Civil Aviation Authority

Safety Regulation Group

CAA Paper 2008/05

HUMS Extension to Rotor Health Monitoring

www.caa.co.uk

Civil Aviation Authority

Safety Regulation Group

CAA PAPER 2008/05

HUMS Extension to Rotor Health Monitoring

March 2009

ISBN 978 0 11792 235 8

Published December 2008
Second Edition March 2009

Enquiries regarding the content of this publication should be addressed to:
Research and Strategic Analysis, Safety Regulation Group, Civil Aviation Authority, Aviation House, Gatwick Airport South, West Sussex, RH6 0YR.

The latest version of this document is available in electronic format at www.caa.co.uk/publications, where you may also register for e-mail notification of amendments.

Published by TSO (The Stationery Office) on behalf of the UK Civil Aviation Authority.

Printed copy available from:
TSO, PO Box 29, Norwich NR3 1GN
Telephone orders/General enquiries: 0870 600 5522
Fax orders: 0870 600 5533
www.tso.co.uk/bookshop
E-mail: book.orders@tso.co.uk
Textphone: 0870 240 3701

List of Effective Pages

Part	Page	Date	Part	Page	Date
Appendix A	39	March 2009			
Appendix A	40	March 2009			
Appendix A	41	March 2009			
Appendix A	42	March 2009			
Appendix B	1	March 2009			
Appendix B	2	March 2009			
Appendix B	3	March 2009			
Appendix B	4	March 2009			
Appendix B	5	March 2009			
Appendix B	6	March 2009			
Appendix B	7	March 2009			
Appendix B	8	March 2009			
Appendix B	9	March 2009			
Appendix B	10	March 2009			
Appendix B	11	March 2009			
Appendix B	12	March 2009			
Appendix B	13	March 2009			
Appendix B	14	March 2009			
Appendix B	15	March 2009			
Appendix C	1	March 2009			
Appendix C	2	March 2009			
Appendix C	3	March 2009			
Appendix C	4	March 2009			
Appendix C	5	March 2009			
Appendix C	6	March 2009			
Appendix C	7	March 2009			

Contents

INTENTIONALLY LEFT BLANK

Revision History

Edition 1 **December 2008**

First issue.

Edition 2 **March 2009**

The paper was re-issued to reflect corrections to paragraph 3.2.2.3 on pages 34 and 35 of the report.

INTENTIONALLY LEFT BLANK

Foreword

The research reported in this paper was funded by the Safety Regulation Group of the UK Civil Aviation Authority (CAA), Oil & Gas UK and Shell Aircraft Ltd, and was performed by GE Aviation (formerly Smiths Aerospace). The work was primarily commissioned in response to industry concerns expressed following the fatal accident to Sikorsky S-76 G-BJVX near the Leman 49/26 F platform in the southern North Sea on 16 July 2002 which was caused by a main rotor blade failure. A secondary motive was to establish a consolidated position in relation to rotor system health monitoring by, inter alia, reviewing and publishing the earlier work performed by MJA Dynamics under contract to the CAA.

The CAA accepts the findings of this study. Although the review of the MOR data clearly indicates that rotor system defects are less of a problem now than when the earlier studies were commissioned at MJA Dynamics, events such as the more recent Super Puma main rotor blade spindle failure in October 2006 serve to raise awareness of the seriousness of such defects.

GE Aviation have consequently been tasked with a study aimed at demonstrating the application of the anomaly detection methods – successfully developed and applied to HUMS transmission data by GE – to the rotor system data already routinely collected on in-service helicopters. In view of the evidence presented in the review of helicopter accident and incident case studies, the scope of this work has been restricted to tail rotor systems.

As regards main rotor systems, consideration is being given to commissioning a PhD-based study into the dynamics of a flawed rotor system with the aim of establishing which faults may be detectable in principle, and how they might best be detected. The study would also be used to investigate the possibilities for the positioning of sensors in both the fixed and rotating frames.

Safety Regulation Group
December 2008

INTENTIONALLY LEFT BLANK

Executive Summary

This report contains the results of a status review of helicopter rotor health monitoring, commissioned by the United Kingdom Civil Aviation Authority (CAA). The review was motivated by the perception that, whilst drive train health monitoring is a mature technology and an integral part of a helicopter Health and Usage Monitoring System (HUMS), the same cannot be said for rotor system health monitoring, despite the fact that the numbers of helicopter accidents caused by rotor failures and drive train failures are comparable. Although rotor vibration is monitored by HUM systems and used for Rotor Tracking and Balancing (RTB) purposes, there are currently few recognised techniques for the isolation and identification of rotor fault induced vibration, such as exist for gear train vibration. Previous research work conducted in the area was largely on a theoretical, analytical basis. Although it was considered to have shown some potential, there was no further development towards a working demonstration or a practical rotor monitoring system.

The objectives of the study were, firstly, to conduct a review of the relevant previous rotor health monitoring research to form a view of the potential for fault detection and identify the most promising methods to pursue for future development. Secondly, an analysis of the available data on accidents and incidents due to rotor system failures was performed in order to assess the potential safety benefit of applying new HUMS techniques to rotors. The information from these exercises was consolidated to form a view of the most promising strategy to adopt for future development towards a working system, and to recommend a programme of work to be performed as the next phase in this development.

The review of previous research showed that there were some possibilities for fault detection, but results at the time the work was concluded did not give sufficient confidence that comprehensive fault detection would be possible. The accident analysis showed that there is potential safety benefit in improved main and tail rotor fault detection, and that at least some rotor fault mechanisms may be detectable through existing vibration measurements. However, both studies indicated that, for a number of faults, it is unlikely that it would be possible to detect these at an early enough stage to provide meaningful safety benefits when using existing fixed frame HUMS vibration measurements. This, together with the general decline in the rotor-related accident rate revealed by the statistics, suggests that a costly and speculative research programme into new technologies would not be appropriate.

The review has determined that the realisation of an effective HUMS capable of detecting rotor faults requires further work in some or all of the following areas:

i) The extension, and trial using HUMS data, of the methods proposed and partially demonstrated through work commissioned by CAA in 1991 at MJA Dynamics Limited (MJAD), now GE Aviation.

ii) The investigation of the potential use of vibration data acquired during unsteady flight conditions to aid the detection of rotor system catastrophic faults.

iii) The investigation of the emergent rotating-frame sensing technologies, including data transfer from the rotor system to the non-rotating fuselage equipment.

iv) The investigation of the application of anomaly detection techniques to existing rotor vibration data from operational HUM systems.

v) Combined with the analysis of HUMS operational data, the use of current rotorcraft modelling techniques to further investigate rotor induced vibration, endorse the theory that rotor faults are manifested in the vibration, determine how the faults might be detected and investigate how various detection techniques could be fused together to obtain more robust diagnostic results.

To advance the state-of-the-art in rotor health monitoring in the most cost effective manner, it is recommended that items (iv) and (v) are progressed as near term tasks, and that item (iv) focuses on the anomaly detection methods successfully developed and applied to HUMS transmission data on a related CAA research programme.

Glossary

AAIB	Air Accidents Investigation Branch
ADC	Automatic Data Correction
AI	Artificial Intelligence
BIM	Blade Inspection Method
FDR	Flight Data Recorder
FUMS™	Fleet Usage Management System (Trade Mark)
HMMs	Helicopter Mathematical Models
HUMS	Health and Usage Monitoring System
JSF	Joint Strike Fighter
MJAD	MJA Dynamics Limited
MoD	The UK Ministry of Defence
MOR	Mandatory Occurrence Reporting
NDT	Non Destructive Testing
PC	Personal Computer
PCFs	Potentially Catastrophic Faults
RTB	Rotor Track and Balance
SFFT	Short Fast Fourier Transform
UK CAA	United Kingdom Civil Aviation Authority
UMARC	University of Maryland Advanced Rotorcraft Code
UoG	University of Glasgow
UoM	University of Maryland

INTENTIONALLY LEFT BLANK

Report HUMS Extension to Rotor Health Monitoring

1 Introduction

1.1 Background

The development and implementation of HUMS for helicopters has demonstrated the advantages that can be gained from transmission system vibration analysis through the early detection of Potentially Catastrophic Faults (PCFs). HUMS transmission technology has now matured into an effective system that provides significant safety benefits, but rotor system vibration monitoring is not so advanced. Although rotor system vibration is monitored for the purposes of performing RTB adjustments, the rotor fault detection aspect of HUMS has not been adequately investigated. This is despite the general consensus that the potential for component failures leading to helicopter accidents or incidents can be roughly equally split between rotor and transmission failures.

Prior to the MJAD work commissioned by CAA in 1990, there was little investigation into the potential for the detection of rotor system PCFs. The reasons for this include:

i) An analysis technique had not (and still hasn't) been widely realised for rotor system PCFs.

ii) A general misunderstanding that by extending RTB techniques, the majority of PCFs can be detected or, worse still, that RTB and PCF diagnostics are one and the same. On the contrary, it has been established through the MJAD work and in-service incidents that adjustments made as a result of RTB measurements may mask PCFs rather than indicating their presence.

