacturer's value of K_f for the geometry of the corrosion pit to reduce the fatigue limit. The second method used test data and assumes that there is 'active corrosion', which also reduces the fatigue limit of the material. The manufacturer defined 'active corrosion' as the early stages of corrosion fatigue which may not leave physical evidence of its presence.

- The results of the first method, using an experimental K_f of 1.3 for a corrosion pit 60 µm deep, are shown at Figure 70. The fatigue limit is decreased by a factor of 1.3 and the Gerber lines for the weld material and parent material move downwards. Consequently, the safety factors are now approximately 1.4 for the 4.2 mm hole and 1.3 for the inner radius.

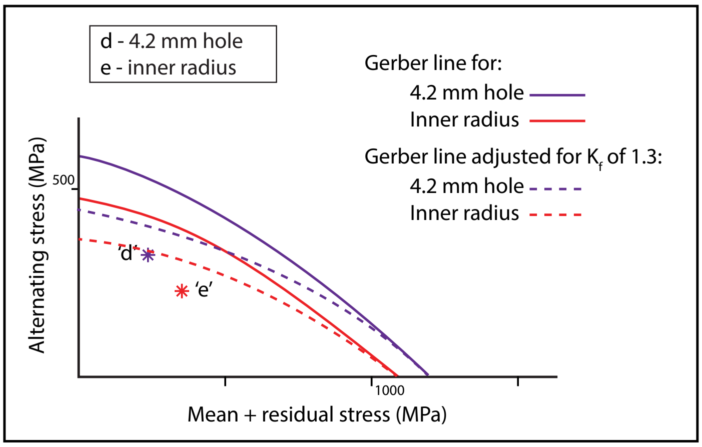


Figure 70

Effect of a corrosion pit 60 µm deep and K_f of 1.3 on the factor of safety

The above analysis does not take into account the variation in the material properties that will affect the fatigue limit, or the tolerance on the estimated stress concentration factor for the corrosion pit. The K_f of 1.3 was based on an R ratio of -1 and it was not established during this investigation if the residual stress would change the R ratio and hence K_f. Moreover, the fatigue limit on the weld fusion line may be lower than on other areas of the weld; this was demonstrated by QinetiQ's coupon testing. Increasing the stress concentration factor K_f of the corrosion pit from 1.3 to 1.8 for the 4.2 mm hole and to 2 for the inner radius would move points 'd' and 'e' onto their respective Gerber lines. The work undertaken by Cranfield University, in support of this investigation, indicated that stress concentration factors of this level are possible and cites some of the manufacturer's data that indicated that the stress concentration factor, taking into account tensile residual stress, could be around 2.

The results of the second method, reduction of the fatigue limit as a result of active corrosion, are shown at Figure 71 and Figure 72. The fatigue limits in the following examples were obtained from the fatigue bending tests of shafts (section 1.16.3.5), which would have included the effect of residual stress.

Figure 71 shows the effect of 'active corrosion' on a crack initiating from a 40 µm corrosion pit on the 4.2 mm hole in the weld. An estimated fatigue limit of ±292 MPa was obtained from the testing of shaft M041. The stress at the 4.2 mm hole is unchanged at point 'd'. The effect of corrosion fatigue is to move the Gerber line downwards such that the safety factor is reduced to 0.9.

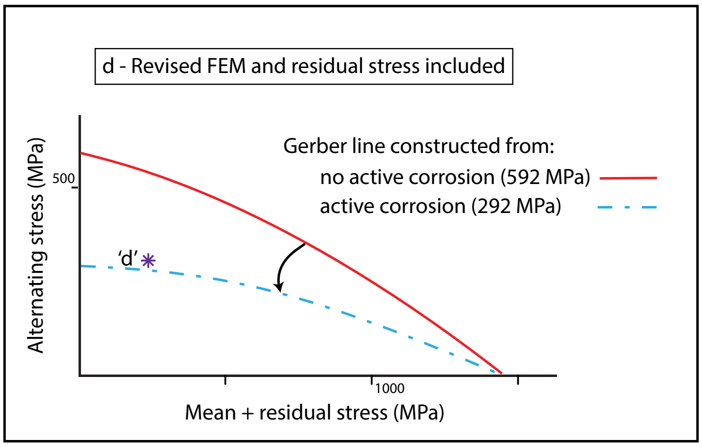


Figure 71

Effect of a corrosion pit 40 µm deep and corrosion fatigue
on the factor of safety at the 4.2 mm hole

Figure 72 shows the effect of ' active corrosion' on a crack initiating from a 40µm corrosion pit on the inner radius. An estimated fatigue limit of ±249 MPa was obtained from the dynamic testing of shaft M662. The stress at the inner radius remains at point 'd'; however, the Gerber line has now moved downwards to an extent that the safety factor is now 1.1.

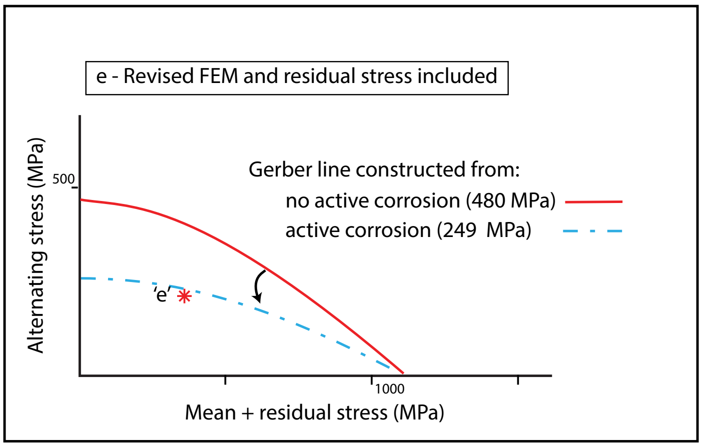


Figure 72

Effect of a corrosion pit 40 µm deep and corrosion fatigue
on the factor of safety at the inner radius

While the use of safety factors and the Gerber line is an analytical approach, it demonstrates that the stress levels, residual stress and corrosion pits and/or 'active corrosion' could explain the fatigue failures of both shafts.

2.6.6 Manufacturer's minimum safety factor

The minimum safety factor, that the manufacturer is required to demonstrate to the regulator during the certification process, is based on service experience and the manufacturer's proprietary formula that takes into consideration the variability of the material properties and small defects such as inclusions in the metal. This safety factor also allows for acceptable levels of corrosion, wear and damage that might occur during normal operation. The manufacturer advised the AAIB that the following aspects were all accounted for in the proprietary formula that they used to establish the minimum safety factor:

- Corrosion that cannot be visibly detected.
- An allowable defect (score) depth of 5µm.
- Variation in the average surface roughness (R_a) resulting from established machining operations.

In the fatigue substantiation document (Issue E[3]) for the bevel gear vertical shaft, the manufacturer's data shows a safety factor of 2.1 for the 4.2 mm hole and 2.3 for the inner radius, which were based on test data. In comparison, the AAIB calculated the safety factors as 1.7 and 2.1 respectively, using an analytical approach. To be conservative, the AAIB used a lower fatigue limit for the 4.2 mm hole that was obtained from coupon testing carried out in the 1980s. The residual stresses are difficult to predict and it was not possible during this investigation to validate the results fully from the models used.

Given the accuracy in modelling the residual stress and the variation of a number of other aspects used in establishing the safety factors, the results of both approaches are relatively close. In determining the acceptability of the minimum safety factor, and to support continued safe operation in conjunction with the HUMS MOD-45 indications, the EASA considered a number of factors. These factors included the service history of the Super Puma fleet, the extent of the testing and the detailed analysis carried out to support the fatigue submission. They also took into account the maintenance activities to detect corrosion on the shafts and design changes that the manufacturer introduced following the accidents. Moreover, the analysis carried out by the AAIB and the manufacturer to calculate the safety factors assumed that the helicopter operated at TOPtran power setting. However, this is a conservative approach as the analysis of the usage of the helicopters in the North Sea revealed that they spent over 99% of their time operating at, or below, MCP.

[3] Fatigue substantiation document, Issue E, was released on 24 May 2013.

2.6.7 Cranfield fracture mechanic model

2.6.7.1 General

To provide confidence that there was no other explanation for the failure of the shafts, Cranfield University was contracted to undertake a fracture mechanics assessment of the failure to determine if the stress levels, including residual stresses, and corrosion pits were sufficient to explain the failures.

Cranfield University used relatively simple models to represent the complex load and stress situation that exists in the bevel gear vertical shaft. Simplifications were made to:

- The local component geometry used in the calculation of the stress intensity ΔK[4].
- The stress intensities due to the residual stresses and the changes in the residual stress as the crack grew.

Moreover, there was only a limited spectrum of EC225 LP in-flight stress data available. Consequently, with the need to use the short crack data for SAE 4340 steel and the variation in material fatigue properties there was some uncertainty in the exact positions of the S-N curves produced by the Cranfield model.

G-REDW and G-CHCN both operated for the majority of time at an engine power setting of MCP.

2.6.7.2 G-CHCN

The results of the Cranfield model show that the fatigue crack on G-CHCN could have occurred with the helicopter operating at MCP using two different data sets:

- When short crack growth data is combined with long crack growth data, excluding the effect of residual stress.
- Long crack growth data, including the effect of residual stress.

[4] ΔK is explained in Appendix G.

2.6.7.3 G-REDW

Using the same data sets, the G-REDW model indicates that the crack on this shaft would have grown only when the power setting was at TOPtran. This power setting was required to produce the tensile stress levels necessary to initially advance the crack. However, as the crack length increased, lower power settings would start to drive the crack.

Combining the short and long crack growth data with the effects of the residual stress might explain how the crack on the shaft fitted to G-REDW was able to propagate when the helicopter was operating at MCP.

2.6.7.4 Summary of findings from Cranfield fracture mechanic model

If the depth of the corrosion pit is considered as a crack then the Cranfield fracture mechanic model indicates that the material properties and the stress levels alone are sufficient to explain the failure of the shaft fitted to G-CHCN. There was less certainty regarding the failure mechanism of the shaft fitted to G-REDW. However, the model showed that it was possible, within the flight operating envelope, for the bevel gear vertical shaft fitted to a EC225 LP helicopter to fail as a result of fatigue cracking from a defect 60 μm deep located in the 4.2 mm hole or on the inner radius.

2.6.8 Availability of short crack data for high strength steels

High strength low alloy steels, such as 32CDV13, are being used at relatively high stress levels in helicopter drive systems. In considering the fatigue life of such systems it is necessary to have an understanding of the effect of high stress components containing small defects such as the 60 μm deep corrosion pits.

While extensive research has previously been carried out into the fatigue performance of metallic materials in general containing small defects, a literature search undertaken as part of this investigation identified few papers that dealt with the fatigue response of high strength steels containing small defects. Moreover, the findings from this previous research was not directly applicable to the situation involving the material and construction of rotating components fitted to helicopters such as the AS332 variants and the EC225 LP for the following reasons:

- Previous research involved different alloys such as aluminium alloy and stainless steel.
- The steel data available was for a much lower strength steel and had a different microstructure (ferritic or bainitic

as opposed to martensitic) from that used in the bevel gear vertical shafts.

- The research did not consider the influence of residual stresses on the growth of short fatigue cracks and fatigue strength.

In order for the regulators to fully understand, during the certification process, the effect on the high cycle fatigue life of defects, such as corrosion pits and scratches, on highly stressed components manufactured from high strength low alloy steel, such as 32CDV13, the following Safety Recommendation is made:

Safety Recommendation 2014-019

It is recommended that the European Aviation Safety Agency commission research into the fatigue performance of components manufactured from high strength low alloy steel. An aim of the research should be the prediction of the reduction in service-life and fatigue strength as a consequence of small defects such as scratches and corrosion pits.

2.6.9 Design of the EC225 bevel gear vertical shaft

The EC225 shaft was based on the design of the AS332 shaft. The main differences, on the EC225 shaft, were:

- The use of a different steel.
- Different case hardening.
- Increased loads at the splines that drive the first stage sun gear.
- The shaft was slightly thicker in the area of the weld and had a new feature identified in this report as the ‘inner radius’.

It was necessary to change from 16NCD13 to 32CDV13 steel to enable parts of the EC225 shaft to be nitrided. Both steels have similar strengths and fatigue limits, and there is no significant difference in their susceptibility to corrosion. The fatigue limit and long crack propagation properties of the 32CDV13 steels are well known but, as with most high strength steels, there is little data on the short crack performance. It was concluded that it was unlikely that the use of 32CDV13 steel instead of 16NCD13 steel was a factor in either accident.

The area where the failures occurred on the shafts fitted to G-REDW and G-CHCN was not case hardened. Therefore the change from a carburising to a nitriding case hardening process was not a factor in either accident.

On the EC225 shaft, the increased load at the splines that drive the first stage sun gear was greater than on the AS332 shaft. This increase in load resulted in an increased rate of wear and the generation of a significant quantity of iron oxide particles. The investigation determined that these iron oxide particles collected in the area of the weld and trapped moisture, causing the corrosion pit from which the fatigue crack on the shaft fitted to G-CHCN initiated. Therefore the increased load on the first stage sun gear splines is considered to be a contributory factor in the development of corrosion pits found on the shaft fitted to G-CHCN.

To accommodate the increased loads, the EC225 shaft was slightly thicker in the area of the weld and FEM analysis showed that the maximum stress in this area was slightly lower than on the AS332 shaft. The residual stresses might have been different. Given that the surface roughness and fatigue limits were similar on both shafts, the change in the thickness in the shaft where the fatigue failures occurred is not considered to have been a factor in either accident.

The feature identified as the inner radius was only present on the EC225 shaft. The local stresses in this part of the area of the weld were similar to those on the AS332 shaft; therefore the change in local stress was not considered to have been a significant factor. However, the inner radius acted to trap wear debris generated by the first stage sun gear, which the MGB oil system was unable to wash away. Therefore the presence of the inner radius is considered to be a contributory factor in the development of corrosion pits found on the shaft fitted to G-CHCN.

2.6.10 Crack propagation

An analysis of the crack propagation rate was carried out to validate the material properties and stress in the shaft. The crack propagation rate was also used to explain the trend on the HUMS condition indicator MOD-45.

The rate that the fatigue cracks grew was dependent on the loads on the shaft, the shaft geometry, the length of the crack and the material properties. This behaviour can be seen on the Paris curve[5] for the shaft material. The crack growth rate predicted by the analysis of the striations and beachmarks differed by a factor of between 2.5 and 3, which was within the normal scatter for fatigue crack growth rates. However, there was a difference in the predictions, using beachmark analysis, made by QinetiQ and the helicopter manufacturer as to the flying hours required for the crack to grow from initiation to final failure.