In order to help address this imbalance, and after a literature survey concluded in 1989, in 1990-1991 CAA commissioned a programme of research at MJAD, now part of GE Aviation. The CAA programme was intended to develop and evaluate appropriate strategies for the detection and diagnosis of helicopter main rotor system faults. The research used mathematical dynamic models and in-service operational data to investigate and demonstrate the potential for fault detection via various diagnostic strategies, including model-based and machine learning techniques. A review of accidents caused by rotor system failures was also included. The work was originally intended to form the first phase of a three-phase programme, Phase 2 being rig testing with seeded faults and Phase 3 a full-scale demonstration. Phases 2 and 3 were not progressed, however, as all the research funding available at the time was directed towards seeded defect tests in support of a transmission HUMS demonstration. This was not due to any perceived deficiency in the first phase of the rotor HUMS work and was likely due to the understandable focus on transmission HUMS following the Chinook fatal accident in 1986.

Further work along similar lines was conducted at a later date at the University of Maryland (UoM), and also appeared to show some potential in the techniques, though entirely on a theoretical basis. Again, there did not appear to be any follow-on work directed towards development of a full-scale rotor system fault detection capability.

The S76 fatal accident near the Leman field in the southern North Sea in 2002 (Reference [1]), however, refocused attention on the area of rotor system health monitoring and led to the study covered in this report.

1.2 Objectives

This report describes a rotor health monitoring status review performed in order to assess the current state of the art and provide recommendations for future development. The objectives of the status review were to:

i) Conduct a review of all relevant work (including the earlier MJAD research) in order to form a consolidated view of the state of the art of the application of rotor vibration and blade position measurement techniques to the detection of potentially catastrophic rotor system failures.

ii) Provide a clear indication of:

  a) The potential safety benefit of applying new HUMS techniques to rotors.

  b) The development work that would be required, culminating in the fielding of a demonstration system for in-service trials.

The key tasks undertaken for this study were as follows:

i) A review of work undertaken by MJAD in 1990-1991, and a review of relevant published literature since then, including that produced as a result of the work conducted at the University of Maryland (UoM) between 1995 and 2002 (Section 2 and Appendices A and B).

ii) A review of accidents and incidents caused by rotor failures to update the earlier review performed by MJAD, and to assess the potential safety benefit and likely health monitoring requirements of a rotor HUMS (Section 3 and Appendix C).

iii) An assessment of the most promising strategies for the detection and diagnosis of helicopter rotor system faults, and the production of an outline definition of the next phases recommended for a rotor system HUMS research and development programme (Section 4).

1.3 Project Team

The project was managed by GE Aviation. In order to benefit from current expertise in the field of rotorcraft modelling, the Aerospace Engineering Department at the University of Glasgow (UoG) was sub-contracted to perform an independent review of the previous rotor fault detection research.

The project team consisted of:

GE Aviation	Mrs S Salter and Mr B D Larder
University of Glasgow	Professor R E Brown

2 Review of Rotor HUMS Research

2.1 Literature Search

The starting point for the review was the set of reports produced by MJAD for the CAA programme of work (References [2] to [7]), and the papers published by UoM (References [8] to [21]). In addition, the use of the Science Citations Index and Engineering Compendex databases with keywords "HUMS", "Chopra", "Ganguli" and "Rotor" produced some additional material specifically related to the UoM study. Tracing back through the reference chain at the end of each article also yielded useful leads.

2.2 Overview

The development of a successful HUMS for the detection and identification of potentially catastrophic rotor faults in helicopters is heavily reliant on the ability to extract the signs of incipient failure from the wealth of data that can be obtained from the system. The task is made extremely complicated by the fact that the aerodynamics of the rotor, and hence the details of its loading, vary considerably with flight condition. The rotor systems are directly exposed to wind, moisture, sand storms, ice and foreign objects, which can induce changes in structural and aerodynamic characteristics with consequent variations in loads and deflections between individual blades and helicopters. Hence, any successful rotor HUMS would have to be capable of distinguishing any rotor damage-induced effects on its input data from naturally very wide variations in the properties of the input data due to operation, usage and/or age.

Two previous studies have used mathematical models to investigate methods to achieve this. The earlier study, conducted by MJAD, focused on identifying so-called 'fault patterns' from rotor hub and vibration measurements at various locations within the fuselage of the helicopter and, despite problems in identifying unique signatures for some rotor faults, demonstrated that fault detection through diagnosis of fuselage vibration could be feasible. A later study, conducted by Chopra's research group at UoM, adopted a somewhat similar approach and demonstrated how suitable analysis of the vibration signatures produced at the rotor hub, as generated by their computational model with various degrees of simulated damage, could be used to detect the presence of the damage within the rotor system. For some faults, the principal difficulty encountered was that of determining a unique signature that would allow robust detection of the fault in the presence of multiple faults, operational effects and/or extraneous noise.

Indeed, the principal lesson that can be synthesized from the two studies is that the inherently complicated, non-linear, and flight condition-dependent nature of the dynamics of the helicopter rotor makes the problem of robust and reliable detection of all potential faults within the system, followed by discrimination between the various types of fault that might be present, very difficult. As a result, both the MJAD and UoM studies did not yield a conclusive answer to the key question as to whether implementation of a rotor HUMS would be practical or cost effective, or would be compromised by this fundamental issue. However, the MJAD work suggested and demonstrated rotor diagnostic strategies and techniques that remain applicable today, with the potential to provide a new rotor HUMS capability.

Both the MJAD and UoM studies advocated the use of machine-learning techniques to aid human operators in the analysis of HUMS data for fault diagnosis. Two fundamentally different approaches to machine learning are possible however, and the MJAD study, in particular, contrasts both of these. In the first, 'supervised' approach, the machine is trained to identify the signatures of specific faults and hence reports the presence of an anomaly in the system only when one of these signatures is discernible within the data. In the second, 'unsupervised' approach, the machine is desensitised to the normal operating condition of the system and reports an anomaly only when deviation from the expected condition of the system is detected.

The MJAD work is reported in a series of technical reports produced for the CAA; the UoM work is published in a number of public-domain theses, journal papers and conference proceedings. An assessment and comparison of the technical approaches adopted in the MJAD and UoM studies is given in the following sections. Section 2.3 presents a summary of the MJAD work, with more detailed information on the MJAD reports given in Appendix A. Section 2.4 presents the results of an independent review of both the MJAD and UoM studies that has been produced by the University of Glasgow (UoG), with a more detailed review of the MJAD reports and UoM publications given in Appendix B.

2.3 Research by MJA Dynamics Limited

Key Researchers: Hesham Azzam and Mike Andrew

2.3.1 General

As mentioned in Section 1.1, prior to the work commissioned by CAA in 1990, the rotor fault detection aspect of HUMS had not been adequately investigated despite the consensus that the potential for component failures leading to helicopter accidents or incidents can be roughly equally split between rotor and transmission failures. This is still the case today despite initiatives such as those of MJAD and UoM; the main reasons are considered to be:

i) The MJAD work commissioned by the CAA only spanned one year and involved one full time researcher.

ii) The CAA research funding available after the MJAD work (1991) was directed towards seeded defect tests in support of transmission HUMS.

iii) Smiths Aerospace (now GE Aviation) acquired MJAD in 1995, and subsequent effort was focussed on the embodiment of HUMS in the UK Ministry of Defence (MOD) helicopter fleet, and on parallel developments of Fleet and Usage Management System (FUMS™) technologies.

iv) The MJAD work was not sufficiently publicised and was not clearly documented; it relied on the readers' prior understanding of fundamentals covering aerodynamics, dynamics, structures and Artificial Intelligence (AI) whilst introducing new concepts and strategies. Hence, the follow-up of this work by other organisations has been difficult.

v) The development task requires fully integrated multi-discipline efforts (aerodynamics, dynamics, structures, Artificial Intelligence, etc.).

vi) The task is extremely complicated because the rotor loads vary considerably with flight condition and rotor systems are directly exposed to wind, moisture, sand storms, ice and foreign objects with consequent effects on structural characteristics and variations in loads and deflections between individual blades and helicopters.

vii) The primary purpose of MJAD's use of comprehensive mathematical models was to aid the establishment of a fault detection strategy for rotor HUMS, not to improve helicopter theory. This emphasis may not have been appreciated, with the consequence being a subsequent focus on mathematical model refinements rather than concentrated efforts on improved fault detection strategies.

Therefore, this section aims to provide a clear and concise description of the research undertaken by MJAD for the CAA. Detailed summaries of the set of reports produced by MJAD covering this work (References [2] to [7]) are presented in Appendix A. This section also summarises some further activities carried out by GE Aviation after the CAA work.

2.3.2 History

The programme of work at MJAD extended over a single year of activity in 1991 and was conducted under contract to the CAA. The output of the work consisted of six progress reports compiled at two-monthly intervals. The work was based on earlier proprietary work to develop a helicopter mathematical model capable of simulating the effects of potentially catastrophic rotor faults (References [22] and [23]).

The work involved the use of representative mathematical models, pre-processing methods and machine learning techniques along with the analysis of in-service data to establish and evaluate diagnostic strategies for helicopter main rotor system faults.

2.3.3 Mathematical Foundation

In the MJAD reports and subsequent publications, it was repeatedly emphasised that the main purpose of the MJAD mathematical models was to support the development of advanced diagnostic strategies, not to use them as part of rotor fault detection methods. As proposed to CAA in 1988 [24], the Helicopter Mathematical Models (HMMs) described in [25] to [30] (1980 to 1986) were considered adequate foundation for the stated purpose.

The HMMs combined a non-uniform induced velocity wake, detailed non-linear aerodynamic data, a non-linear flap-lag-torsion aeroelastic model, a fuselage multi degrees of freedom model and a general trim algorithm. Whilst simplified wake models were included as options, a representative prescribed wake was implemented: the wake was considered to consist of strong tip vortices and near interactive blade vortex sheets. The tip vortices were allowed to contract below the rotor and then expand. The strengths of the tip vortices were related to the flapping response of the blades and the strength of the near interactive wake was determined from the blade loading. The geometry of the tip vortex was established by investigating available flow visualisation results and extending these results to cover the speed range using appropriate mathematical models. Experimental and theoretical observations were used to determine the wake parameters. The model was described and validated in [25]-[30] for single, coaxial and tandem rotors. For example, the induced velocities below a model rotor in hover were predicted at various radial and azimuth locations and found to agree with experimental results. The blade aerodynamic loadings were calculated from non-linear aerodynamic data and validation tests showed good agreement between load predictions and flight test data.

The blade elastic model described the flap-wise, drag-wise and torsional motions and considered full inertia and aerodynamic coupling. An assumed mode method was developed for non-uniform rotor blades; along with modal analyses, the modes were used to evaluate the blade degrees of freedom; validation tests indicated that the model was capable of predicting the lateral flapping angle at low speed; there was only one reported model at the time of publication known to the author which had similar capability; this model did not converge below 0.04 advance ratio. The trim model solved the fully coupled equations of motion for various helicopter types and combined an arbitrary non-uniform induced velocity profile, non-linear aerodynamic data and an arbitrary blade elastic deformation model. The trim algorithm stemmed from a method where closed-form-non-linear equations were developed and solved;

the analytical equations were used to monitor the convergence of the representative scheme, which used local blade aerodynamics, etc.

2.3.4 Fault Simulations

Having chosen representative comprehensive HMMs, MJAD simulated a suite of rotor faults in 1989 and integrated the fault simulations in HMMs, [22]-[23]. MJAD classified a rotor fault as either a primary fault or a combination of primary faults, namely a compound fault. It was postulated that the primary fault influences a single blade property. This represents an ideal case for mathematical modelling; in reality, it is more likely that the compound fault will be encountered as an actual fault can effect multiple blade properties. However, it was demonstrated that decomposing a compound fault into its primary components would effectively facilitate the practical implementation of fault detection strategies.

The faults which were simulated included: adjustable faults (mass imbalance, pitch link errors and tab errors), composite blade delamination/damage, metal blade cracks, swash-plate/control rod faults, binding/jammed hinges caused by bearing faults, a jammed damper causing a rise in root stiffness, crack development in lugs, faults caused by manufacture and operational tolerances, damper faults, chord-wise mass imbalance, and an example of a component fault initially developing in one element of a component and then progressively affecting the complete component (this required a staged mathematical model to be applied).

Simulations of blade aerodynamic faults were relatively straightforward; the simulations induced deviations in the aerodynamic coefficients of faulty blades through an 'individual blade concept' where aerodynamic characteristics, geometric configurations and structural properties were allowed to vary across blades and, hence, the contribution of each individual blade to the induced vibration could be evaluated. Tabs are introduced in helicopter blades in order to change the local lift and pitching moment characteristics without incurring major drag penalties and, thus, provide a means of attenuating undesired 1R vibration. The aerodynamic coefficients associated with a tab adjustment are usually deduced from experimental data. A blade strip may be corroded or damaged; a strip fault is expected to cause deviations in the lift and pitching moment coefficient as well as the drag coefficient; the values of the deviations depend on how the strip deforms.

Metallic component cracks or composite structure damage have been found to induce stiffness reductions. In order to simulate these faults, a "Dirac Delta" model was developed where the distribution of the flexural rigidity was modified at the damage position; the coefficient of the "Dirac Delta" gave a measure of the damage severity. This approach did not require substantial modification to the original HMMs and does not entail any computational penalties.

Hydraulic dampers typically operate initially as viscous dampers until a "blow off" lead-lag velocity is exceeded whereupon the damping characteristics are more frictional in nature. The lead-lag response was therefore evaluated using a model simulating undamped, viscous damping and Coulomb damping conditions. Each condition is valid at certain azimuth locations. The behaviour is non-linear by nature because the damper transfer function is time dependent even if each one of the three models is linear; the damper behaviour associated with Coulomb friction is itself non-linear.

Bearing disparities were simulated by appropriate changes in the equivalent spring stiffness. Control system irregularities and frequency adaptor incompatibilities were modelled in a similar way. The effects of mass imbalance were modelled by a Dirac Delta model. Effects of mass changes due to manufacturing tolerances or moisture absorption were simulated by changing the mass of the faulty blades at the affected radial locations.

Measured in-service symptoms including fuselage vibration and blade tip deflections induced by rotor system adjustable faults were considered for validation purposes; for mass, pitch link and tab adjustments, the agreement between the simulation results and the in-service measurements was found to be reasonable in hover and at forward speeds; the symptoms of a lag damper fault were in reasonable agreement with measured symptoms. The crack/delamination model was validated by comparing its results with a well known fracture hinge model that has been reported to agree with experimental results. Details of the fault simulation models and the validation results can be found in [31] to [34].

2.3.5 The Fuselage Model

A fuselage mathematical dynamic model was developed and implemented to evaluate vibration induced by main rotor system faults along the helicopter fuselage, indicating locations and measurement components that significantly reflect fault signatures.

The rigid body response of the helicopter fuselage gives rise to six out-of-trim vibration disturbances, three translations and three rotations. Therefore, the six disturbances were simulated through six non-linear, coupled differential equations describing fuselage rigid body dynamics. The equations were made linear by neglecting the second order terms of disturbances. Combined with a harmonic analysis technique, the equations were used to evaluate the required rigid body response.

The helicopter fuselage is a very complicated structure, which consists of several elements having certain masses, damping and stiffness characteristics. The exact prediction of the fuselage elastic response requires an extensive finite element analysis. Nevertheless, the dominant components of the elastic response can be estimated from the transverse, lateral and torsional motions of a simple free-free structure (i.e. a structure that is not supported at either end). Therefore, differential equations of free vibration were used to calculate the fuselage natural frequencies and mode shapes. The forces and moments acting at the hub were represented by Dirac Delta functions. Modal analysis was applied to derive the differential equations of the elastic degrees of freedom. A harmonic balance technique was used to solve the differential equations.

Comparisons between measured vibration and model results indicated a good agreement between experiment and theory. It was found that the vibration signature is considerably influenced by the fuselage elastic response.

2.3.6 Simulation of Helicopter Unsteady Conditions

MJAD found that structural faults such as blade cracks might not induce observable vibration during steady conditions; nevertheless, a structural mode might be modified by the presence of structural defects. Therefore, it was appropriate to identify operating conditions where the modified mode could be sufficiently excited and produce measurable vibration so that early fault detection might be realised. MJAD extended their HMMs to include a model for unsteady manoeuvres such as rotor start-up and shutdown to establish a diagnostic strategy based on data collected during these unsteady conditions. The model allowed the time histories of each of the helicopter controls to be arbitrarily specified by up to eight polynomials. The model considered the five helicopter controls; namely, collective angle, longitudinal cyclic, lateral cyclic, tail rotor collective angle and rotor angular speed. The model allowed the user to input the geometry of the helicopter including the number of blades 'b' and to specify the required number of degrees of freedom of the rotor blade 'N' and fuselage 'M'.

Using fundamental dynamics and helicopter theories, together with the analysis methodology presented in [30], the following equations were developed, incorporated in the model and solved.

i) N b equations that describe the elastic degrees of freedom of b blades.

ii) Six equations that describe the fuselage "rigid body" degrees of freedom.

iii) M equations that describe fuselage elastic degrees of freedom.

iv) Further equations that describe other components (e.g. tail fin).

The above equations are non-linear differential equations with time varying coefficients. The evaluation of the generalized loads of the above equations required the evaluation of load equations that described the forces and moments induced by the main rotor based on the formula presented in [30]; the forces and moments of other components such as the tail rotor were also considered.

2.3.7 Investigation of Rotor Fault Detection Strategies

Recommendations and strategies for sensor placement and fault detection were established; some of the recommendations and strategies were illustrated using mathematical model results and/or in-service data. It was apparent that most of the diagnostic difficulties arise from the direct use of data in fault detection processes. It was therefore postulated that data pre-processing should be implemented to remove rogue data and attenuate the effects of age and operational factors prior to the application of the diagnostic methods.