Striations can be very difficult to observe and interpret; consequently they are not considered to be as reliable as beachmarks. However, as beachmarks

[5] The Paris curve is explained in Appendix G.

could not be detected in the first 4.2 mm of the fracture surface it was necessary to use the striations in this part of the crack when analysing the crack propagation.

It was not possible to establish how long it took for the cracks on both shafts to initiate and grow to the first beachmark. QinetiQ estimated that the time for the cracks to grow from the first beachmark to final failure was similar for the shafts fitted to both helicopters and estimated that it took between 14 and 21 hours. The manufacturer carried out additional tests which, with the analysis of the beachmarks on the accident shafts, determined that it probably took between 20 and 24 flying hours for G-CHCN and 31 flying hours for G-REDW. The difference in the estimation of the time was due to the difficulty in interpreting the beachmarks and the assumptions used in the analysis. However, the results of both analyses showed that once the cracks had initiated and grown to the first beachmark they would grow very quickly.

If the torque into the gearbox remained the same, then the growth rate (da/dN) of the crack would increase as the crack grew in length. This is a consequence of the crack tip stress intensity increasing as the crack length increased and the amount of material connecting the two parts of the shaft together decreased. However, while the growth rate of the crack growth did increase, it was less than that predicted by the Paris curve. This suggested that the load on the shaft decreased as the crack grew around the shaft.

The fatigue cracks should have extended approximately 40% around the circumference before the stress reached a level that would cause the remainder of the shaft to fail in static overload. However, only 1% of the cross-sectional area on both shafts failed as a result of static overload, indicating that the load in the shaft must have been redistributed into the upper bearings.

The HUMS MOD-45 indicator, which monitors the meshing frequency for the bevel gears, detected a rising trend approximately 6 flying hours before the accident on G-REDW and 4.5 flying hours before the accident on G-CHCN. The teeth on the bevel wheels of both shafts also showed evidence of different contact patterns. From the timing obtained from the analysis of the beachmarks it was established that the MOD-45 indicator would start to increase when the combined crack, around the circumference of the shaft, reached a length of approximately 200 mm. A more accurate prediction of the crack growth was obtained from the testing of seven shafts in a dynamic test rig, and one shaft tested in-flight. This demonstrated that the MOD-45 indicator would increase beyond the red threshold after the crack reached a length of between 97 and 100 mm. The contact pattern on the bevel gear teeth and the rising MOD-45 trend was evidence that as the crack grew, the shaft became less rigid allowing a redistribution of the loads in the shaft into the bearings.

The fracture mechanics model was modified with these findings and the predicted crack growth was found to have a close correlation with the Paris curve for the material.

In summary, the fracture mechanics analysis showed that the crack growth rate and wear marks on the bevel wheel teeth could be explained by the redistribution of the shaft loads into the bearings as the crack grew. The work also showed that the combined length of the cracks around the shaft had to grow to between 87 and 100 mm before the shaft would start to flex sufficiently to cause the HUMS indicator MOD-45 to exceed the red threshold.

2.7 HUMS

HUMS, which on the EC225 LP forms part of the M'ARMS, is intended to detect wear and degradation of rotating systems with a low propagation rate. It was not intended to provide warning of the type of failure that occurred to the shafts on G-REDW and G-CHCN. However, for both accidents, the MOD-45 indicator provided an early warning of the shaft failure. MOD-70 also provided warning of the failure on G-REDW, but not on G-CHCN.

The operator of G-REDW identified the rising trend on MOD-45 and MOD-70 from the HUMS data and followed the procedures in the appropriate maintenance manual. As a result, the helicopter was put on close monitoring and an EDR was sent to the manufacturer. The operator was waiting for a response when the accident occurred. The manufacturer subsequently advised the AAIB that the actions taken by the operator were appropriate.

The last download and analysis of the HUMS data for G-CHCN was carried out by the operator on the day prior to the accident flight in a period between two flights. Subsequent analysis of the recovered HUMS data for the flight following the download indicated that there was no discernable trend that required any maintenance action to be taken prior to the first flight on the morning of the accident.

At the power levels used on both accident flights, the propagation tests and the HUMS data from G-REDW and G-CHCN established that the MOD-45 indicator should detect an increasing trend approximately five hours before the shaft failed.

Since the accidents, the MOD-45 indicator has been used to warn of the possible presence of a crack on the bevel gear vertical shaft fitted to EC225 LP helicopters. A number of Service Bulletins and Alert Service Bulletins have been issued. The significant changes introduced were:

- Initially lowering the fleet-wide maximum amber threshold.
- The introduction of red thresholds, both learned and maximum.
- The subsequent removal of the maximum amber alert threshold.
- The lowering of the fleet-wide maximum red alert threshold to 0.2.
- Reduction of the data downloading interval.
- A review of the MOD-45 indicator recordings.

The final action relating to the MOD-45 indicator was the publication of ASB No EC225-45A010 '*Central Maintenance System – HUMS – M'ARMS MOD45 on-board monitoring system*', dated 8 July 2013. The purpose of this ASB was to upgrade the MFDAU (Miscellaneous Flight Data Acquisition Unit) software to:

- Calculate the MOD-45 indicator in real time.
- Increase the acquisition rate.
- Display the MOD-45 indicator status on the HUMS Control Panel or (Man-Machine Interface).
- Continuously save the MOD-45 indicator in the HOMP data.

The ASB also introduced a 'HUMS' light on the instrument panel to indicate for any of the following reasons:

- The MOD-45 threshold has been exceeded.
- An invalid MOD-45 acquisition.
- No MOD-45 acquisition for 30 minutes.
- No data received from the HUMS.

2.8 Manufacturing of the bevel gear vertical shaft

2.8.1 Introduction

With the exception of the inner countersink on the 4.2 mm hole on the shaft fitted to G-REDW (M385), the bevel gear vertical shafts fitted to G-REDW and G-CHCN had been manufactured to the design specifications. The out-of-tolerance geometry of the inner countersink on the shaft fitted to G-REDW was not considered to have been a contributory factor to its accident.

2.8.2 Condition of the 4.2 mm hole

While the bevel gear vertical shaft was classified as a critical item, the low level of stress at the countersinks in the 4.2 mm hole led the manufacturer to assess this feature as non-critical. Following the accident to G-REDW, it was established that the stress at the 4.2 mm hole was much higher than originally assessed and that small defects could result in fatigue cracking. Consequently, the countersink was reclassified as a critical feature.

As the countersink was initially determined to be a non-critical feature, the final inspection at the end of the manufacturing process only required a visual inspection of the countersinks on all shafts. In addition, 10% of the shafts in the batch were subject to a detailed dimensional check of the countersink. If any of the countersinks were found to be outside the design specification, then the countersinks on all the shafts in the affected batch were subjected to the dimensional check. The manufacturing process also included a statistical analysis of the measured dimensions on the completed parts; this analysis did not identify any problems with the manufacturing of the countersinks on the EC225 shaft.

The manufacturer stated that they considered that the spiral scratch in the 4.2 mm hole and the geometry of the inner countersink on the shaft fitted to G-REDW, and on a number of additional EC225 shafts examined during this investigation, was outside the acceptable tolerance. They were unable to explain why the shafts had been released in this condition, but explained that it is difficult to examine visually this small feature from the inside of the shaft.

As a result of these findings the manufacturer introduced a number of safety actions and extended the requirement for a dimensional inspection of the countersinks to all the bevel gear vertical shafts at the end of the manufacturing process. The manufacturer’s quality department also informed the AAIB that they had reviewed the manufacturing process and criticality of features such as the 4.2 mm hole on other similar components to ensure that the lessons learnt from this investigation had been applied, where necessary, to the manufacture of other components.

2.8.3 Surface condition

The geometry and surface condition is an important factor in the fatigue life of the shaft; it is also a factor in the trapping of debris and contaminants that might lead to corrosion. Moreover, testing undertaken by QinetiQ, on coupons manufactured from 32CDV13 steel, showed that small features such as defects and damage in the shaft must be avoided as they could lead to the initiation of fatigue cracks. The fatigue cracks on the shafts fitted to G-REDW and G-CHCN initiated at surface damage (corrosion pits) approximately 60 μm deep.

2.8.4 Surface roughness

The investigation determined that while the average surface roughness, R_a, over the length of the bore of the 4.2 mm hole on G-REDW may have been within the design tolerance of 1.6 µm, over part of the hole the R_a was 2.5 µm. The deepest feature, R_z, was approximately 60 µm to 70 µm deep. The R_a on both shafts, in the area of the weld, was found to be inside the design specification of 3.2 µm. However, the R_z on both shafts was approximately 12 µm and the machining marks in this area appeared to be more pronounced than on other areas on the shaft.

The manufacturer stated that the tooling and machining process used on the EC225 bevel gear vertical shaft was the same as that used on other components and that the variation in R_a and R_z was taken into account in the formula used to calculate the minimum safety factor. The design specifications did not include a limit for R_z as it was considered to be a function of the tooling used and the specified R_a. Nevertheless, the manufacturer considered that the depth of the machining marks in the area of the weld was deeper than they would normally expect and believed that this was due to the increased hardness of the metal in this area. As a result of this finding, the manufacturer advised the AAIB that they have reduced the allowable R_a in the area of the weld, which will have the effect of decreasing the R_z.

2.9 DOA holder design assumptions regarding corrosion

2.9.1 Design assumptions

An important design assumption in establishing the fatigue life of the bevel gear vertical shaft was that it would not corrode in the oil mist environment in the MGB. It was also assessed that corrosion that could not be detected visually would not impact the fatigue life of the shaft. Moreover, significant amounts of corrosion that might adversely affect the fatigue life would be detected during the overhaul carried out every 2,000 flying hours. These assumptions were based on the manufacturer's experience of designing and manufacturing helicopter gearboxes and were supported by the fact that prior to these accidents there had been no reports of corrosion on either the AS332 variants or EC225 LP shafts whose fleets had flown, collectively, approximately 4.5 million flying hours.

2.9.2 Validation of design assumptions

The validation of the design assumptions was obtained by feedback from the inspection carried out during the 2,000-hour overhaul. The DOA holder was required to produce inspection criteria for all critical parts and, for the bevel gear vertical shaft, this criteria was detailed in the Overhaul Manual 63.26.37.820.

While paragraph 1.2.2 of the General Notes in the overhaul manual required the bores in the shaft to be visually inspected for corrosion during the overhaul, the PTFE plug only had to be removed from the 4.2 mm hole if it was necessary to carry out repairs in this area. Consequently the shafts were normally inspected for corrosion with the plugs still fitted. This meant that it was not possible to inspect the countersinks and the internal bore of the 4.2 mm holes for corrosion. The manufacturer stated that this omission had been an oversight and the intention of the DOA holder was for the inspection to be carried out with the plug removed.

This oversight was not considered to be a contributory factor in either accident as the shaft fitted to G-REDW had only flown 167 hours and had not been overhauled, and the crack on the shaft fitted to G-CHCN initiated at a corrosion pit on the inner radius. The PTFE plugs are now required to be removed prior to the inspection.

With approximately 63% of the EC225 shafts being scrapped at overhaul, the possibility that shafts were scrapped after wear was identified, but before the visual inspection for corrosion had been carried out, was considered. The overhaul process was reviewed and it was concluded that as the visual inspection was carried out after the cleaning process and before any measurements were made, corrosion would probably have been identified and documented prior to any decision to scrap the shaft due to wear.

The manufacturer advised the AAIB that there had been no previous reports of corrosion having been found on the AS332 or EC225 shafts. However, a survey of twenty-eight EC225 shafts, undertaken as part of this investigation, found evidence of some light corrosion on a number of these shafts. The possibility that corrosion had been found and not reported back to the DOA holder was considered.

The manufacturer had a system in place where repair centres can report defects or anomalies via a discrepancy or occurrence report. The EASA had undertaken two reviews of the manufacturer's occurrence reporting system over the last four years and advised the AAIB that they were satisfied with their feedback process. Taking everything into account, the AAIB concluded that if corrosion had been found then it would most probably have been fed back to the DOA via the Customer Technical Service Department.

2.9.3 Overhaul of the shaft from G-CHCN

The shaft fitted to G-CHCN at the time of the accident was within 150 hours of its second overhaul and was considered to have been the EC225 LP shaft fleet leader. Corrosion was dependent on the, unforeseen, trapping of the wear products (iron oxide) resulting from the higher maximum torque; compared to the AS332. The manufacturer and repair stations advised the AAIB that the red deposit seen on the inside of the EC225 shaft had not previously been seen on the AS332 shaft, which would explain why these shafts had not suffered from corrosion. Where the manufacturer found corrosion, following the accidents, it was always located under the build-up of the red deposit. This deposit was less, and the corrosion much lighter, than that seen on the shaft fitted to G-CHCN. It is therefore possible that the corrosion pits were not sufficiently large to have been detected during the first 2,000-hour overhaul.

2.10 Summary of failure of the bevel gear vertical shafts

The initiating cause of both accidents was the failure of the bevel gear vertical shaft in the MGB. During certification of the EC225 shaft, the maximum stress had been underestimated as a result of incorrect modelling in the FEM and the omission of the effects of the residual stresses arising from the welding of the two parts of the shaft. Even with this higher-than-predicted maximum stress, the safety factor should have been sufficient to prevent the initiation and propagation of fatigue cracks, providing there were no surface defects such as corrosion pits.

Inhibiting oil was used during the manufacturing process to protect the shafts from corrosion. Moreover, the manufacturer's experience was that the oil mist environment within the MGB would protect in-service shafts from corrosion. However, fatigue cracks still initiated and propagated from corrosion pits on the shafts fitted to G-REDW and G-CHCN.

It was not possible to establish how the inner countersink on G-REDW became corroded. However, the crack initiated at a corrosion pit on the weld fusion line where coupon testing indicated that cracks can initiate at a stress lower than the weld fatigue limit. The safety actions taken, for production and inspection, should reduce the likelihood of corrosion pits occurring in this area.

The corrosion pits on the shaft fitted to G-CHCN occurred as a result of a sequence of events. Safety actions have been taken to reduce the build up of the red deposit, containing iron oxide and moisture, on the inner radius by the introduction of modified oil jets and a cleaning and inspection regime to detect corrosion pits and cracks in the shaft.