2.3.7.1 Sensor Fit

The use of a minimum of six accelerometers was recommended in order to yield three translations and three rotations; the accelerometers should be placed at a minimum of three physical locations on the fuselage (two tri-axial accelerometers would not supply the desired data), and preferably on major frames. The sensors should be distributed around the rotor (e.g. fore and aft) to measure changes in loads and transfer functions induced by rotor system faults. Locations away from, or less sensitive to, noise sources such as gearboxes and engines would produce more reliable normalised features.

The second sensor type would measure blade positions, both in track and lag. Whilst one such device would be imperative and most ideally used to record information at a rotor azimuth of 180 degrees, a second such sensor might be necessary to fully discriminate between faults.

Sufficient promise was indicated as to not warrant placing sensors in the rotating frame: Advanced analysis techniques applied to vibration measured by fixed frame sensors at steady and unsteady flight conditions could be implemented to aid the diagnosis of PCFs; the techniques could include advanced methods such as pre-processing models, anomaly detection techniques and model-based supervised methods. Furthermore, the technology associated with the application of sensors to rotating components was not mature when the work was carried out; there are still practical difficulties with the use of these types of sensor today.

2.3.7.2 Data Integrity

As a first diagnostic step, it was recommended that data validity be checked and corrupted data records be separated; malfunction of a sensor could be identified by analysing sensor signals and comparing the results with a reasonableness threshold and a matrix of cross-validation criteria between system sensors.

2.3.7.3 **Attenuating the Influence of Operational Effects**

The influence of operational effects on sensor measurements could lead to a false interpretation of sensor data and consequent false alarms. Models in the form of simplified equations have been derived and implemented to demonstrate the feasibility of mitigating operational effects through a model-based approach. For example, it was demonstrated that effects of variations in temperature, altitude, weight and speed could be attenuated through model-based equations that related the vibration to these parameters.

2.3.7.4 **Pre-process Mapping**

Mounting a mass on a rotating blade induces vibration proportional to the centrifugal force; adding another mass doubles the vibration level. Hence, classification of faults on the basis of measured vibration symptoms can give rise to an infinite number of faults. If fault severities and a relationship between severities and symptoms are known, the infinite number of vibration symptoms of a fault type can be reduced to a single pattern. The severity for a mass fault is the centrifugal force of added/removed masses, and the relationship is linear. Since multiple faults generate a large number of symptoms, a diagnostic system that classifies faults from symptoms has little practical significance. Intelligence should be implemented to reduce the infinite number of symptoms to a manageable number. It was proposed that this intelligence could be achieved by pre-process mapping, which is a series of transformations applied to HUMS data. For example, theoretically generated data was mapped to vibration modes and it was found that the use of normalised vibration improved the diagnostic capabilities of unsupervised techniques; a vibration mode was defined as the vibration measured along the fuselage divided by the vibration at one of the fuselage locations; a normalised vibration value was defined as the local value of a vibration mode.

Operating on S61 data, it was demonstrated that the use of normalised vibration could isolate the effects of two maintenance actions to rectify high vibration levels (the first action was incorrect, and actually increased vibration levels). The 1R vibration was normalised using different sensors; for most of the normalising sensors (8 out of 10 sensors), the normalised amplitudes identified distinct signatures pre and post adjustments. It was noted that the diagnostic resolution would depend on the choice of the normalising sensor; dividing by the amplitude of a noisy sensor would reduce the quality of results.

2.3.7.5 **Unsupervised Fault Detection Technique**

The capability of an unsupervised data clustering and a pre-processing technique was investigated. Initial attempts at separating fault classes by grouping vibration data using a clustering technique highlighted the problem that a fault from a given class could migrate from one data group to another because of its severity. However, by using normalised vibration, fault types were separated. It was concluded that unsupervised learning is important because it provides unbiased classification of faults: the feature space is divided into regions with each region relating to a fault, an operational condition or both; knowledge about these regions can be used to set an appropriate 'expected fault' model or establish training set requirements for supervised methods.

2.3.7.6 **Supervised Fault Detection Technique**

A supervised neural network can interpolate; nevertheless, extrapolation cannot happen in any predictable way. Therefore, a large dataset (representing a-priori knowledge) is required to train the network; the dataset should contain a range of helicopter measurements that describe the possible range of faults, multiple faults and varying fault severities. Neural networks can provide promising diagnostic tools if the problem of initially limited a-priori knowledge is overcome. This can be achieved through further implementation of mathematical dynamic models to generate fault vibration signatures; training sessions can be initiated using theoretically generated features and then updated by measured features as they become available.

A non-linear supervised method that could cope with limited a-priori knowledge about faults was demonstrated; for each fault, a non-linear equation (model) that relates vibration amplitudes to fault intensity was explicitly expressed; then, a least square error minimisation method was used to determine the weights of the models using measured vibration amplitudes; extrapolation capabilities could thereby be achieved for an adequately representative number of fault models. The reported method could be regarded as an advanced non-linear extension of the well known RTB methods. Trained with vibration measurements of primary faults, it was found that the non-linear supervised technique could reasonably predict primary and compound faults even if the value of fault severity was outside the training range. In other words, the technique could cope with a limited training set.

It was observed that supervised techniques require prior knowledge about the number 'N' of expected faults. Fault diagnosis should be based on an N-expected fault model. Maintenance actions made with the view that the vibration is induced by a number of faults less than the actual 'N' number might never remedy all fault symptoms and could mask serious faults. Unsupervised methods, which divide the feature space into a number of regions, could be used to check whether the correct number of expected faults is chosen. By attaching engineering knowledge to each region, it is possible to determine the number of expected faults. It is also feasible to observe the formation of new regions and update the number of expected faults accordingly.

2.3.7.7 **Diagnostic Strategies of Rotor Faults**

The MJAD investigations indicated the following three conceivable fault discrimination strategies:

i) Identifying the vibration and relative blade displacement disparities induced by faults.

ii) Monitoring a matrix of blade natural frequency perturbations induced by faults.

iii) Estimating the disparities of the system parameters induced by faults, ageing, and/ or environmental disturbances.

The mathematical model results and available HUMS data were used to illustrate the techniques required to implement the first strategy, which included data integrity checking, pre-processing, unsupervised and supervised fault detection techniques.

To date, the HUMS data are acquired at steady operational conditions such as hover and steady forward speed. It was shown that faults such as blade cracks would not induce significant vibration during steady conditions and, hence, measurements at steady conditions would not allow the detection of these faults. Therefore, the mathematical models were used to demonstrate that techniques such as Short Fast Fourier Transform (SFFT) would enable HUMS to be extended to cover these types of faults when applied to vibration data acquired during unsteady conditions such as rotor start-up and shutdown.

It was also demonstrated that available measured vibration could be used to improve the fault predicative capability of the mathematical models through 'a theory-to-test transfer' analysis. Having improved the mathematical model results, the limited a-priori knowledge about faults could be augmented to aid, for example, the establishment of supervised techniques without the need for extensive seeded fault tests.

Limited analysis and formulation were presented to illustrate the second and third strategies.

2.3.8 Further MJAD and GE Aviation Investigations Post-Dating the CAA Work

2.3.8.1 Mathematical Models

Along with minor refinements to the HMMs, GE Aviation concentrated on the development of finite element models for Lynx, Sea King and Chinook helicopters; the models were coupled with HMMs to generate airframe strain and vibration induced by rotor system faults and dynamic events such as hard-landing; the coupled models were used to investigate the relationships between rotor system faults, vibration, airframe loads and structural damage [35] to [37].

2.3.8.2 Diagnostic Strategies for Rotor Faults

A system was developed that implemented the second diagnostic strategy proposed by MJAD, i.e. monitoring a matrix of blade natural frequency perturbations [33]. A Personal Computer (PC) system was developed to determine the health state of helicopter blades by performing modal tests at a blade maintenance bay and, by implementing pattern recognition techniques, the system provided a means of checking the integrity of a blade and determining its best compatible match.

2.3.8.3 Diagnostic and Prognostic Techniques

Further work on pre-diagnostic data normalisation was demonstrated using, for example, a suite of AS332L Flight Data Recorder (FDR) parameters and associated vibration measurements. The full vibration normalisation process involved three steps: filtering out individual helicopter effects on vibration; using a suite of flight parameters and model-based reasoning to simulate the vibration due to operational influences; and filtering out noise. Applying the first two steps to data from a fleet of AS332L aircraft, it was shown that the normalisation significantly reduced the variation in measured vibration between flights, and moved flight records from the extremities of a probability distribution of the vibration towards a central value [38]. The derivation of the model-based equations required for normalisation along with the application of pre-processing, supervised and unsupervised techniques to oil and vibration data were reported in [39].

Data quality algorithms were developed, validated and demonstrated. For example, GE Aviation developed an Automatic Data Correction (ADC) algorithm to identify short period corruptions in flight data and strain measurements of three legacy systems. The identified short period corruptions were classified as spikes, multi spikes, spike-step transitions, steps, hesitant steps, step reversals, dropouts, DC signals, complex corruptions and jump [40]; the ADC algorithm was chosen for Joint Strike Fighter (JSF) airborne implementation. The GE Aviation data quality algorithms identified long period corruptions caused by temporarily or persistently inoperative sensors and calibration problems. For a target sensor, long period corruption was identified by comparing the statistics of the sensor data across a number of sorties, and by cross correlating its data with other sensor data and/or with synthetic data generated by virtual sensors. The statistics of the sensor data were computed over the entire sortie, over its most probable data levels and at a number of predefined flight conditions referred to as 'hypercubes' or 'points in the sky'. A decision making process was implemented to fuse sensor health indicators derived from the above statistics. Generally, the decision making process could use logic, Bayesian belief networks, and/or fuzzy logic to fuse the derived health indicators and increase the detection probability of sensor failures [40].

Limited further work on the application of unsupervised and supervised learning was published. For example, a cluster analysis of 11 simulated adjustable and non-adjustable faults was used to investigate the effects of random noise on vibration data; the vibration features used were 1R and 2R harmonics of lateral and vertical vibration measured at four sensor locations; random noise inducing random variability of up to 30% was superimposed on the vibration harmonics. The classification of the contaminated harmonics produced unsuccessful results, with five clusters containing more than one fault type; nevertheless, by filtering the noisy data the classification results of the filtered features gave a classification success rate of 97%, with only one cluster containing two fault types [41]. Supervised learning was also demonstrated through the use of a neural network to classify the 11 simulated faults from 24 input features. The network was trained using 50% of the available data examples; then it was shown that the network could provide a 100% successful classification of the remaining 50% of the data [41].

GE Aviation also developed, validated and demonstrated a suite of virtual sensors that could compute rotor system loads/torque from flight parameters; the results of the virtual loads could be used to anticipate faults caused by repeated use or by overloading rotating dynamic components [42].

2.4 UoG Independent Review of Previous Research by MJAD and UoM

This section presents a brief summary of the results of an independent review of the previous research by MJAD and UoM that has been produced by the University of Glasgow (UoG). A more detailed commentary on both the MJAD reports and the UoM publications is presented in Appendix B.

2.4.1 UoG Review of Research by MJAD

In the MJAD series of reports, a variety of concepts and ideas that might lead to a practicable rotor HUMS are explored in greater or lesser detail. The approach focuses on the identification of the presence of damage in the rotor system from measurements of the vibrations experienced at a number of fixed points on the helicopter fuselage. Significant effort is devoted to various means of identifying the characteristic signatures of various types of rotor damage in the vibration signal, with qualified success. The most promising technique appears to involve the clustering of measurements of vibration phase and amplitude at the various characteristic frequencies of the system into distinct groups, elevated vibration energy levels in any one of these groups then indicating the presence of a rotor fault. Numerical simulations reveal significant difficulties, though, in defining a set of groups that are robust to variations in fault severity or the presence of multiple, simultaneous faults in the system. Unlike the later UoM study, the value of comprehensive modelling of the rotor dynamics is not seen simply in terms of a numerical test-bed on which various ideas can be formulated and evaluated, however, but also as a direct adjunct to the practical implementation of a rotor HUMS. The major power of numerical modelling is foreseen in its use as a tool to generate a reference signal against which the measured vibration signal from the rotor can be compared and the anomalous signature of the presence of a rotor fault identified. Yet the limitations of any numerical model in matching exactly the physical behaviour of the real world are fully acknowledged and methods are proposed to 'correct', empirically, the deficiencies in the model to render it more able to provide an accurate baseline against which the real signal can be compared. A somewhat disappointing characteristic of the series of reports however, is that, although many indications of potential are scattered throughout, few of the many ideas contained therein are explored to the extent of offering a clear path to a practical rotor HUMS.

It is unfortunate that none of the internal structure of the neural network used in the study is described anywhere within this series of reports. Neural network performance is known to be rather sensitive to the number and configuration of internal nodes. This omission makes it very difficult to assess whether or not the performance of the neural network could have been enhanced by a more complex architecture or by additional data input to the model.

2.4.2 UoG Review of Research by University of Maryland

Key Researchers: Inderjit Chopra (UoM academic, PhD supervisor), Mao Yang (PhD Student), Ranjan Ganguli (PhD Student) and David Haas (Military/Industry Participant)

2.4.2.1 History

The programme of work at UoM extended over a period of about seven years from 1995 - 2002 and appears to have been conducted as a PhD project (Reference [44]), with significant input during the first half of the programme from a second postgraduate researcher, Ranjan Ganguli (Reference [45]). The project appears to have been conducted under the auspices of the Carderock Division of the US Naval Surface Warfare Center (of which David Haas was an employee). The publicly accessible output of the work consisted of six published papers in archival journals, nine conference papers and one directly relevant PhD thesis (Yang). As such, the work appears to post-date the work performed at MJAD, but it is difficult to assess to what extent the UoM project was informed or guided by the earlier work. There are certainly very strong technical parallels between the two programmes, and the inspiration for the project does certainly appear to be contained in the MJAD paper (Reference [32]) that is cited in the majority of the published works from UoM.

Direct Continuation of Work

There appears little evidence that the work in this area is ongoing at UoM. The last conference paper on the work (generally conference publications give a more reliable guide to project duration than journal publications given the usual publication delay imposed by the editorial process) appears to have been presented at the 2002 American Helicopter Society Forum; the last journal paper dates to 2004. The reasons why the work is not ongoing are unclear, but some inference may be obtained from the technical summary below.

Indirect Continuation of Work

An article co-authored by Ranjan Ganguli (currently located at the Department of Aerospace Engineering, Indian Institute of Science, Bangalore), M. Vijaya Kumar and Prasad Sampath (Hindustan Aeronautics Limited) in the April 2006 Journal of the American Helicopter Society (Reference [46]) suggests that elements of the work may be continuing as an unrelated offshoot programme in India. This AHS paper, although tangentially relevant, is interesting as it shows considerable success in identifying the helicopter rigid-body dynamics directly from flight test data using a neural network approach, and hence in implementing directly the approach that was suggested earlier to supplant and simplify the coupled 'mathematical dynamic model' plus 'theory to test transfer function' scheme advocated in the MJAD series of reports. The work does not, as yet, appear to have been applied to the identification of faults within the rotor system, however. A paper presented by Ganguli at the 2006 European Rotorcraft Forum in Maastricht, showed some evidence of a return to work on the rotor fault-detection problem, but with distinctly mixed results.

2.4.2.2 Summary of Technical Aspects of UoM Work

The basic motivation behind the UoM study appears to have been to assess the ability of a HUM system, based on a machine learning approach (neural network), to isolate the signs of a developing rotor fault (or faults) from the vibration signature of the vehicle as measured at various points in the non-rotating frame of reference of the helicopter fuselage. To this end, a number of idealised rotor/blade 'faults' were introduced into a comprehensive helicopter modelling code (UoM's UMARC code). This consists of an individual, finite element, flexible-blade representation of the helicopter main rotor coupled to a finite element, flexible fuselage model. The helicopter aerodynamics are modelled using a lifting line blade model coupled to a free vortex filament representation of the rotor wake. No real fault data was used in the analysis; some of the consequences of operation of the system in the 'real world' were simulated instead by injecting noise into the vibration signals obtained from the simulations. A suitably trained neural network was able to detect the simulated rotor faults, but the reliability of detection degraded if the signal-to-noise ratio was decreased or if the number of faults in the system was increased.

In interpreting the UoM work it is important to realise that the arguments regarding the applicability of neural networks to rotor HUMS are generally made by analogy - the simulations are merely intended as a surrogate for the real world, allowing various ideas to be tested in an easily accessible environment. In the earlier papers in the series, it does not ever appear to have been the intention of the authors that the content of their simulations ever be extrapolated directly to full-scale practice. In fact, they are very careful not to make this connection, and implicit throughout is the acknowledgement that, although their ideas may have merit in guiding the architecture of such a system, any practical implementation of a machine learning based HUMS would rely, for instance, on more realistic, real world fault data being available for the purposes of training the system. To this extent the UoM work is significantly less ambitious and more conservative than the earlier MJAD programme. The later papers in the series (those that have Yang as the primary author) show a distinct change in emphasis towards trying to model correctly the influence of rotor faults on the dynamics of the fuselage; in this respect the UoM study quite closely parallels the earlier elements of the MJAD study, but the UoM programme appears to have been terminated before any significant success was achieved in this respect.

2.4.3 Common Technical Issues

It is an underlying assumption of both the MJAD and UoM approaches that information from the non-rotating frame of reference, in particular vibration data from sensors distributed throughout the fuselage, be used as the primary input to any rotor HUMS. Although this approach is obviously very convenient given the pre-existence of such sensors for other purposes, ease of data transfer to a centralised processing unit and so on, there must be concern that this basic constraint on the form of the system is at the heart of the difficulties experienced in devising a fully robust technique for reliably capturing rotor faults. This situation is likely to be exacerbated in future for several reasons, amongst which are:

i) The increasing use of composites in fuselage structural elements, particularly if this results in increased levels of structural damping. This development may result in signal to noise ratios at sensor locations that are remote from the source of the damage-induced excitation of the system that are significantly reduced compared to the assessments of the earlier research. The same concerns apply to the increased use of elastomeric damping devices in blade root attachments that may significantly attenuate the high-frequency components of any blade damage signature, reducing the crispness of its definition and making it more difficult to distinguish from background vibration levels.

ii) The greater prevalence of active and passive rotor vibration suppression systems in modern designs, with the possibility of significant complication of the transfer function linking the source of the fault on the rotor to the sensor location on the fuselage. This will in all likelihood require a level of complexity in the signal processing algorithms required to distinguish fault signatures from the background vibration that is somewhat more advanced than has been proposed up to now.