The analysis by Cranfield University revealed that the failure of both shafts could probably be explained by the stress levels and the geometry of the corrosion pits. There was no physical evidence of corrosion fatigue on the fracture surfaces of the shafts fitted to G-REDW and G-CHCN. However, tests by the manufacturer indicated that a combination of a corrosion pit and high humidity could reduce the fatigue limit sufficient to cause the failures, without leaving any physical evidence of corrosion fatigue; a process which they described as ‘active corrosion’.

Regardless of whether ‘active corrosion’ was part of the failure mode, the manufacturer is currently redesigning the shaft to address the factors, identified during this investigation, that caused the failure of both shafts.

3 Conclusions

(a) Findings

General

1. The bevel gear vertical shafts on both G-REDW and G-CHCN failed as a result of a 360º circumferential high cycle fatigue crack.
2. Failure of the bevel gear vertical shaft resulted in the loss of drive to the main and standby oil pumps.
3. Loss of oil pressure from the main and standby pumps required the use of the emergency lubrication system.
4. Within a minute of the crews activating the emergency lubrication system, the MGB EMLUB caption illuminated.
5. The emergency procedure required the crew to *'land immediately'* if the MGB EMLUB caption illuminates.
6. Both helicopters ditched in the North Sea; the flotation system activated and the helicopters remained upright.
7. In both accidents, the passengers and crew evacuated the helicopters onto liferafts.
8. There were no reported serious injuries.
9. Neither helicopter sustained any structural damage as a result of the ditching.

Operational aspects

10. Both crews were properly licensed, qualified to conduct the flights and rested.
11. The flights were uneventful until the indication of the loss of the MGB oil pressure.
12. In each case the flight crew actioned the appropriate checklists.
13. The crew of G-CHCN were aware of the accident to G-REDW and had read reports on the initial findings, including the fact that the emergency lubrication system had operated.
14. It took 8 minutes and 55 seconds from the loss of oil pressure until G-REDW ditched.
15. It took 7 minutes and 6 seconds from loss of oil pressure until G-CHCN ditched.

16. The helicopter manufacturer does not provide an emergency checklist and is not required to do so.
17. The operators are responsible for providing their own checklists based on the manufacturer's documentation.

G-REDW CPI

18. The CPI did not deploy automatically following the ditching, nor was it manually activated by the flight crew.
19. In May 2012, the operator's Emergency Procedures contained no requirement for manual activation of the CPI.
20. No defects were found with the components in the system which would have prevented automatic deployment of the CPI.
21. The failure of the CPI to deploy did not adversely affect the search and rescue effort.

G-CHCN CPI

22. The CPI was selected manually by the flight crew to TRANSMIT during the final preparations for the ditching.
23. The design of the CPI system prevents automatic deployment, following manual activation, unless a system reset is performed.

CPI Standards

24. The EASA determined that the Type 15-503 CPI system was not fully compliant with the Minimum Operational Performance Standards specified in EUROCAE ED-62.

Liferafts

25. G-REDW and G-CHCN were fitted with Type 18R MK3 liferafts.
26. Some of the passengers on G-REDW commented that the liferafts were slow to deploy.
27. The Type 18R MK3 liferaft did not meet the certification requirement for a maximum inflation time to a suitable boarding condition of 30 seconds at -30ºC.
28. During inflation of G-CHCN's left liferaft the mooring lines and rescue pack lines became entangled, preventing the liferaft from being used.
29. On G-CHCN, the co-pilot was able to un-twist the lines to free the raft.

30. The CMM for the Type 18R MK3 liferaft did not provide clear diagrams and descriptions on how to route the rescue pack and mooring lines.

31. An inspection of liferaft installations on a sample of Super Puma helicopters revealed two installations where the mooring lines were routed incorrectly. In one of these cases the rescue pack lines were twisted round the mooring lines.

32. The AMM for the Super Puma helicopters did not contain diagrams clearly depicting how the mooring and rescue pack lines should be routed.

33. The tests to certify the Type 18R MK3 liferaft installation on the Super Puma included two tests conducted with a sponson partially submerged in water. No deployment tests from a sponson were carried out in simulated choppy sea conditions.

34. The EASA certification requirements do not specify any deployment reliability or sea state conditions for externally mounted liferafts fitted to offshore helicopters.

35. Following the ditching of G-REDW and G-CHCN the occupants of the liferafts were concerned about the proximity of the rotor blades to the raft, so they cut the long mooring line.

36. The long mooring line on the Type 18R MK3 liferaft is 12 m long which is 8 m less than the 20 m length specified in the AMC to JAR-Ops 3.830.

37. The certification requirements relating to the length of the long mooring line on liferafts do not make any reference to the size and geometry of the helicopter.

Emergency lubrication system

38. In both accidents the emergency lubrication system, once activated, appeared to have successfully cooled and lubricated the main rotor gearbox.

39. A mixture of oil, water and glycol was found on the transmission decking aft of the MGB and down the sides of both helicopters.

40. EC225 LP helicopters, with MOD 0752520 embodied, have a pressure switch configuration that results in illumination of the MGB EMLUB failure caption once the system is activated and after the 30-second delay.

41. The bleed-air pressure from the engine is, under certain conditions, lower than the pressure used in the design and certification of the emergency lubrication system.

42. In some areas of the operational envelope, the Hydrosafe 620 and the bleed-air pressure is such that the pressure switches, which are within specification, can generate a low pressure signal when the emergency lubrication system is operating normally. This would result in an erroneous MGB EMLUB caption.

43. Both Hydrosafe 620 pumps were tested and operated to specification. Both pumps would have operated during the accident flights.

44. Several minutes after activation of the emergency lubrication system, the pressure in the Hydrosafe 620 system decreased to around 0.7 bar relative. This value is higher than the threshold for the pressure switches fitted to the accident helicopter, but lower than the maximum specification for these components.

MGB general

45. There were no external leaks from the MGB and the fluid found on the transmission decking and on the outside of the helicopter had come out of the MGB vent.

46. The MGB on both helicopters had been correctly assembled and with the exception of the damage to the bevel gear vertical shafts, there was no evidence of damage or signs of overheating to any other components in the gearboxes.

47. No additional loads, or resonant frequencies, were identified during the testing of the bevel gear vertical shaft and MGB other than those previously identified during the certification of the EC225 LP helicopter.

G-REDW history of the bevel gear vertical shaft

48. The shaft (M385) fitted to G-REDW was manufactured in March 2012 and had been kept in the manufacturer's stores for a year before it was fitted to the MGB.

49. At the time of the accident, the shaft fitted to G-REDW had flown 167 flying hours and approximately 20 million shaft cycles. The MGB had been fitted to the helicopter two months prior to the accident.

G-CHCN history of the bevel gear vertical shaft

50. The shaft (M122) fitted to G-CHCN was manufactured in April 2008.

51. The shaft, and its MGB, had undergone a 2,000 hour overhaul 1,813 flying hours and sixteen months prior to the accident.

52. At the time of the accident, the shaft had flown 3,845 flying hours and approximately 533 million shaft cycles. The shaft had remained with the MGB since new, but prior to its overhaul had been fitted to another helicopter.

53. At the time of the accident, the shaft fitted to G-CHCN was considered to be the fleet leader on the EC225 LP.

Bevel gear vertical shafts

54. 63% of EC225 LP shafts are scrapped at the first overhaul, of which approximately 50% are due to excessive wear on the splines that drive the first stage sun gear.

55. In comparison with the AS332 shaft, the EC225 shaft is 1.2 mm thicker in the area of the weld and incorporates a new feature identified as the inner radius. There is also approximately 15% more load on the splines that drive the first stage sun gear.

56. In common with other gearbox components, the bevel gear vertical shaft had no surface protection, other than the oil in the MGB, to protect it against corrosion.

Examination of the bevel gear vertical shafts

57. With the exception of the inner countersink on the shaft fitted to G-REDW (M385), both shafts had been manufactured to the design specification and the welds were correctly formed.

58. Corrosion was found in the inner countersink of the 4.2 mm hole on both shafts. This corrosion occurred after the PTFE plugs had been fitted into the 4.2 mm holes.

G-REDW bevel gear vertical shaft examination

59. The geometry of the inner countersink on the shaft fitted to G-REDW was outside the design tolerance.

60. The change in angle of the countersinks and the out of tolerance inner countersink on G-REDW were not factors in this accident.

G-CHCN bevel gear vertical shaft examination

61. A red deposit which contained iron oxide was found in the inside of the top section of the bevel gear vertical shaft fitted to G-CHCN.

62. The deposit on G-CHCN was concentrated in three rings located at the inner radius, and above and below the splines that drive the first stage sun gear.

63. Corrosions pits were discovered under the concentrated areas of deposits on the shaft fitted to G-CHCN. Corrosion pits were not discovered elsewhere on the shaft.

64. The deposit was found on a small number of EC225 LP shafts in the same areas as on the shaft fitted to G-CHCN. There was evidence of corrosion in the same areas as on G-CHCN.

Metallurgic examination of the bevel gear vertical shaft

65. Both shafts failed as a result of a 360º circumferential fatigue crack in the area of the weld that joined the two parts of the shaft.

66. The crack on the shaft fitted to G-REDW initiated in a corrosion pit 60 µm deep, located on the inner countersink in the 4.2 mm hole on the fusion line of the weld.

67. Cracks in the fusion line may initiate and propagate at stress levels lower than the fatigue limit of the weld.

68. The crack on the shaft fitted to G-CHCN initiated in a corrosion pit 60 µm deep located on the inner radius in the parent material.

69. It is difficult to detect corrosion pits visually approximately 60 µm deep located in the inner countersink or inside the shaft in the area of the weld.

70. Prior to these accidents, there had been no previous reports of cracks or corrosion on the Super Puma bevel gear vertical shafts.

71. The area of the shafts that failed is not subject to the carburising or nitriding case-hardening process.

72. The change in case-hardening and the high strength low alloy steel used in the bevel gear vertical shaft were not a factor in the accidents.

73. There was no evidence of corrosion fatigue on the fracture surfaces of either shaft.

74. Beachmarks and striations, which are characteristic of fatigue, were present on the fracture surfaces of both shafts.

75. It is not known how long it took for the cracks on the shafts to initiate and propagate to the first beachmark.

76. Beachmark analysis estimated that the time for the cracks to propagate from the first beachmark to the final failure of the shafts was 15 to 21 flying hours for G-REDW and 14 to 21 flying hours for G-CHCN.

77. The change from 16NCD13 steel to the 32CDV13 steel used in the manufacture of the EC225 shaft was not a factor in these accidents.

Stresses within the bevel gear vertical shaft

78. The EC225 bevel gear vertical shaft was classified at certification as a Critical Part.

79. The EC225 shaft was derived from the AS332 shaft and certification of the EC225 shaft was based on the results of an FEM.

80. The maximum stress in the area of the weld is similar on the AS332 and EC225 shaft.

81. In the initial fatigue substantiation document (Issue A) for the EC225 shaft, the 4.2 mm hole was identified as Critical Area 2. The inner radius was not identified as a critical area.

82. In the FEM used to establish the maximum stress for the certification of the EC225 shaft, the boundary conditions for the upper roller bearing were incorrect.

83. The maximum stress at the 4.2 mm hole occurs when the relative angle between the 4.2 mm and 29 mm hole is 40º. On the shaft fitted to G-REDW the relative angle between these features was 38º.

84. No account was taken of the relative position of the 4.2 mm hole in the weld and 29 mm lubrication hole in the original FEM.

85. Electron beam welding of the two parts of the shaft generates compressive and tensile residual stresses in the area of the weld.

86. There are significant tensile residual stresses, at a depth of 60 µm, in the inner countersink on the 4.2 mm hole and the inner radius in the locations where the cracks initiated in the shafts fitted to G-REDW and G-CHCN.

87. The original fatigue substantiation document for the EC225 shaft made no allowance for the residual stresses.

88. From the data in the initial fatigue substantiation document (Issue A) it was calculated that the safety factor at the 4.2 mm hole in the EC225 shaft was 5.4.

89. Following the revision of the FEM, and incorporation of residual stress, the manufacturer calculated that there was a safety factor of 2.1 at the 4.2 mm hole and 2.3 at the inner radius.

90. The different methods used in the certification of the AS332 and EC225 shafts meant that it was not readily apparent that the maximum stress in the area of the weld had been underestimated.

91. The EASA considered a safety factor of 2.1 for the 4.2 mm hole and 2.3 for the inner radius to be acceptable, providing there is no corrosion in these areas.

Moisture in MGB

92. Low levels of water were found in the oil sampled from a small number of EC225 LP helicopters operating from Aberdeen.

93. Moisture can enter the MGB through the vents located in the gearbox and mast.

94. Moisture in the atmosphere was assessed as previously causing corrosion on the inside of the rotor mast fitted to the EC225 LP helicopters, an area that was not protected by the oil mist in the MGB.

95. The iron oxide generated by wear of the splines that drive the first stage sun gear was trapped at the inner radius on G-CHCN.

96. The MGB oil lubrication system was unable to remove the deposit containing the iron oxide from the inside of the shaft.

97. Moisture in the oil and gearbox became trapped in the deposit resulting in the formation of corrosion pits.

HUMS

98. As the cracks propagated, the load in the shafts was redistributed into the upper bearings, which increased the vibration levels detected by HUMS MOD-45 indicator.

99. The HUMS MOD-45 indicator amber threshold would not have been exceeded until the combined cracks in the bevel gear vertical shaft reached a length of between 87 and 100 mm.

100. The HUMS MOD-45 indicator exceeded the 'learned' amber threshold on both G-REDW and G-CHCN's penultimate flight.

101. The time from the MOD-45 indicator exceeding its amber threshold and the shafts failing was 4.62 hours for G-REDW and 4.75 flying hours for G-CHCN.

102. On identifying the MOD-45 exceedence, the operator of G-REDW followed the appropriate maintenance procedures. These procedures allowed the helicopter to continue flying under 'close monitoring'.

103. Analysis of the HUMS data from G-CHCN, prior to the start of the first flight on the day of the accident, would not have detected an increasing trend on the HUMS MOD-45 indicator.

(b) Causal factors

The following causal factors were identified in the ditching of both helicopters:

a. A 360º circumferential high-cycle fatigue crack led to the failure of the main gearbox bevel gear vertical shaft and loss of drive to the oil pumps.

b. The incompatibility between the aircraft wiring and the internal configuration of the pressure switches in both the bleed-air and water/glycol (Hydrosafe 620) supplies resulted in the illumination of the MGB EMLUB caption.