It may be worthwhile to survey the state of the art in the emergent technology concerned with the development of on-blade sensors in connection with current research activities in the field of 'smart' rotors.

2.4.4 Outcomes from Review of Existing Research

The previous research was not conclusive as to the extent to which rotor faults can be detected via vibration or blade position measurements, and the technical path that might yield the fastest progress towards a practical rotor HUMS. There would be merit in re-visiting the fundamental mathematical characterisation of the rotor dynamics to more clearly determine what faults can, in principle, be discriminated from within data obtained from the rotor. This may strengthen the foundation for a robust approach to the characterisation of faults from within signals emanating from the rotor system, or prove that certain faults are inherently undetectable from dynamic data measured in a certain way, for example from within the non-rotating frame of reference provided by the helicopter fuselage.

Both the MJAD and UoM studies used machine-learning techniques in their approaches to rotor fault diagnosis. The MJAD study investigated both 'unsupervised' and 'supervised' learning approaches. It is a concern, supported by the results of the studies, that the current state of the art in the area of rotor dynamic modelling is not yet equal to the task of simulating the performance of a fault-ridden rotor to the very high accuracy required for the supervised training of a diagnostic machine, especially since the training, and thus eventual success of such a machine, would be almost wholly dependent on the quality of such simulations. Certainly, for the moment, the less-rigorous demands placed on the accuracy of any simulations used in its training motivates strongly towards the unsupervised approach to the detection of faults within the rotor system, especially since, at least in principle, the behaviour of the machine might be reinforced and improved over time by exposure to real rotor measurements. This approach would allow for the possibility that any early behavioural tendencies leading to spurious alarms, as might result from modelling inadequacies in the early data used to condition the system for example, might be 'forgotten' given the passage of time.

The following inferences may be drawn from the studies:

- The reports suggest that there is variability in the extent to which defects are manifested in the vibration signature, with only the advanced stages of some faults producing distinguishable vibration characteristics.
- The previous research indicates that continuation along the supervised learning approach to the problem is unlikely to yield much benefit for a practical system other than possibly clarifying which faults are detectable, since the demands on any practical system outstrip current modelling technology.

- The studies gave little consideration to the possibility of fault detection by measuring vibration at the hub (in the rotating frame of reference), so there is little evidence on whether faults would be easier or harder to detect in this way.

2.5 Other Relevant Research on Rotor Blade Track and Lag Measurements

Some other relevant UK MoD funded research on main rotor blade track and lag measurements using a passive optical sensor was performed by Stewart Hughes Limited (now GE Aviation) in 1985/6. In one piece of work, trials were conducted on a Puma helicopter using three optical trackers to provide averaged measurements of flap behaviour at three locations along the blade span at a single azimuth of 90 degrees. The results were used for a modal analysis of blade bending. It was concluded that the trial successfully demonstrated the potential to determine the first three flap modes with the analysis showing that, as expected, the first mode was most important. A second piece of work involved the testing of an optical tracker's ability to measure blade incidence on a scaled rotor test rig. If further full scale validation on an aircraft had been performed, the sensor may have provided useful additional fault diagnostic information.

2.6 Example of Current HUMS Rotor Fault Detection Capabilities

Examples of rotor fault detection methods implemented in a current HUMS are those specified by Bell Helicopter and implemented in the Stewart Hughes Limited (now part of GE Aviation) AHUMS™ for Bell 212s and 412s. These illustrate how certain rotor fault detection capabilities can be achieved by the combined analysis of multiple current HUMS rotor vibration and blade track measurements.

The HUMS incorporates techniques for detecting the following five Bell 212/412 non-adjustable main rotor faults:

1 Degradation of hub elastomerics: A hub elastomeric fault is indicated if vertical 1/rev vibration in the flight regimes of 'hover' and 'climb' is higher than the level in normal cruise, and there is a related change in rotor blade lead-lag measurements.

2 Lead-lag damper faults: A lead-lag damper fault is indicated if the lateral 1/rev vibration in flight is above a prescribed limit, and the lead-lag difference between blade pairs changes significantly between different flight regimes (e.g. 'idle' and 'hover', and also 'hover' and 'letdown').

3 Hung droop stop: A droop stop fault is considered to be the most probable cause of high lateral 1/rev vibration at 'idle' if this has increased significantly when no maintenance action has been performed. The fault is indicated if the lateral 1/rev vibration at 'idle' exceeds a prescribed limit, and also the phase of the 1/rev and the track of the suspect blade match defined criteria.

4 Degradation of pylon mounts: If the pylon mounts soften over time due to degradation of the elastomer, the sensitivity of the lateral 1/rev vibration to rotor adjustments increases significantly. Therefore the fault is indicated when there is a significant increase in the trend of the response of the aircraft to balance weight adjustments.

5 Main rotor hub shift: If the cones securing the hub to the mast shift slightly during operation, the main rotor 1/rev vibration can change during flight. Data acquired on different flights are compared to a baseline, and a hub shift is indicated if multiple vibration and blade track criteria are met. These include a change in 1/rev vibration relative to the baseline of greater than a prescribed amount, and the track of two blades in normal cruise changing an equal amount, but in opposite directions.

3 Aircraft Accident and Incident Analysis

The review of accidents and incidents caused by rotor system failures was based on the data contained within the CAA Mandatory Occurrence Reporting (MOR) scheme database. This contains entries for all reportable incidents and accidents to airborne vehicles, and has been running since 1976. Although the majority of the data is sourced from UK civil operations, there are some entries for military accidents and incidents, and some cover accidents and occurrences to foreign aircraft. The data relating to rotorcraft was extracted from this database by the CAA and supplied for the analysis. As described below, the data was filtered to include only entries concerning failure of a rotor component on the larger, Part 29 aircraft and the statistics of this data are presented in Section 3.1.

In addition to the statistical analysis, to provide insight into the requirements for a rotor health monitoring system, a study was undertaken into the more significant helicopter accidents in the database. The data supplied from the MOR database did not, however, contain all the helicopter accidents of interest, either because they concerned non-UK/military aircraft or because they occurred after the data was extracted. These additional accidents were included in the study, and are detailed in Section 3.2, which contains case histories and analysis for five main rotor and three tail rotor occurrences.

3.1 MOR Database Analysis

The contents, classification and statistics of the data from the MOR database are detailed below.

3.1.1 Data Classification

The data extracted covered the time period from the start in 1976 to 4th April, 2006. The relevant data contained in the database was as follows:

Date of occurrence	
Aircraft Type	
Occurrence classification:	Occurrences, Military Occurrences, Serious incidents, UK reportable accidents, All Other Accidents.
Aircraft registration	
AAIB Bulletin number	
Details of injuries/ fatalities caused	
Pre-title:	Short description of the occurrence
Précis:	Fuller details of the occurrence

The data supplied from the full database contained 2,685 entries. While they were all related to the rotor system in some way, not all were caused by a rotor system failure. Many were due to transmission or engine failures, control or instrumentation problems, maintenance errors or external factors. Where there was a rotor failure, this was often due to something striking the blades or being ingested, or a transmission failure (e.g. overspeed, desynchronisation) leading to blade strikes. It was only possible to distinguish the rotor system failure cases by examination of the text entries in each case, which was a time consuming process. Since it was considered that incidents on the larger, passenger carrying aircraft were of most interest, it was decided to restrict the analysis to just the larger aircraft, i.e. those classified as Part 29 (MTOW over 3,175 kg). This resulted in a reduced database comprising 1,908 entries.

3.1.1.1 Classification of Cases

Each of the Part 29 aircraft cases was classified according to the location of the primary cause of the occurrence (e.g. main rotor, tail rotor, drive, engines). For the main or tail rotor cases, there was then a further assessment to determine the cause of the problem (component failure or external factor). Cases included in the subsequent analysis were those caused by failure of the rotor blades (e.g. cracking, delamination, tape lifting) or the hub assembly (e.g. pitch links, bearings, dampers, flapping hinges). Other causes of rotor problems (e.g. blade strikes, ingestion, control/ hydraulic failures, maintenance errors) were filtered out of the data. There were a small number of cases where it was likely that there was a component failure, but the location could not be determined; these were classified as 'nk', i.e. not known.

For each of the Blade or Hub failure cases, an attempt was made to further classify the location of the problem. The following categories were assigned to the cases:

Table 1 Classification of Rotor Faults

Blade	Balance	Blade improperly balanced
	Crack	A crack detected in the blade (BIM, inspection, breakage)
	Delamination	Delamination of a composite blade
	Fenestron	Damage to the outer parts of a fenestron type tail rotor
	Pocket	Damage, lifting or missing pockets in metal blades
	Root	Damage to the blade where it attaches to the hub
	Sleeve	Cracks or other damage in the blade attachment
	Tape	Lost or lifting blade tape, erosion strip, bonding/capping strips
	Tip Cap	Cracks or attachment problems in the blade tip fitting
	NK	Location could not be identified
	Other	Problem with another part of the blade, normally peripheral
Hub	Bearing	Unidentified hub bearing
	Counterweight	Damage to counterweight or its attachments
	Damper	Damper failure, damper attachment problem, damper bearing fault
	Drag Brace	Failure of drag brace or its fittings
	Frequency Adapter	Frequency adapter bearing, bolts, mounting lugs, tie rod, spacers
	Hinge	Blade hinge or hinge bearing
	Hub	Hub or hub splines, spacers, bolt damage
	Mast	Mast or mast bearing
	Pillow	Pillow block seal or bolts
	Pitch Control	Swashplate, rotating/stationary scissors, sleeve, spider, starflex or related bearings
	Pitch Link	Damage to pitch links, attachments or bearings.
	Spindle	Damage to spindle or bearing
	Yoke	Crack or attachment failure
	NK	Location could not be identified
	Other	Other Rotor head problem

3.1.1.2 Benefit Assessment

Since any rotor monitoring system would be aimed at preventing accidents, previous studies have concentrated on analysing only accident or serious incident data. There were relatively few of these in the database; 29 for the main rotor and 24 for the tail rotor. Also, the nature of the accidents meant that there was less information about the cause of failure in these cases, so it would be hard to draw general conclusions from this data. The bulk of the other cases in the database, classified as "occurrences" or "military occurrences", were minor problems, perhaps necessitating a diversion or a precautionary landing, e.g. excessive vibration in flight. It was considered that a statistical analysis of these cases would help to identify where most rotor problems occur and what advantage could potentially be gained from a monitoring system. In order to provide a quantitative assessment of the potential usefulness of rotor monitoring for each case, the following 'benefit analysis' was applied to all the main and tail rotor failure cases.

A rating between 0 and 1 was assigned to each of the parameters judged relevant to the usefulness of such a system for each failure case. The parameters used were as follows:

a)	Vibration likelihood:	The probability that detectable vibration or modification of vibration characteristics would be generated by the fault.
(b)	Vibration before event:	The likelihood of detection before any catastrophic failure during the last flight.
(c)	Vibration before last flight:	The likelihood of detection using data downloaded from previous flights.
(d)	Detection by other means:	The probability that the fault would be detected by existing monitoring, inspection, scheduled maintenance, existing vibration techniques/ HUMS, BIM (Blade Inspection Method), pilot detection of vibration or noise, etc.
(e)	Accident likelihood:	The chances of the fault leading to a failure or vibration which would cause an accident. In addition to catastrophic failure or loss of control, this is high for any fault which causes vibration, since an unscheduled landing or ditching in the sea is inherently hazardous.

These parameters were combined to give a 'benefit rating' for a ground-based and an in-flight warning system. This takes account of the possibility that there is still benefit to be gained even if it is likely that the fault will be detected anyway; a rotor HUM system would provide confirmation and could remove the need for some maintenance/ inspections. The benefit rating is calculated as follows:

$a * ((1-d)/2 + 0.5) * c * e$ for a ground based system.

$a * ((1-d)/2 + 0.5) * b * e$ for an in-flight system.

These ratings are intended to give rough estimations of the potential number of accidents which could be prevented by a rotor monitoring system.

3.1.1.3 Helicopter Flying Rate Data

For UK civil operation only, data on the number of flights and hours flown by each helicopter type by year was available. This was used to give a ratio of faults per flight for each of the Part 29 helicopter types.

3.1.2 Outcomes

A summary of the data analysis results is given below.

Of the 2,685 entries in the database, 1,908 concerned Part 29 aircraft, and of these, 800 were rotor cases, with 1,009 being transmission related and 99 concerning other components (e.g. controls, hydraulics, instrumentation).

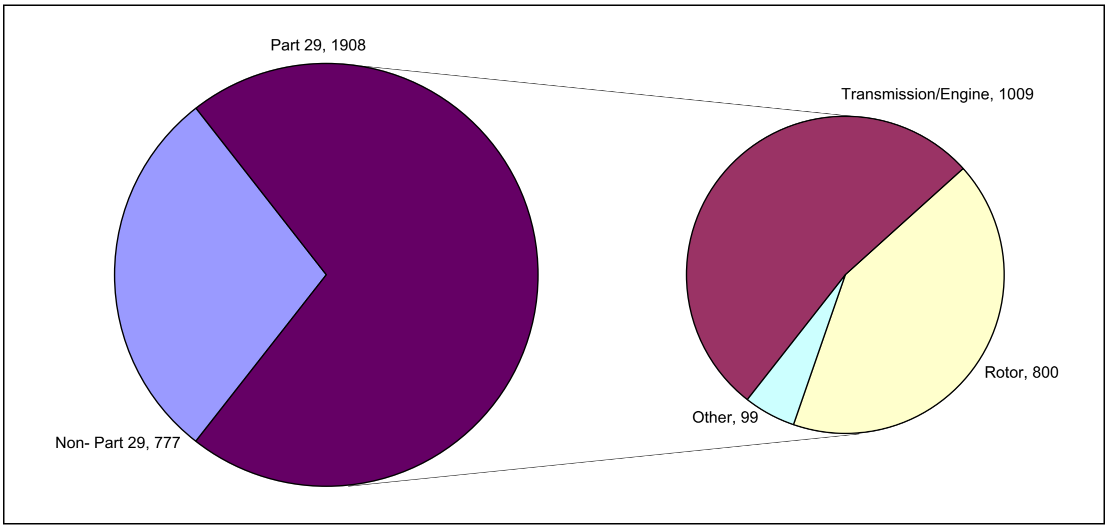

Figure 1 Classification of all Cases

The root causes of the 800 rotor cases are detailed in Figure 2 below.

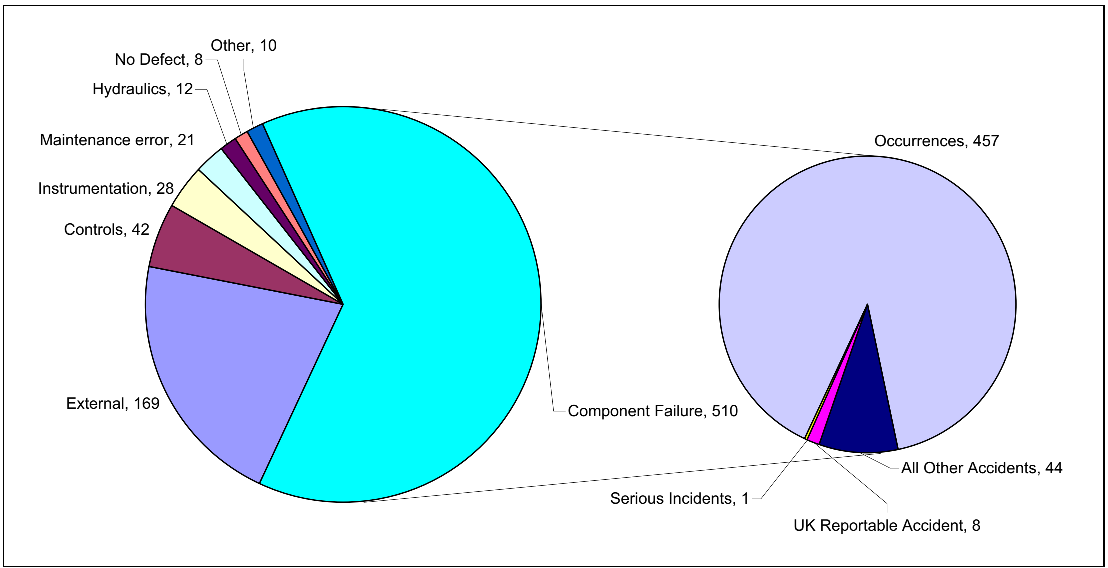

Figure 2 Classification of all Rotor Cases

The 800 rotor cases included 510 attributed to a fault with a rotor component; of the remainder, most were caused by external factors, such as the rotor striking or being struck by an external object. Of the 510 rotor fault cases, 457 were occurrences, with one serious incident and a total of 52 accidents. The 510 fault cases were distributed across the main and tail rotors as shown in Figure 3.