The following factors contributed to the failure of the EC225 LP main gearbox bevel gear vertical shafts:

a. The helicopter manufacturer's Finite Element Model underestimated the maximum stress in the area of the weld.

b. Residual stresses, introduced during the welding operation, were not fully taken into account during the design of the shaft.

c. Corrosion pits were present on both shafts from which fatigue cracks initiated:

   i. On G-REDW the corrosion pit was located at the inner countersink in the 4.2 mm hole and probably resulted from the presence of moisture within the gap between the PTFE plug and the countersink.

   ii. On G-CHCN the corrosion pit was located at the inner radius and probably resulted from moisture trapped within an iron oxide deposit that had collected in this area.

4 Safety Recommendations and actions

4.1 **Safety Recommendation 2012-034 issued on 17 October 2012**

It is recommended that the European Aviation Safety Agency requires Eurocopter to review the design of the main gearbox emergency lubrication system on the EC225 LP Super Puma to ensure that the system will provide the crew with an accurate indication of its status when activated.

In April 2013 the EASA provided the following response to the Safety Recommendation:

'The root cause of the in-flight Emergency Lubrication (EMLUB) false alarm has been identified. For both helicopters (registered G-REDW and G-CHCN) events, it has been caused by wiring discrepancies found between the electrical outputs of the Air & Glycol pressure-switches of the EMLUB system and the helicopter wiring harness connecting the switches to the EMLUB electronic card. This design non-conformity only exists on helicopters equipped with pressure-switches manufactured by the sensor supplier Industria. The corrective actions have consisted in the following: Eurocopter have developed, through design change MOD 07.53028, a fix at aircraft wiring harness level for helicopters equipped with Industria pressure-switches. The retrofit of the fleet with this EASA approved design change is handled with Eurocopter's Alert Service Bulletin No.05A032, which EASA mandated with Airworthiness Directive (AD) 2013-0037.

From the extensive design review of the EMLUB system, components examinations, system testing and analysis completed during the investigation, it has been furthermore determined that the actual average engine bleed-air pressures for the EMLUB air circuit are lower than the certified design specifications, and indirectly it may also affect the pressures normally expected in the Glycol circuit of the EMLUB system. This brings the potential of triggering the thresholds of the Air and Glycol pressure-switches in some marginal flight conditions. To address this additional EMLUB system issue, Eurocopter are currently designing new pressure-switches with redefined lower pressure thresholds. After their approval, EASA will require Installation of these redesigned pressure-switches for the fleet by another AD.'

This has been assessed by the AAIB as 'accepted – closed'.

4.2 **Safety Recommendation 2013-006 issued on 18 March 2013**

It is recommended that the European Aviation Safety Agency requires the manufacturers of aircraft equipped with a Type 15-503 Crash Position Indicator system, or similar Automatically Deployable Emergency Locator Transmitter, to review and amend, if necessary, the respective Flight Manuals to ensure they contain information about any features that could inhibit automatic deployment.

In September 2013 the EASA provided the following response to the Safety Recommendation:

> *'EASA, in cooperation with the manufacturer, has re-examined the requirements of the Emergency Locator Transmitter EUROCAE ED-62 and studied the system specifications again and it was concluded that the equipment is not 100% compliant to the Minimum Operational Performance Standards (MOPS). The manufacturer is preparing an update to change the behaviour of the system to only allow deployment and activation as being one event. Once the Service Bulletin is available EASA will prepare a corresponding Airworthiness Directive to mandate the system update.*
>
> *This proposed solution, meeting the intent of the requirements, is still under discussion with the applicant to reach a final design change as the ultimate fix for the problem.'*

This has been assessed by the AAIB as 'partially accepted – open'.

4.3 **Safety Recommendation 2013-007 issued on 18 March 2013**

It is recommended that the Federal Aviation Administration requires the manufacturers of aircraft equipped with a Type 15-503 Crash Position Indicator system, or similar Automatically Deployable Emergency Locator Transmitter, to review and amend, if necessary, the respective Flight Manuals to ensure they contain information about any features that could inhibit automatic deployment.

In April 2013 the FAA provided the following response to the Safety Recommendation:

> *'Depending on the type of operation and operating airspace, the FAA may require rotorcraft to have an operating ELT. However, the FAA does not require the installation of a deployable ELT or CPI on helicopters; therefore, the loss of this function is not considered an unsafe condition. In addition, the FAA can only require a change to a design through an airworthiness directive, which requires the determination of an unsafe condition. As a result, the FAA lacks the justification to adopt safety recommendation 13.031, and we plan no further actions.'*

This has been assessed by the AAIB as 'rejected'.

The following additional Safety Recommendations have been made:

4.4 **Safety Recommendation 2014-013**

It is recommended that the European Aviation Safety Agency provide Acceptable Means of Compliance (AMC) material for Certification Specification (CS) 29.1585, in relation to Rotorcraft Flight Manuals, similar to that provided for Aeroplane Flight Manuals in AMC 25.1581 to include cockpit checklists and systems descriptions and associated procedures.

4.5 **Safety Recommendation 2014-014**

It is recommended that the liferaft manufacturer, Survitec Group Limited, revises the Component Maintenance Manual for the Type 18R MK3 liferaft to include clear instructions and diagrams on how to route the rescue pack lines and mooring lines when packing the liferaft.

4.6 **Safety Recommendation 2014-015**

It is recommended that the aircraft manufacturer, Eurocopter Group, revise the Super Puma Aircraft Maintenance Manual Task 25-66-01-061 'Removal-Installation of the Liferaft Assembly' to include clear instructions and diagrams on how to route the rescue pack lines and mooring lines when installing the liferaft.

4.7 **Safety Recommendation 2014-016**

It is recommended that the European Aviation Safety Agency review the installation of the Type 18R MK3 liferaft in the EC225 sponson to ensure that there is a high degree of deployment reliability in foreseeable sea conditions.

4.8 Safety Recommendation 2014-017

It is recommended that the European Aviation Safety Agency develop certification requirements for externally mounted liferafts fitted to offshore helicopters which ensure a high degree of deployment reliability in foreseeable sea conditions.

4.9 Safety Recommendation 2014-018

It is recommended that the European Aviation Safety Agency amend the regulatory requirements to require that the long mooring line on liferafts fitted to offshore helicopters is long enough to enable the liferaft to float at a safe distance from the helicopter and its rotor blades.

4.10 Safety Recommendation 2014-019

It is recommended that the European Aviation Safety Agency commission research into the fatigue performance of components manufactured from high strength low alloy steel. An aim of the research should be the prediction of the reduction in service-life and fatigue strength as a consequence of small defects such as scratches and corrosion pits.

4.11 Summary of safety actions

4.11.1 Main gearbox bevel gear vertical shaft

On 18 May 2012, shortly after the accident to G-REDW, the EASA issued Emergency Airworthiness Directive 2012-0087-E. This required helicopters with certain bevel gear vertical shafts and equipped with the Eurocopter VHM system to download the VHM data and to review the MOD-45 and MOD-75 indicators every 3 flight hours. Helicopters fitted with the affected bevel gear vertical shafts and not equipped with VHM were restricted to day VFR flights when flying over water.

On 11 June 2012, the EASA issued Airworthiness Directive 2012-0104 which superseded 2012-0087-E. This altered the applicability of bevel gear vertical shafts and also increased the time between VHM downloads to 4 flight hours.

On 14 June 2012, the EASA issued Airworthiness Directive 2012-0107 which superseded 2012-0104 which retained the requirements but changed the effective date.

On 28 June 2012, the EASA issued Emergency Airworthiness Directive 2012-0115E which superseded 2012-0107. This retained the requirements of 2012-0107; however, it now required inspection of the VHM indicators in accordance with Eurocopter AS332 ASB No. 01.00.82 or EC225 ASB No. 04A009 both dated 27 June 2012. For the EC225 LP the download interval remained at 4 flight hours.

On 25 October 2012, shortly after the accident to G-CHCN, the EASA issued Emergency Airworthiness Directive 2012-0225E. This superseded the previous EAD 2012-0115E. This retained the requirements of 2012-0115E but increased the applicability to all bevel gear vertical shafts and reduced the interval between VHM inspections; this became 3 flight hours on the EC225. Helicopters with an unserviceable VHM were prohibited flight over water. This referred to changes in Revision 1 to Eurocopter AS332 ASB No. 01.00.82 and EC225 ASB No. 04A009 both dated 24 October 2012.

On 25 October 2012, the CAA issued a Safety Directive SD-2012/002 which stated that UK operators must not conduct a public transport flight or a commercial air transport operation over a hostile environment with any AS332 or EC225 helicopter to which European Aviation Safety Agency Emergency Airworthiness Directive 2012-0225-E dated 25 October 2012 applies. The Norwegian CAA also issued a similar Safety Directive 2012208342-1.

On 21 November 2012, the EASA issued Emergency Airworthiness Directive 2012-0250E which reflected Revision 2 of Eurocopter AS332 ASB No. 01.00.82 and EC225 ASB No. 04A009 both dated 21 November 2012. This required the amendment of the Emergency procedures of the Eurocopter RFM, which introduced the need to reduce engine power to *"MAXIMUM CONTINUOUS TORQUE LIMITED TO 70% DURING LEVEL FLIGHTS AT IAS≥ 60 KTS"* when operating over areas where emergency landing to ground was not possible within 10 minutes at V_y. It also required the continued monitoring of the VHM at regular intervals. For helicopters not equipped with VHM, the AD restricted operations which did not enable emergency landing on the ground within 10 minutes at V_y.

On 9 July 2013, the EASA issued Emergency Airworthiness Directive 2013-0138E, superseding 2012-0250E, which reflected modifications and procedures, introduced by Eurocopter Service Bulletins EC225 ASB No. 04A009 Revision 2 dated 21 November 2012, ASB No. EC225-04A009 Revision 3 dated 8 July 2013, ASB No. EC225-45A010 dated 8 July 2013, ASB No. EC225-05A036 dated 8 July 2013, AS332 ASB No.01.00.82 Revision 2 dated 21 November 2012, ASB No.AS332-01.00.82 Revision 3 dated 8 July 2013, and ASB No. AS332-05.00.96 dated 8 July 2013. These introduced several modifications including the M'ARMS MOD-45 monitoring system. Prior to installing the modified system, the requirement for a regular download of VHM data remained. Also, they required the cleaning of the bevel gear vertical shaft and installation of improved MGB oil jets. For helicopters without VHM or an unserviceable VHM, the power restrictions remained and it introduced an ultrasonic inspection at regular intervals.

On 10 July 2013, the CAA issued Safety Directive SD-2013/001 which removed the restrictions on carrying out public transport or commercial air transport flights over a hostile environment providing certain actions in EASA AD 2013-0138E had been complied with. An updated CAA Safety Directive SD 2013/002 was issued on 16 July 2013 to reflect a revision to EASA AD 2013-0138E dated 15 July 2013.

On 18 December 2013, the EASA issued Emergency Airworthiness Directive 2013-0301, superseding 2013-0138R1, which reflected that some of the requirements in AD 2013-0138R1 had expired, and that Eurocopter issued ASB No.AS332-01.00.82 at Revision 4 dated 17 December 2013 to introduce an Ultrasonic NDT method to detect vertical shaft cracks as alternative method to the only Eddy Current inspection available so far for the AS 332 helicopters.

Additional safety actions

The helicopter manufacturer undertook a number of measures and safety actions to detect damage and prevent corrosion in the area of the 4.2 mm hole in the weld during manufacturing of the shaft. These included new tooling, a final polishing operation, improved inspection techniques, a sealant to fill the gap between the PTFE plug and countersink, a 5 μm inspection criterion for defects and a more detailed inspection at the end of the manufacturing process.

During the investigation the helicopter manufacturer issued several Safety Information Notices and repair letters to operators and maintenance organisations.

The helicopter manufacturer is currently working on a redesigned bevel gear vertical shaft which takes into account the findings of the investigation. The EASA is reviewing this redesign as part of the certification requirements and applying the knowledge gained in the investigation to assess the various safety factors.

4.11.2 Emergency lubrication system

On 22 February 2013, the EASA issued AD 2013-0037 which relates to Eurocopter EC225 EASB No. 05A032 dated 22 February 2013. The AD requires the air and glycol pressure-switches in the emergency lubrication system to be identified. Depending on the type fitted, the switches may require replacing and the helicopter wiring harness may need to be modified (MOD 07.53028). In addition, this AD requires scheduled electrical functional testing of the emergency lubrication system.

On 28 May 2013, the EASA issued AD 2013-0113 which relates to Eurocopter EASB No.04A010 dated 27 May 2013. This updated the RFM by amending the emergency procedure to require an immediate landing as soon as the emergency lubrication system was activated.

On 18 July 2013, the EASA issued AD 2013-0156 which superseded AD 2013-0037 and 2013-0113. The requirements of the previous ADs were retained pending modifications to the emergency lubrication system within 4 months. The modifications are specified in Eurocopter ASB No EC225 05A033 dated 14 July 2013 and introduces new glycol pump, new pressure switches, check of the aircraft wiring and new PCB. Once these modifications are complete the RFM is amended to reintroduce the "land as soon as possible maximum flight time 30 min" to the emergency procedure after the emergency lubrication system is activated.

4.11.3 Crash position indicator

The CPI manufacturer amended the Type 15-503 CPI Operating Manual to reflect that the CPI system must be reset following a manual TRANSMIT selection, in order to restore full automatic functionality.

On 18 March 2013, Eurocopter issued Safety Information Notice No. 2567-S-25, dated 18 March 2013 and amended the Flight Manual for all Eurocopter helicopters equipped with a Type 15-503 CPI system.

On 17 January 2014, the EASA issued Airworthiness Directive 2014-0019, introducing a temporary amendment of the AFM and installation of a placard near the CPI cockpit control panel, to prevent use of the manual TRANSMIT function over water, for all aircraft equipped with a Type 15-503-134 or Type 15-503-134-1 CPI system. This AD also requires replacement of the SIU with a modified SIU incorporating automatic deployment following a manual activation, as a terminating measure for the temporary AFM amendment and placard installation.

On 27 February 2014 the CAA published CAP 1144 *'ADELT Review Report'*, which contains a number of recommendations aimed at optimisation of ADELT installation and designs to maximise the likelihood of an ADELT deploying and transmitting correctly.

4.11.4 Liferafts

The liferaft manufacturer has stated that they will review the CMM and publish a Service Letter highlighting to liferaft maintenance organisations the importance of the lines exiting the rear of the valise and not the front.

The EASA RMT.0120 working group is aware of the issues relating to the liferafts found in the investigation and is considering proposing changes to certification requirements for externally mounted liferafts that would also take aircraft attitude into account.

4.11.5 Other survival equipment

The operator of G-REDW has changed the type of immersion suit used by pilots to an orange and black, closed-neck-seal design.

The supplier of immersion suits has added a further layer of tape over the seam for the toes of the sock to all of its suits to provide increased resistance to damage.

The EBS manufacturer is upgrading the existing re-breathers to include a new means of locating and opening the mouthpiece cover, as well as a retaining strap to hold the mouthpiece in place prior to use when the cover is open.

Information on the following areas affecting survivability was passed to the EASA RMT.0120 and the relevant manufacturers for consideration:

- Seasickness
- Jettison handle positioning and emergency egress
- Safety knives and line cutters
- Immersion suits
- Emergency Breathing Systems

4.11.6 Checklists

Following the accidents the operator of G-REDW made changes to their checklists based on the findings of this investigation.

Appendix A

CHECKLISTS

The manufacturer Eurocopter provides emergency procedures in a section of the Flight Manual. At the front of the section is an explanation of the terminology relating to the urgency of the failure situation.

SECTION 3.1

EMERGENCY PROCEDURES

1 GENERAL

1.1 Wording

The emergency procedures describe the actions the pilot must perform according to the various cases of failure. However, according to the high diversity of the outside factors and type of terrain overflown, the pilot may have to adapt to the conditions from his experience. To help the pilot in his decision, the following four expressions are used:

LAND (OR DITCH) IMMEDIATELY

Self-explanatory.

LAND AS SOON AS POSSIBLE

Emergency conditions, land at the nearest site where a safe landing is possible.

LIMIT DURATION OF FLIGHT

Land at the nearest site at which technical assistance may be expected. Extended flight is not recommended. Duration of flight is left to the pilot's discretion taking into account the operational environment.

CONTINUE THE FLIGHT

Self-explanatory.

(*Courtesy of Eurocopter)*

Figure A-1

Manufacturer's explanation of urgency

2 Definitions

LAND / DITCH IMMEDIATELY

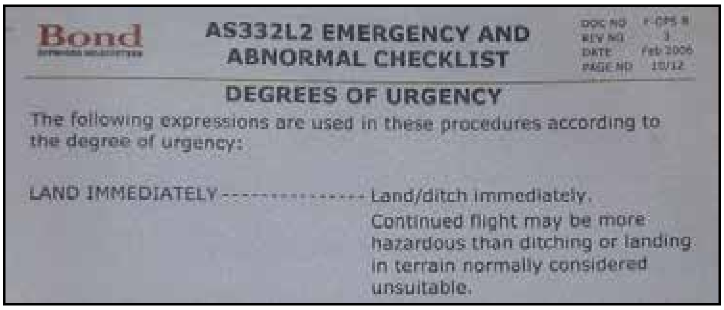
The consequences of continued flight are likely to be more hazardous than, for example, ditching or landing in trees. Where the instruction *"Land or ditch immediately"* is given in a checklist, the procedure for 'emergency landing' should be followed.

(*Courtesy of CHC)*

Figure A-2

G-CHCN Operation's Manual definition of land/ditch immediately

Appendix A (cont)


Bond

AS332L2 EMERGENCY AND ABNORMAL CHECKLIST

DEGREES OF URGENCY

The following expressions are used in these procedures according to the degree of urgency:

LAND IMMEDIATELY --------------- Land/ditch immediately. Continued flight may be more hazardous than ditching or landing in terrain normally considered unsuitable.

(*Courtesy of Bond*)

Figure A-3

Bond definition of land/ditch immediately

The operator provides their crews with an emergency checklist based on material contained in the Aeroplane/Rotorcraft Flight Manual. For example, Figure A-4 shows the emergency procedure for total loss of MGB oil or failure of both oil pumps in the RFM. The operators then take this information to create their individual checklists in their operation's manual.

Symptoms	Condition	Consequences and procedures
MGB P + M.P + S/B.P OIL P 0.4 BAR Oil pressure less than 0,4 bar. + on MGB control box	Loss of MGB lubrication.	**Procedures:** Power .... Reduce to obtain IAS = Vy. .... Press. LAND AS SOON AS POSSIBLE MAXIMUM FLIGHT TIME: 30 min MGB EMLUB EMLUB system failure. LAND IMMEDIATELY

(*Courtesy of Eurocopter*)

Figure A-4

Manufacturer's Flight Manual Procedure for Total loss of MGB oil or failure of both oil pumps

Appendix A (cont)

Bond

EC225LP EMERGENCY AND ABNORMAL CHECKLIST
SECTION 7

DOC NO F-OPS 8
REV NO 5
DATE Dec 2011
PAGE NO 3/10

7/1

TOTAL LOSS OF MGB OIL PRESSURE

INDICATIONS:

WARN - MGB.P - CAUT - XMSN - Aural Gong

M.P - **S/B.P** illuminated on the vehicle monitoring system panel SHOT on MGB control panel.

Oil pressure gauge reading below 0.4 bar

ACTIONS:

Power -- Adjust to obtain Vy

EMLUB SHOT ------------------------------- Press

Clock -- Start
Land --- As soon as possible
Max flight time 30 MINS

NOTE 1: If MGB EMLUB illuminates **LAND IMMEDIATELY**

NOTE 2: If engine no.1 has failed, the emergency lubrication system will not be available. **LAND IMMEDIATELY.**

DRILL COMPLETE

TOTAL LOSS OF MGB OIL PRESSURE 7/1

(Courtesy of Bond)

Figure A-5

G-REDW, total loss of MGB oil pressure checklist

37 TOTAL LOSS OF MGB OIL PRESSURE

INDICATIONS

- MGB.P and XMSN
- M.P and S/B.P
- On MGB control panel: EMLUB SHOT (will not show on the ground)
- MGB oil pressure below 0.4 bar

ACTIONS

1.	EMLUB SHOT	Press
2.	Stopwatch	Start
3.	Airspeed	Set Vy

4. Land as soon as possible (maximum flight time 30 minutes)

5. If **MGB EMLUB** **illuminates on the CWP:**

a. **LAND / DITCH IMMEDIATELY** .... CHECKLIST 123 EMERGENCY LANDING - POWER ON

(Courtesy of CHC)

Figure A-6

G-CHCN, total loss of MGB oil pressure checklist

Appendix A (cont)

3.1 Ditching

Apply appropriate normal or emergency landing procedures as described in Section 3 and Section 4 taking into account the following precautions:

- Check that the emergency floatation gear system is armed. Otherwise, arm it:

NOTE

Only light number 2 illuminates when power is supplied from the battery alone.

- Inflate the balloons:

  Press any of the three inflation pushbuttons on the collective lever grips or the control unit.

NOTE

It is advisable to inflate the emergency floatation gear at an airspeed below 80 kt.

- When possible approach the water facing the wind with the lowest possible forward speed, regardless of the direction of the swell.
- Stop the rotor if possible without using the rotor brake.
- Jettison emergency exits when aircraft is afloat.
- Jettison the life rafts (if fitted),
- Evacuate the aircraft,
- Check auto activation of survival type emergency locator tranmitter (if any) or activate it manually in accordance with the corresponding user manual specifications.

NOTE

Ditching is authorized with landing gear extended or retracted.

13 EMERGENCY LANDING

- **Procedures:**

  If immediate landing is unavoidable:
  . Attempt to ensure the best combination of sink rate and forward airspeed for the touchdown site.
  . Immediately on touchdown pull the General CUT-OUT handle if appropriate (Section 3.1 § 2).

(*Courtesy of Eurocopter)*

Figure A-7

Eurocopter Ditching and Emergency Landing Procedures
(The ditching procedure is provided in a supplement to the Flight Manual.)

Appendix A (cont)

Bond

EC225LP EMERGENCY AND ABNORMAL CHECKLIST SECTION 2

DOC NO F-OPS B
REV NO 5
DATE Dec 2011
PAGE NO 2/10

POWER-ON DITCHING **2/1**

ACTIONS:

Descend	As required – 80 kts into wind
Landing gear	Down
Floatation gear	Arm
Radio	Mayday call (Normally PF action)
Passengers	Brief
Landing/Flood lights	ON
Floatation gear	Inflate at 400' agl
Hover	To assess sea state

On touchdown:

Doors and windows	Jettison
Engines	Secure
Stop rotors	If possible without use of rotor brake: if not apply gently
Evacuate aircraft	When rotors have stopped

DRILL COMPLETE

(Courtesy of Bond)

Figure A-8

G-REDW Checklist; power on ditching and evacuation

123 EMERGENCY LANDING - POWER ON

ACTIONS

1. Collective .... Maintain minimum 20% torque
2. Radio call .... Complete
3. Satellite tracking .... Switch to emergency
4. Landing gear .... Down over land / up over water
5. Parking brake .... OFF
6. Nose wheel .... Locked
7. Floats (over water) .... ARM
8. Passengers .... Brief
9. 500 feet RADALT .... Adjust descent
10. 200 feet RADALT .... Level OFF
11. Emergency landing .... Perform
12. If ditching:
- **a.** Floats .... INFLATE (max 6500 feet above ditching altitude)
- **b.** Groundspeed, drift and ROD .... Reduce to minimum
- **c.** Ditch .... Vertically with wind 10º - 20º from the left

13. After touchdown:
- **a.** CHECKLIST 125 EMERGENCY EVACUATION ON WATER or CHECKLIST 124 EMERGENCY EVACUATION ON LAND

(Courtesy of CHC)

Figure A-9

G-CHCN Emergency landing power on checklist

Appendix A (cont)

125 EMERGENCY EVACUATION ON WATER

ACTIONS

1.	Collective	Lower gently
2.	Sea anchor	Deploy
3.	Engines	Stop
4.	Rotor brake	Apply with caution
5.	CPI	Confirm deployed
6.	Doors	Jettison
7.	Life-rafts	Deploy when rotors stopped
8.	Passengers	Evacuate into life-rafts
9.	LHP	Take torch, board port life-raft
10.	RHP	Take torch and first aid kit, board starboard life-raft
11.	Short / long painters	Cut
12.	Life-rafts	Tie together if possible
13.	Head count	Complete
14.	Survival	Prepare
15.	PLB	Activate one per raft only
16.	**If life-rafts are not available:**	
a.	All passengers	Evacuate through starboard door
b.	Life-jackets	Connect using buddy lines
c.	PLB	Confirm transmitting
d.	Head count	Complete

(*Courtesy of CHC*)

Figure A-10

G-CHCN Emergency evacuation on water checklist

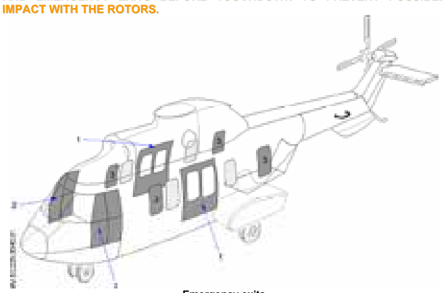
FLIGHT MANUAL

4 EMERGENCY EVACUATION

- **Access Doors:**
  . Pilot's door, co-pilot's door:
  Quick jettison system actuated from inside or outside the aircraft.
- **Windows** (including windows of sliding doors):
  . Jettison by pushing out strongly after removing the seal-retaining strip (depending on version).
- **Stairway Door** (if installed):
  . Open from the inside using the covered handle.
  . Open from the outside using the handle.
- **Sliding Doors** (if installed):
  . Door may be jettisoned from the outside, or from the inside by pushing outward, or using the optional cockpit control.

CAUTION

IN CASE OF EMERGENCY LANDING OR DITCHING, DO NOT JETTISON DOORS AND EMERGENCY EXITS BEFORE TOUCHDOWN TO PREVENT POSSIBLE IMPACT WITH THE ROTORS.



Emergency exits

1. Sliding doors, stairway doors (depending on version). 2. Pilot and co-pilot's doors.
3. Large windows.

Figure 3

EASA APPROVED: EC 225 LP **3.1**

10-40 Page 5

(*Courtesy of Eurocopter*)

Figure A-11

Eurocopter emergency evacuation procedures

Appendix B

DESCRIPTION OF WELD AREA ON BEVEL GEAR VERTICAL SHAFT

The region of the weld consists of two areas, the Heat Affected Zone (HAZ) and the melt; the transition between the two areas is called the fusion line. The design definition specifies the width of the melt and HAZ on the inside surface of the EC225 shaft as 2 ± 0.5 mm. The surface hardness, which is an indication of the strength of the material, reduces slightly between the parent material and HAZ and then increases by approximately 25% across the melt (Figure B-1).

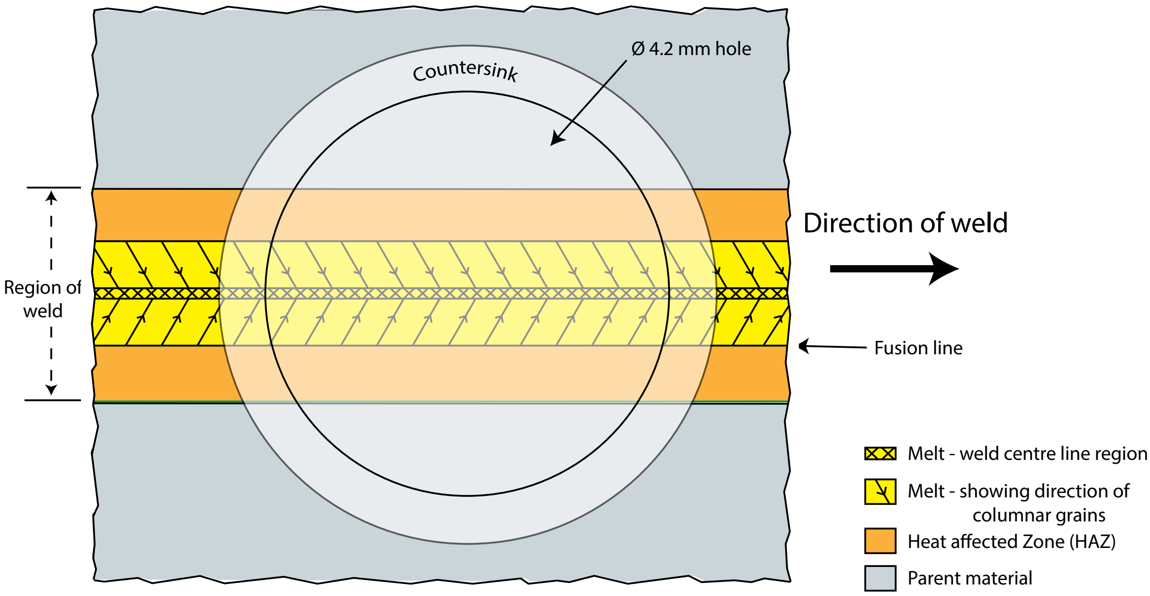


Figure B-1

Features of the weld on the bevel gear vertical shaft

As the melt solidifies, grain boundaries form along the edge of the HAZ and metallic columnar grains grow from this region forming chevrons that point in the direction of the weld with the centre line solidifying last. Within the region of the weld, cracks will tend to grow in either the fusion line or the weld centre line depending on their direction of propagation (Figure B-2).

Appendix B (cont)

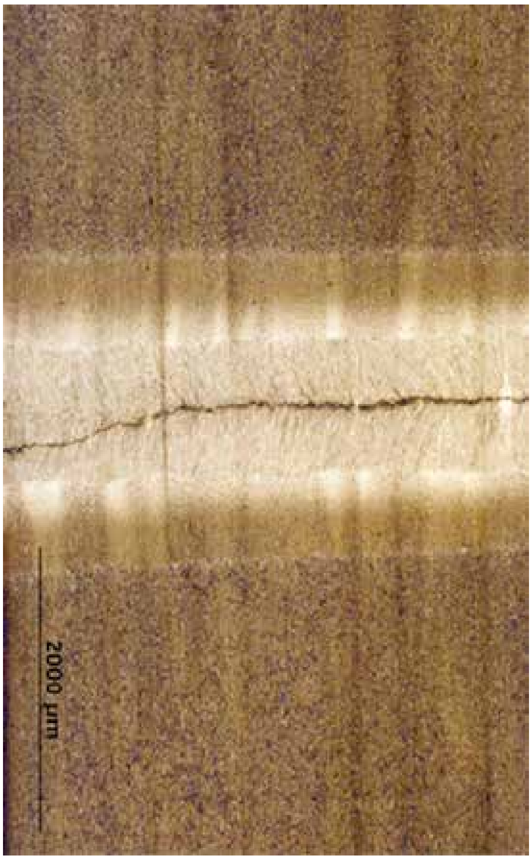


Figure B-2

Microstructure across the region of the weld

In the design of the bevel gear vertical shaft on the EC225, the FEM used to establish the fatigue life of the shaft did not include the properties of the weld and assumed that the parent material and region of the weld had the same Young's Modulus. Metallurgy and coupon tests undertaken by QinetiQ during this investigation have shown that the region of the weld has higher strength and hardness, and exhibited different crack propagation properties to the parent material. However, the helicopter manufacturer's coupon tests showed that the crack propagation rate is similar in the parent and weld material.

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Appendix C

MANUFACTURING OF THE BEVEL GEAR VERTICAL SHAFT

1. Overview

The bevel gear vertical shaft consists of two parts; the bevel wheel and the vertical shaft. Production of the bevel gear vertical shaft is controlled by a route card, which details the work procedures to be carried out at each stage of the manufacturing process. Initially the bevel gear (part number 332A32510100) and the vertical shaft (part number 332A3251102001) are manufactured and controlled by their own route card. Once the parts are welded together, the combined part assumes the serial number of the bevel gear and production continues using its route card.

2. Manufacture of vertical shaft fitted to G-REDW

The vertical shaft (serial number M358) fitted to the MGB on G-REDW was one of 12 identified as batch 20000468361 which were given the serial numbers M349 to M360. The route card was annotated *'Piece & Gamme Critiques'*, which classifies the part as a critical item. Production of the parts commenced in March 2009 and the final inspection report (Bulletin De Contrôle) records no faults or concessions against serial number M358.

3. Manufacture of bevel gear fitted to G-REDW

The bevel gear (serial number M385) fitted to the MGB on G-REDW was one of 10 identified as batch 20000531804 that were given the serial numbers M382 to M391. The route card was also annotated *'Piece & Gamme Critiques'*. Production of the bevel gear commenced in October 2009 and the two parts were welded together in August 2010. There are no records of any faults or concessions having been raised against serial number M385 in the area of the weld.

4. Welding – Operation 190

The nitriding of the bevel wheel assembly is carried out at a temperature of 550°C and takes place prior to Operation 190 when the vertical shaft is joined to the bevel wheel by electron beam welding. The process is as follows:

- The bevel wheel and vertical shaft are degreased, demagnetized and visually inspected for damage or corrosion.
- Both parts are mounted in the fixture and a clamping force is applied by applying a torque of between 5 and 6 mdaN to the bolt that clamps the parts together. This clamping force

Appendix C (cont)

ensures that both parts of the shaft remain in contact during the welding process.

- The concentricity of both parts of the shaft is checked using a Dial Test Indicator (DTI) to ensure that it is within 0.03 mm.
- The fixture is loaded into the automatic welding machine, a vacuum is applied and programme SF016 is uploaded (Figure C-1).

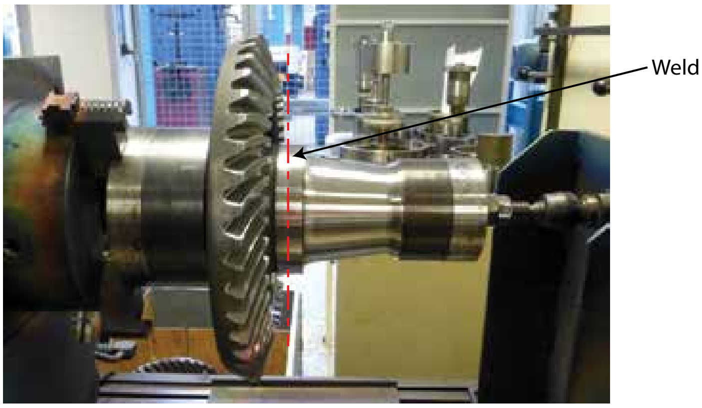

Figure C-1

Bevel gear and vertical shaft mounted in welding machine

- The operator aligns the target on his control screen with the ends of both parts and the welding machine automatically forms one spot weld every 90° around the circumference of both parts (Figure C-2).

Figure C-2

Operator lining the target onto the weld

Appendix C (cont)

- The operator checks the concentricity of the parts by monitoring the target on his control screen as the shaft is rotated. The welding machine automatically forms a superficial weld around the circumference and the concentricity is rechecked. A second 'deep' weld is then formed around the circumference with the end of the weld overlapping the start by about 13º.
- The complete shaft is removed from the welding machine and the weld is visually inspected to ensure that it has correctly formed and that there is a small dimple in the weld where the welding process finished.
- An ultrasonic inspection of the weld is carried out. This inspection does not form part of the QA process but is used as a manufacturing go/no go check.

5. Stress relief – Operation 200 or 201

A stress relief heat treatment is carried out within 24 hours of the welding operation where the shaft is heated to 520ºC, in a vacuum, for 4 hours. This heat treatment is not hot enough, or long enough, and is not intended, to relieve the residual stress introduced during the welding process. Its aim is purely to temper the material that would have been hardened during the welding process.

Due to the size of the oven, the maximum batch size for this operation is four parts.

6. Machining of weld face – Operation 210

During this operation the work instruction requires the external welded flange to be machined to a diameter of 152 ± 0.05 mm and the internal flange to 141.6 ± 0.2 mm.

7. Drilling of hole in weld – Operation 220

During Operation 220 the dimple that formed in the weld during operation 190 is drilled to a diameter of 4.2 mm, the inner and outer countersink is formed and the hole is reamed to ensure that R_a is less than 1.6μm. The manufacturer advised the investigation that the tools that are used to carry out these tasks are issued and returned at the start and end of each batch.

The work instruction in use when the shaft fitted to G-REDW (M385) was manufactured was DI230 issue G, dated 14/6/10. The process is as follows:

Appendix C (cont)

- The bevel gear vertical shaft is placed in a fitting that is mounted into a pillar drill stand.
- Guides are used to line the dimple on the weld with the centre line of the pillar drill.
- A 3.8 mm hole is drilled through the dimple in the weld.
- The external countersink (90º±1º) is formed with a countersinking tool fitted to the pillar drill.
- The internal countersink (90º±1º) is formed using a countersink tool manually turned by hand (Figure C-3).

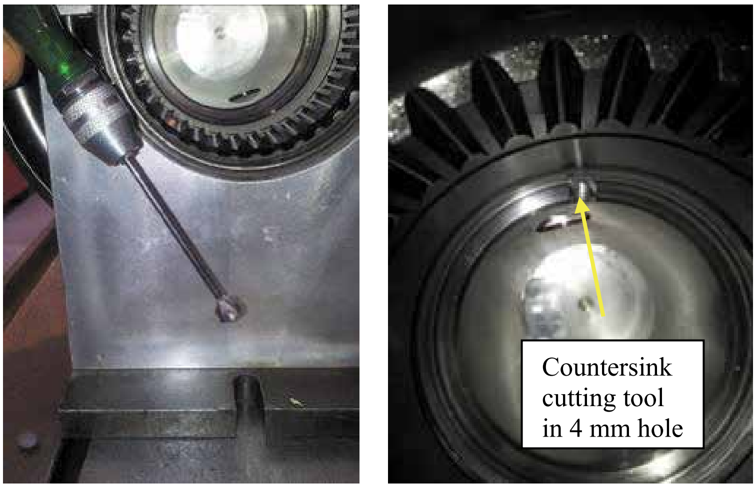

Figure C-3

Hand-operated countersink tool

The hand-operated tool was replaced in April 2012 with a countersink tool fitted to a 90º handheld electrically operated drill (Figure C-4). After the countersinks have been formed, the hole is reamed to a diameter of 4 mm + 0.7 / -0 mm.

Figure C-4
90º electrically operated, handheld drill

Appendix C (cont)

8. Non-destructive inspection – Operation 230 and 240/241

Following the forming of the hole, the weld is subject to an ultrasonic and x-ray inspection. The x-ray plates for the shafts fitted to G-REDW and GCHCN revealed no flaws in the weld and the documentation recorded that both parts passed the ultrasonic inspection.

9. Dimensional inspection – Operation 310

Following a number of further machining operations, away from the weld zones, the part is subject to a dimensional check using a CMM and visual examination. Any request for a concession or rework is documented in the 'Bulletin De Contrôle' and the 'Demande d'Accord Non-Conformité Sur Produit'. Apart from the width of the weld on the test items checked during the manufacture of the shaft fitted to G-CHCN, there was no record of any concessions or rework carried out in the area of the weld on the shafts fitted to G-REDW or G-CHCN.

The dimensional inspection is undertaken to ensure that the component meets the design specification. An inspection document 332A32510100-DI926 lists the design features and specifies the percentage of components that need to be accurately inspected. This document calls for 10% of the countersinks in the hole in the weld to be checked using a replicast and shadow board. The normal practice is to first visually check all the countersinks using a torch and mirror before selecting one of the parts for the replicast inspection. Should the replicast fail the inspection, then a replicast would be made of all the parts in the batch. No record is maintained as to which part in the batch is subject to a replicast inspection.

The average roughness (R_a) of the hole in the weld is established using an automatic process. At the time of the accident there was no documented guidance on the acceptable criteria for scores and scratches in the hole or countersink. The accepted practice was, providing the average roughness was within design limits, to ignore such marks. Following the accident to G-REDW, the manufacturer introduced acceptance criteria to limit scores, in this area, to 5 µm.

The last stage in operation 310 is to fit the PTFE plug into the hole in the weld. The normal practice is to wait until the dimensional inspection has been carried out on all the parts in the batch before fitting the plugs. The plug is fitted when the component is still in the degreased condition and is first softened using a hot air gun before being inserted from the inside of the shaft.

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Appendix C (cont)

10. Packaging and storage

Immediately following the fitting of the PTFE plug, the part is immersed in a bath of protective oil. It is then sent to the packaging ('Conditionnement') department where it is wrapped in 'Volatile' corrosion inhibitor paper before being placed in a cardboard container. The part is then dispatched to the manufacturer's warehouse and retained in storage until required. The EASA Form 1 is issued just prior to the final delivery.

11. Dates when shaft M385 was degreased during the manufacturing process

The operations, and dates, during when the shaft fitted to G-REDW (M385) was degreased and then dipped in a bath of oil, following the drilling of the hole in the weld, are summarized in Table C-1.

Date complete	Activity
16/9/10	Hole drilled in weld
4/10/10	Ultrasonic inspection
25/10/10	X-ray weld
8/11/10	Machining operation, non-weld
11/11/10	Machining operation, non-weld
15/11/10	Niteau inspection
17/11/10	Magnetic particle inspection
23/11/10	Machining operation, non-weld
26/11/10	Dimensional inspection – CMM
3/12/10	Dimensional inspection – manual
9/2/11	Rework – dimensional check, non-weld
20/2/11	Rework – machining, non-weld
28/2/11	Rework – dimensional check
7/3/11	PTFE plug fitted
7/3/11	Packaged

Table C-1

Summary of operations where M385 was degreased and dipped in oil

Appendix D

ENGINE POWER LIMITATIONS

The interaction of the teeth on the pinion and bevel wheel in the MGB causes the bevel gear vertical shaft, which is constrained by the upper and lower bearings, to bend. As the shaft rotates this bending generates an alternating stress in the area of the weld the amplitude of which is dependent on the MGB input torque.

To ensure that the torque through the MGB remains within the design limits, the EC225 LP is cleared to operate at the following operational engine power settings[1]:

- Maximum continuous power (MCP) is the engine power setting that the helicopter can operate at for an unlimited period. On the cockpit torque gauge this represents 85.4% below 40 kt and 82.7% above 60 kt; between 40 kt and 60 kt there is a linear decrease. This engine power setting generates a torque on the bevel wheel of 7,830 Nm ± 5%.
- Takeoff Power (TOP) is the engine power setting that the helicopter can operate at below 45 kt (IAS) for a maximum of 5 minutes. On the cockpit torque gauge this represents 100%. This engine power setting generates a torque on the bevel wheel of 9,474 Nm ± 5%.

Takeoff power transient (TOPtran), also called Max transient, is the maximum engine power that the helicopter can operate at and is limited to a maximum of 2 minutes. On the cockpit torque gauge this represents 110%. This engine power setting generates a torque on the bevel wheel of 9,807 Nm ± 5%.

[1] The power from the engines is used to drive the bevel wheel, tail rotor system and ancillaries. The torque values were achieved from calculation and differ from the TOPtran in the fatigue substantiation document which was obtained from flight test data.

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Appendix E

RECOMMENDATIONS FROM CAA CAP 1144 *'ADELT REVIEW REPORT'*

No.	***Recommendation***
1	*It is recommended that EASA develop guidance material to assist designers of future ADELTs/CPIs and aircraft ADELT/CPI installations to demonstrate compliance with CS XX:1301, 1309, 1529 and 1581. This guidance material should address the issues associated with:* *a. Determination of the appropriate location of an ADELT/CPI with respect to the transport joint and the main rotors to maximise the likelihood of ADELT deployment and transmission.* *b. Installations that could compromise emergency exits or any safety related functions or parts of the aircraft to ensure that overall airworthiness is maintained and that the likelihood of passenger survival is not decreased.* *c. Appropriate selection and location of activation sensors to take account of the functional capabilities and the intended role of the aircraft (e.g. environments where the aircraft will be operated, especially if hostile, and whether the aircraft has flotation equipment).* *d. The location and type of power supplies for all elements of an ADELT/CPI system to maximise the likelihood of ADELT deployment and transmission during an accident or incident.* *e. Mitigation of any relevant human factors issues including:* *1. Flight deck human factors issues (such as inadvertent activation/arming) and* *2. Maintenance human factors issues (such as misleading or incomplete maintenance instructions).* *f. The type and frequency of required maintenance tasks to ensure that all elements of the ADELT/CPI system are appropriately maintained.* *g. Labelling on ADELT/CPI component parts (e.g. batteries) to ensure that any necessary labels (e.g. usage instructions or battery shelf life information) are appropriately visible, unambiguous and permanent to minimise the possibility of incorrect usage.*
2	*In the light of the recent accidents in the North Sea where ADELTs/CPIs failed to deploy, it is recommended that EASA consider the need to re-evaluate current ADELT installations that are being carried by rotorcraft known to be operating in a hostile environment.*
3	*In the light of the recent accidents in the North Sea where ADELTs/CPIs failed to deploy, it is recommended that, where rotorcraft are being operated in a hostile environment, the operators of those rotorcraft include an evaluation of the suitability of their current ADELT/CPI installations for the intended function and operational environment of their rotorcraft as part of their SMS risk assessment process.*
4	*It is recommended that EASA develop specific design requirements for ADELTs (e.g. an ETSO) based on the content of CAA Specification 16, ED-62A and the recommendations of this report.*
5	*It is recommended that designers of flight recorders and aircraft flight recorder installations consider re-locating rotorcraft flight recorders into a part of the rotorcraft that is not subject to tail break issues and/or consider the use of deployable memory.*

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Appendix F

OIL SAMPLING

As part of the investigation into the accidents involving G-REDW and G-CHCN, in November 2012, oil samples were taken from the MGBs of six EC225 LP helicopters based at Aberdeen. Samples were not available from the remaining EC225 LP helicopters as the oil had been removed in order for the operators to inspect the bevel gear vertical shafts. Two reference samples were taken from fresh samples of Aerogear 823 oil that was used in the gearboxes.

The results of three of the samples, summarised in Table F-1, showed levels of iron above the manufacturer's advisory limits. On G-REDV and G-REDR the amount of iron was above the 'early warning' threshold of 13.5 ppm, but below the manufacturer's warning threshold of 18 ppm. The water content in both these samples was similar to the water content in the reference samples. The sample from G-CHCJ appeared to be darker than the other samples and contained 33.8 ppm of iron, which was above the 'warning' threshold of 18 ppm. The water content in this sample was also an order of magnitude higher than on the other helicopters and the two reference samples. The laboratory that undertook the analysis advised that the Total Acid Number (TAN), when taken in isolation, was unlikely to cause excessive corrosion within the MGB.

Helicopter	**MGB** S/N	**MGB life** (Hours)	**Shaft** S/N	**Shaft** (Hours)	**Oil life** (Hours)	**Iron** (ppm)	**TAN** (mg KOH/g)	**H_2O** KF(%)
G-REDV	M5180	1,380	M299	1,381	563	15.2	0.10	<0.01
G-REDR	M5195	754	M5582	446	440	15.4	0.11	<0.01
G-CHCJ	M1505	2,207	M183	2,360	63	33.8	0.21	<0.11
Reference Samples	-	-	-	-	-	<0.1	0.22 /0.28	<0.01

Table F-1

Summary of oil samples from MGB found above advisory limits

Appendix G

STRESS

1. Definitions

Yield strength

Yield strength is the stress corresponding to the point above which a material will permanently deform.

Ultimate Tensile Strength

Ultimate Tensile Strength (UTS) is the level of stress at which a material will fail when subject to a tensile force.

Fatigue

Fatigue is the cracking of a component as a result of a cyclic tensile stress at levels below the UTS. Failures that occur below 100,000 cycles are typically called low-cycle and those above 100,000 cycles are called high-cycle fatigue.

Fatigue limit / endurance limit

From fatigue testing, the stress (S) required to cause the failure of a material is plotted against the number of cycles (N) required to produce an S-N curve, which is also known as the Wöhler curve. The minimum stress on this curve is known as the fatigue limit or endurance limit (σ_e) of the material, below which the material should not fail in fatigue. However, the curve is normally plotted through the mean values at each stress range and therefore there will be some scatter about this curve. The S-N curve for a specimen of 32CDV13 steel, with a surface roughness of 1.6 µm, used in the design of the bevel gear vertical shaft is reproduced at Figure G-1. This figure shows a mean fatigue limit of ± 642 MPa. When other factors such as a higher surface roughness and defects such as corrosion pits and scratches, are taken into consideration, the fatigue limit could be significantly lower than ± 642 MPa.

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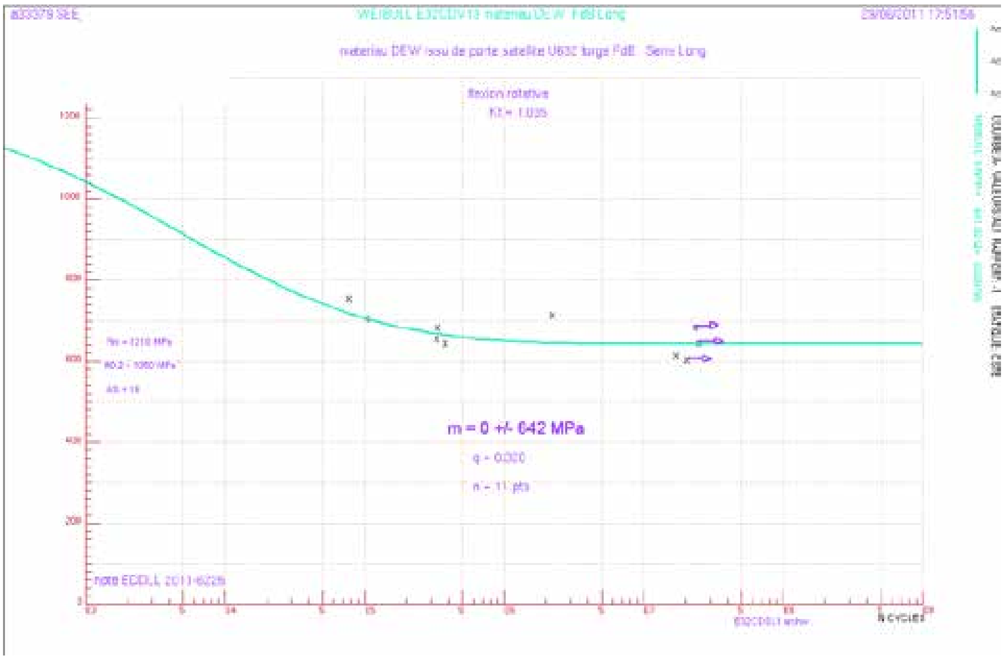
Appendix G (cont)



(*Courtesy of Eurocopter)*

Figure G-1

S-N curve for 32CDV13 steel

Stresses

When considering fatigue damage it is necessary to consider the maximum alternating tensile stress σ_a and the mean stress σ_m. At Figure G-2, tensile stress is plotted as a positive value and compressive stress as a negative value. The helicopter manufacturer refers to σ_m as σ_{stat} and σ_a as σ_{dyn}.

$$\sigma_a = (\text{maximum stress } (\sigma_{max}) - \text{minimum stress } (\sigma_{min})) / 2$$

$$\sigma_m = (\text{maximum stress } (\sigma_{max}) + \text{minimum stress } (\sigma_{min})) / 2$$

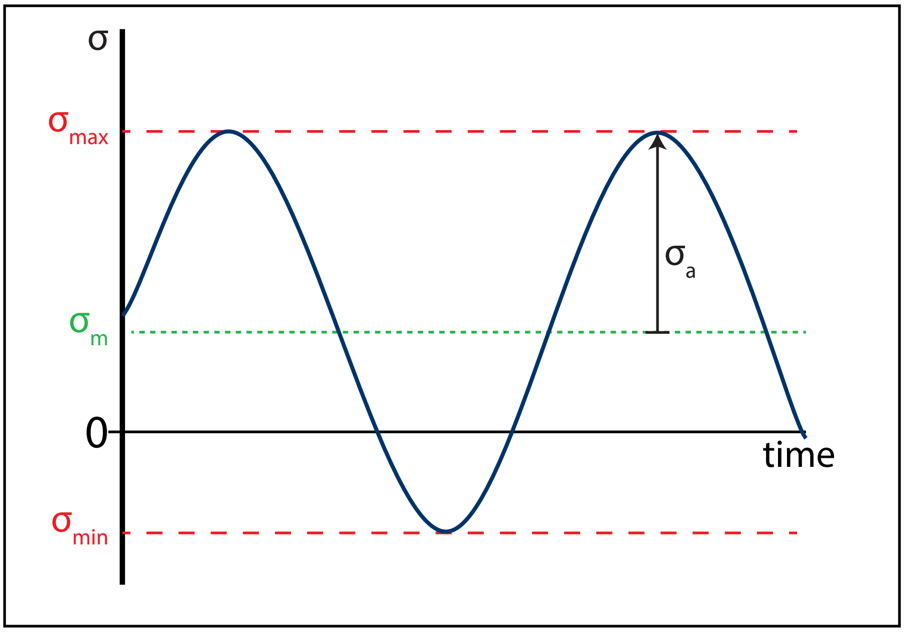


Figure G-2

Definition of alternating and mean stress

The R ratio is defined as the minimum stress divided by the maximum stress ($R = \sigma_{min} / \sigma_{max}$). An R ratio of -1 means that the load is fully reversible with zero mean stress.

2. Effect of alternating and mean stress on fatigue failure

The effect of the alternating and mean stress can be seen on the Goodman and Gerber Diagram at Figure G-3. For high strength steels the fatigue limit (σ_e) is approximately half of the UTS at zero mean stress, which is plotted on the vertical axis. The UTS is plotted on the horizontal axis and the Goodman and Gerber lines are drawn between these points. If the combination of position of σ_a and σ_m, for example as represented by σ_p in Figure G-3 falls below these lines then the component is unlikely to fail in fatigue. Defects such as corrosion or scratches will lower the fatigue limit, which will change the gradient of the Goodman Line and lower the Gerber line. The effect on the Goodman line is seen in Figure F-3. The manufacturer uses the Gerber line when considering the fatigue behaviour of high strength steels such as 32CDV13.

Appendix G (cont)

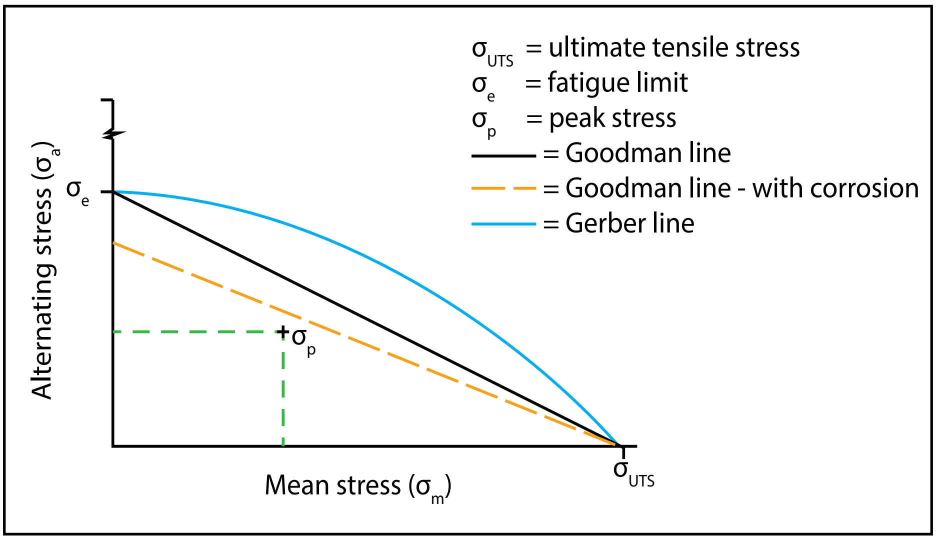


Figure G-3

Goodman and Gerber diagram

The equations for the Goodman and Gerber lines are:

Goodman line:-

$$\frac{\sigma_a}{\sigma_e} + \frac{\sigma_m}{\sigma_{UTS}} = 1$$

Gerber line:-

$$\frac{\sigma_a}{\sigma_e} + \left(\frac{\sigma_m}{\sigma_{UTS}}\right)^2 = 1$$

3. Residual stress

Residual stresses are stresses that are present within a component in the absence of any external load and must exist as both tensile and compressive stresses. These residual stresses balance out within the component and can have a positive or negative impact on the fatigue life. Tensile residual stresses cause the mean stress to increase, with a resulting reduction in the fatigue life, whereas compressive residual stresses can improve the fatigue life and the resistance of the material to failure. The residual stresses can have an effect on both initiation and crack propagation stages of fatigue. The propagation of a fatigue crack can alter the distribution of the residual stresses within a component.

Appendix G (cont)

Manufacturing processes are the most common cause of residual stress and welding is known to cause residual stresses to form around the melted zone as a result of the weld material cooling and contracting. Electron beam welding uses a high intensity beam as the power source, which has the advantages that there is a relatively low heat input and distortion of the component, and therefore lower levels of residual stress.

During this investigation, the manufacturer identified a possible relationship between the force used to clamp both parts of the shaft together prior to welding and the level of residual stress in the component. However, tests proved that this was not the case.

4. Fracture mechanics

Fracture mechanics is the study of the propagation of cracks under static and cyclic loading in structures and components. It uses models to estimate the maximum crack or defect length that a material can withstand before the crack becomes unstable and grows to failure.

The stress intensity factor K is used to characterise the stress state near the tip of a crack under a remote stress σ. K is dependent on the geometry of the component, the size (a) and location of the crack and the magnitude and distribution of stresses within the component. In general terms:

$$K = \sigma \sqrt{\pi a} \beta$$

where β is a factor dependent on the component geometry and loading inputs. Any combination of crack length, stress and β producing the same value of K will have the same crack driving parameter.

In fatigue cracks the Paris Law relates the change in the stress intensity factor ΔK, with the fatigue crack growth rate (da/dN). ΔK can be regarded as the fatigue crack driving parameter. Paris Law is expressed as:

$$\frac{da}{dN} = C \Delta K^{m}$$

where:

a is the crack length,
N is the number of cycles,
ΔK is the change in the stress intensity factor,
C and m are empirical constants for the material in question.

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Appendix G (cont)

Figure G-4 shows a generic plot of the log of da/dN against log ΔK where it can be seen that Paris Law holds true for the centre part of the graph with deviations at low and high crack growth rate. Accurate fatigue life calculations can be made by integrating the curve between the start and finish of the crack length.

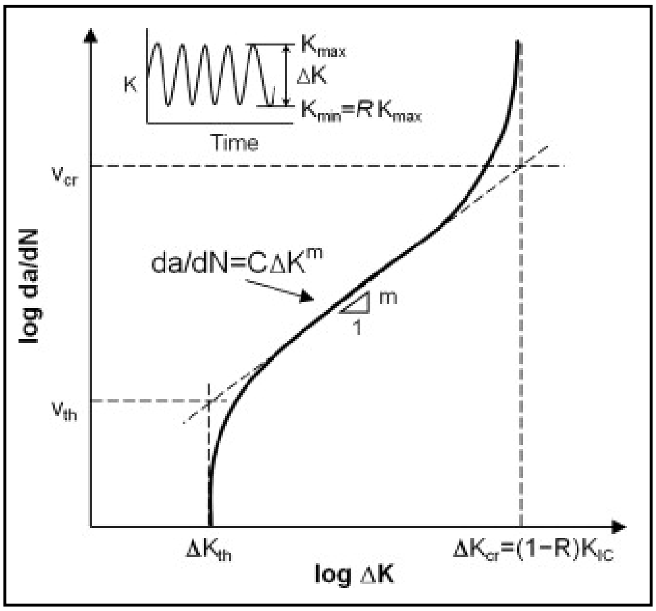

Figure G-4

Generic graph showing Paris Law

Another practical use of the relationship between fatigue crack growth rates and ΔK is that as ΔK is proportional to the torque applied to the helicopter gearbox (ΔK is proportional to (Torque2 / Torque1)^m) it can be used to predict how the crack growth rate will change at different power settings. For 32CDV13 steel the constant m is 2.45. Therefore increasing the torque by 15% would increase the rate that the crack grows by 40% and decreasing the torque by 15% would decrease the rate that it grows by 33%.

ΔK_{th} is the stress intensity threshold below which a fatigue crack will not grow. The threshold value marks the boundary between growing cracks and cracks which will not grow. Its value can be used to predict which combination of stress and crack length (or equivalent defect size) will grow and which will be benign and will not grow under service stresses.

While the simple fracture mechanics model described above is accurate for cracks of greater than 1 mm in length, it is found experimentally that as cracks and defects become smaller than 1 mm their behaviour can deviate from the

Appendix G (cont)

long crack model. The consequence is that short cracks can grow faster and have reduced threshold values compared with long cracks. For high strength steel, cracks less than 0.1 mm (100µm) in length can be considered to be short. The initiation and growth of the fatigue cracks from the 60 µm deep corrosion pits in the shafts fitted to G-REDW and G-CHCN will come into the short crack regime. Consequently, the use of long crack data to calculate their fatigue lives may produce unrepresentative results.

The growth of short cracks is less predictable than long cracks and the rate of growth can accelerate and decelerate as the crack grows. This is due to two reasons. Firstly, the relative size of the length of the crack and the size of the plastic zone at the crack tip means that the plastic region has an influence on the distribution of stress. Secondly, as a small crack grows across a grain its growth rate accelerates and then decelerates as it approaches the grain boundary.

An understanding of the behaviour of the short crack is important as most of the fatigue life of a crack occurs when the crack is small; the region of growth from the 60 µm deep corrosion pits to a crack length of 100 µm falls within the short crack regime.

Appendix H

SIGNIFICANT RESULTS OF MANUFACTURER'S SHAFT TESTS

1. The manufacturer carried out a number of fatigue bending tests to determine the effect of corrosion pits and humidity on the fatigue limit. The results of the significant tests are summarised in this appendix.

2. Test M662

This test used shaft serial number M662 in which two 4 mm holes were drilled in the area of the weld. The manufacturer considered that this test would explore the worst case combination of a high clamping[1] force (10 daNm), corrosion pits and high humidity. The test was started at a load of 2,400 daN. The test programme ran for a number of cycles in between which the load was increased by 300 daN increments. The test was finally stopped after it had run for 220,000 cycles at a load of 3,300 daN.

One hole, identified as 'Hole 1', was drilled in the centre of the weld and pre-corroded in salt water. The second hole, identified as 'Hole 2', was drilled close to the inner radius and subject to a constant drip of tap water throughout the test to simulate a high humidity environment; the temperature at the inner radius was measured at 85ºC. At the end of the test, two small cracks were detected in the area of the inner radius close to 'Hole 1' and six cracks in the area of 'Hole 2'. Three of the cracks started on the edge of the 4 mm hole, one on the countersink and one close to the inner radius in the parent material.

The cracks detected on the inner radius close to 'Hole 1' were 2.7 mm and 1.78 mm long. Both fracture surfaces displayed characteristic signs of fatigue and there was some evidence of inter-granular cracking towards the end of the cracks. This inter-granular cracking indicates that corrosion fatigue may have been present towards the end of the crack. Striations with spacing of approximately 0.1 µm were also present. Corrosion pits were identified along the machining marks, near 'Hole 1', and both cracks had initiated at a corrosion pit approximately 30 µm deep and, 200 µm and 300 µm wide. While there was no evidence of corrosion on the fracture surface, corrosion products were found in the corrosion pit. The manufacturer established that the estimated fatigue limit at the inner radius close to 'Hole 1' was ± 249 MPa and estimated that crack initiation occurred after 200,000 cycles at a load of 3,300 daN. The features on the fracture surfaces were similar to those seen on G-REDW and G-CHCN. However, the spacing of the striations on the test shaft was larger

[1] The clamping force is the torque used to clamp the two parts of the shaft together before they are welded. See Appendix C.

Appendix H (cont)

than those experienced on G-REDW and G-CHCN. This was because the test loads were greater than the flight loads and consequently the crack growth rate would have been faster.

The fracture surface of the cracks in the area of 'Hole 2' displayed signs of fatigue. There was also a high ratio of inter-granular to trans-granular cracking along the cracks consistent with corrosion fatigue [2]. All the cracks initiated from corrosion pits approximately 40 µm deep. However, the surface had been highly damaged by corrosion and it was difficult to measure the width of the corrosion pits; it was also not possible to observe any striations. The manufacturer established that the estimated fatigue limit at the inner radius close to 'Hole 2' was ± 224 MPa and on the inner countersink ± 300 MPa. The crack initiation was estimated to have occurred after 100,000 cycles at a load of 3,300 daN. Overall the features on the cracks from 'Hole 2' were different to those seen on the fracture surfaces on G-REDW and G-CHCN.

3. Test M664

This test used shaft M664 and electro arcing to create simulated corrosion pits in the inner radius close to two 4 mm holes: these pits had a smoother profile compared to real corrosion pits. The temperature close to the inner radius was maintained at approximately 85ºC, the clamping load was 2 mdaN and the shaft was subject to the normal humidity in the laboratory. The test was started at a load of 2,400 daN. The load was then increased incrementally by 300 daN and was stopped after running for 86,751 cycles at a load of 4,500 daN.

During the test two cracks initiated from the simulated corrosion pits that were approximately 120 µm deep and, 950 µm wide. The manufacturer established that the estimated fatigue limit was ± 332 MPa and the cracks initiated after 260,000 cycles at a load of 4,200 daN.

4. Test M175

This shaft was pre-corroded with salt spray and the test was run without the addition of humidity. A crack initiated at a corrosion pit 70 µm deep, located on the inner radius. The profile of this corrosion pit was similar to the corrosion pits found on G-CHCN. The estimated fatigue limit resulting from this test was established as ± 282 MPa.

[2] Inter-granular cracking is normally associated with corrosion fatigue; in comparison trans-granular cracking is normally associated with fatigue.

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Appendix H (cont)

5. Test M787

This shaft was pre-corroded with salt spray and the test was run without humidity. A crack initiated at a corrosion pit 34 μm deep, located on the inner radius. The estimated fatigue limit resulting from this test was established as ± 358 MPa.

6. Test M225

This shaft was pre-corroded with salt spray and the test was run without the addition of humidity. A crack initiated at a corrosion pit 60 μm deep, located on the inner radius. The estimated fatigue limit resulting from this test was established as ± 300 MPa.

7. Test M041

This shaft was pre-corroded with salt spray and the PTFE plug was wetted before being inserted into the 4 mm hole. The test was run in a humid atmosphere. A crack initiated at a corrosion pit 40 μm deep, located in the 4 mm hole. The estimated fatigue limit resulting from this test was established as ± 292 MPa.

8. Tests on other bevel gear vertical shafts

The manufacturer carried out tests on a number of bevel gear vertical shafts and used the results from shaft M391 and M422 to establish the fatigue limit used in the fatigue substantiation document (Issue E) produced following the accidents involving G-REDW and G-CHCN.

9. Test M391

Shaft M391 was a new shaft that came from the same batch as shaft M385 that was fitted to G-REDW. The part was degreased prior to the strain gauges being fitted and the test was carried out in laboratory conditions without the addition of any additional humidity. The test was stopped when a fatigue crack was observed to grow from a corrosion pit, 23 μm deep, in the 29 mm hole. No cracks were detected at the 4 mm hole, or inner radius. From this test the manufacturer calculated the fatigue limit of the weld as ± 670 MPa and the parent material as ± 483 MPa.

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Appendix H (cont)

10. Test M422

Shaft M422 was a new part that was degreased prior to the strain gauges being fitted. The 4 mm hole was exposed to a salt atmosphere in a climatic chamber for a period of 84 hours prior to the start of the test. The salt solution was not applied to the 29 mm hole. At the end of the test, three cracks were identified as initiating from corrosion pits on the 4 mm hole and one crack from a corrosion pit on the 29 mm hole, 20 µm deep. From this test the manufacturer calculated the fatigue limit of the weld from one of the cracks that initiated from a corrosion pit 2 to 5 µm deep, as ± 629 MPa. The fatigue limit of the parent material was calculated as ± 453 MPa.

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Appendix I

TEXT OF NPRM 29.571

NPRM 29.571 text taken from CRI-04 as applied by Eurocopter to the EC225 LP bevel gear vertical shaft:

> *'(a) General. A fatigue tolerance evaluation of the principal structural elements (PSEs) identified in paragraph (c) below must be performed and appropriate actions established, as defined in paragraph (g), to avoid catastrophic failure during the operational life of the rotorcraft. This evaluation must consider the effects of both fatigue and the damage determined in paragraph (d)(3) below. Parts to be evaluated include, but are not limited to, rotors, rotor drive. Systems between the engines and rotor hubs, controls, fuselage, fixed and movable control surfaces, engine and transmission mountings, landing gear, and their related primary attachments.*
>
> *(b) The compliance methodology must be submitted to the regulatory authority for approval.*
>
> *(c) Considering all structure and/or structural elements and assemblies, the PSEs must be identified. PSEs are structural elements that contribute significantly to the carrying of flight or ground loads and the fatigue failure of which can lead to catastrophic failure of the rotorcraft.*
>
> *(d) Each evaluation required by this section must include:*
>
> > *(1) In-flight measurement in determining the fatigue loads or stresses for the PSEs identified in paragraph (c) above in all critical conditions throughout the range of limitations in § 29.309 (including altitude effects), except that manoeuvring load factors need not exceed the maximum values expected in operations.*
> >
> > *(2) Loading spectra as severe as those expected in operation based on loads or stresses determined under paragraph (d)(l) of this section, including external load operations, if applicable, and other high-frequency power-cycle operations.*
> >
> > *(3) A determination for the PSEs identified in paragraph (c) above of the probable locations, types, and sizes of damage considering fatigue, environmental effects, intrinsic/discrete flaws, or accidental damage that may occur during manufacture or operation.*

Appendix I (cont)

(4) A determination of the fatigue tolerance characteristics for the PSEs with the damages identified in paragraph (d)(3) above to support accomplishment of paragraph (g) below.

(5) Analyses supported by test evidence and/or service experience.

(e) A residual strength determination is required to establish the allowable damage size. For inspection interval determination based on damage growth, the residual strength requirement is limit load considered as ultimate.

(f) The effect of damage on stiffness, dynamic behaviour, loads and functional performance must also be considered.

(g) Based on the evaluations required by this section, inspections and retirement times or approved equivalent means must be established to avoid catastrophic failure. The inspections and retirement times or other means as applicable must be included in the airworthiness limitation section of the Instructions for Continued Airworthiness required by Section 29.1529 and Section A29.4 of Appendix A of this part.

(h) If inspections for any of the damage types identified in part (d) (3) above cannot be established within the limitations of geometry, inspectability, or good design practice, then supplemental procedures, when available, must be established that will minimize the risk of each of these types of damage being present or leading to catastrophic failure.'

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Unless otherwise indicated, recommendations in this report are addressed to the appropriate regulatory authorities having responsibility for the matters with which the recommendation is concerned. It is for those authorities to decide what action is taken. In the United Kingdom the responsible authority is the Civil Aviation Authority, CAA House, 45-49 Kingsway, London WC2B 6TE or the European Aviation Safety Agency, Postfach 10 12 53, D-50452 Koeln, Germany.