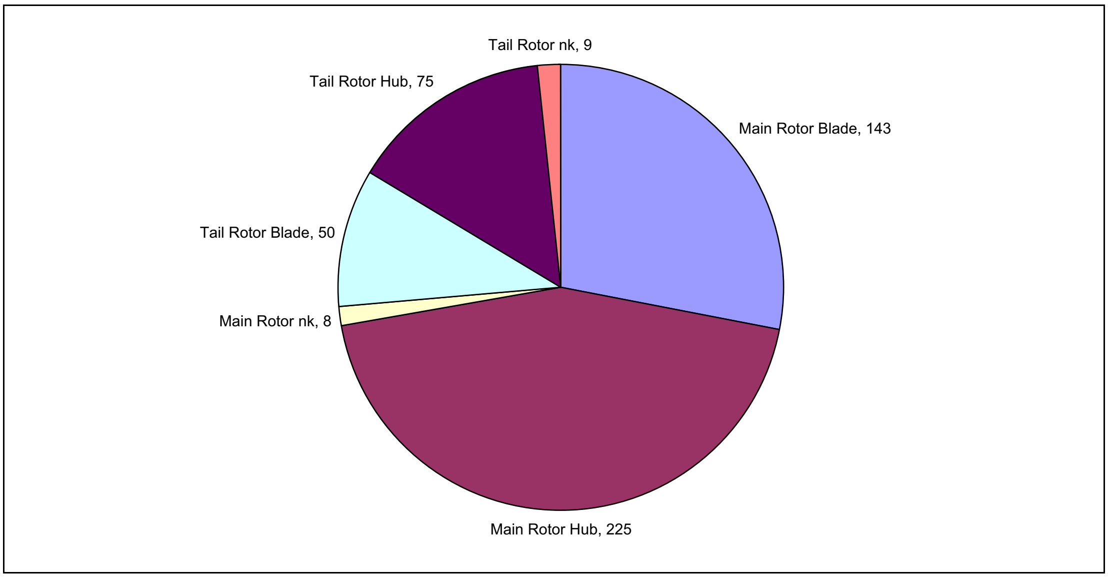

Figure 3 Classification of Rotor Fault Cases

The 53 accidents and serious incidents were distributed as shown in Figure 4.

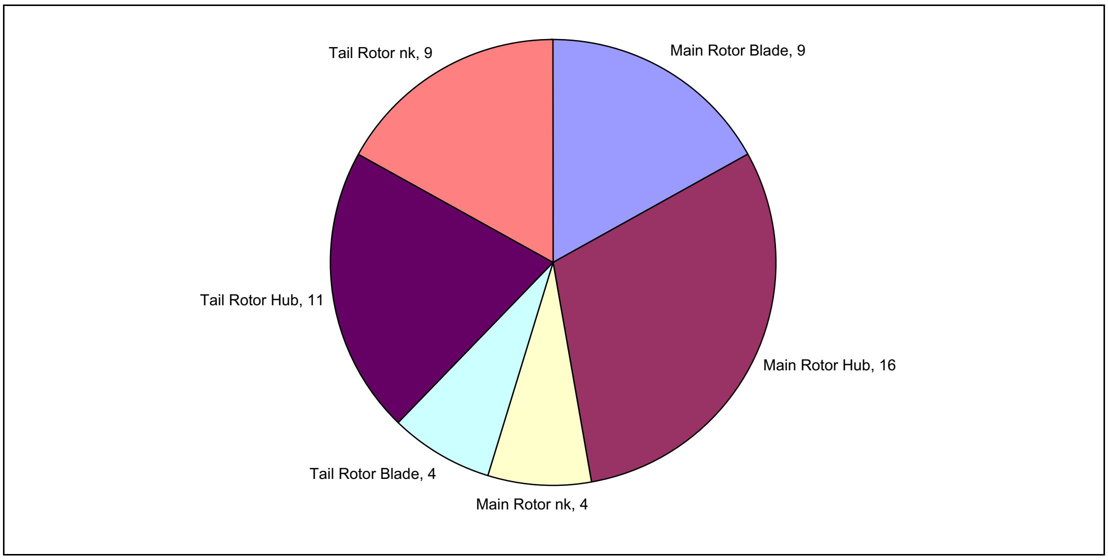

Figure 4 Classification of Rotor Accidents and Serious Incidents

The 510 rotor fault cases were distributed across civil and military accidents and occurrences, in the UK and elsewhere, as shown in Figure 5. There were 411 concerning UK civilian aircraft, nine of those being accidents or serious incidents. It should be noted that the greater ratio of accidents to occurrences in the military and non-UK data reflects the data selected for the database, rather than the relative accident rate.

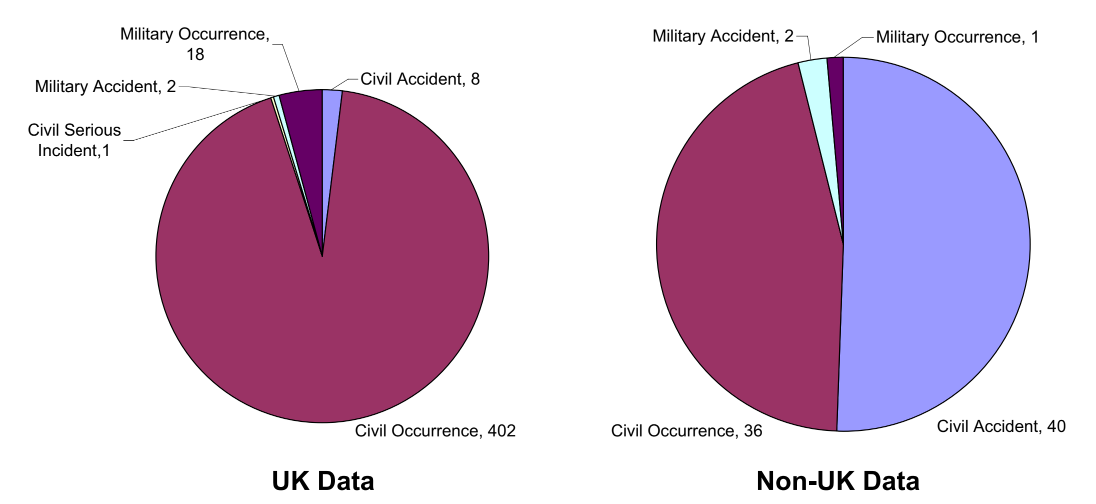

Figure 5 Classification of Rotor Cases by Nationality and Operator Type

3.1.2.1 **Rotor Fault Data**

This section gives more detail on the statistics of the 510 Part 29 rotor fault occurrences.

General Trends

Figure 6 presents the total rotor occurrences and trends for the main and tail rotors by year; it shows a general decline in rotor faults, although the main rotor faults increased slightly from 1995 to 2003. There have been, and continue to be, more main rotor faults than tail rotor faults, probably due to the greater complexity of the main rotor system.

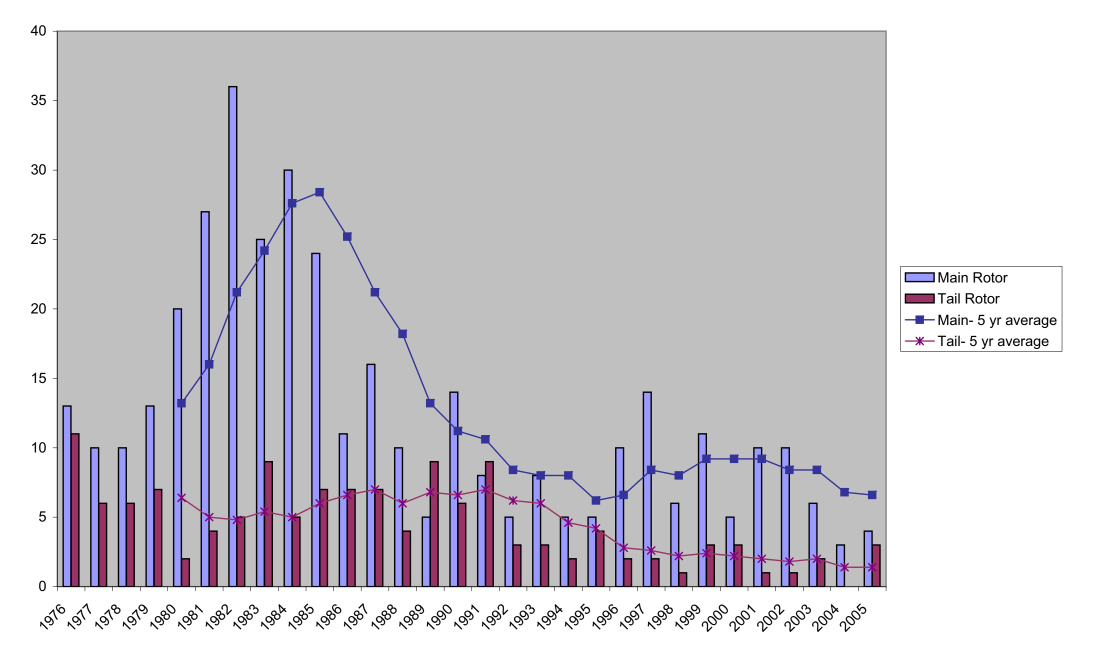

Figure 6 All Rotor Occurrences by Year

Figure 7 shows the combined main and tail rotor trend for the UK civil data only, including a measure of the occurrence rate factored by the number of flights in that year. It can be seen that the UK civil occurrence rate peaked and then declined in the 1980s along with the per flight rate. However, the per flight rate has been roughly constant since then, so the general slow decline in rotor faults since 1990 is probably due to the decreasing flight rate.

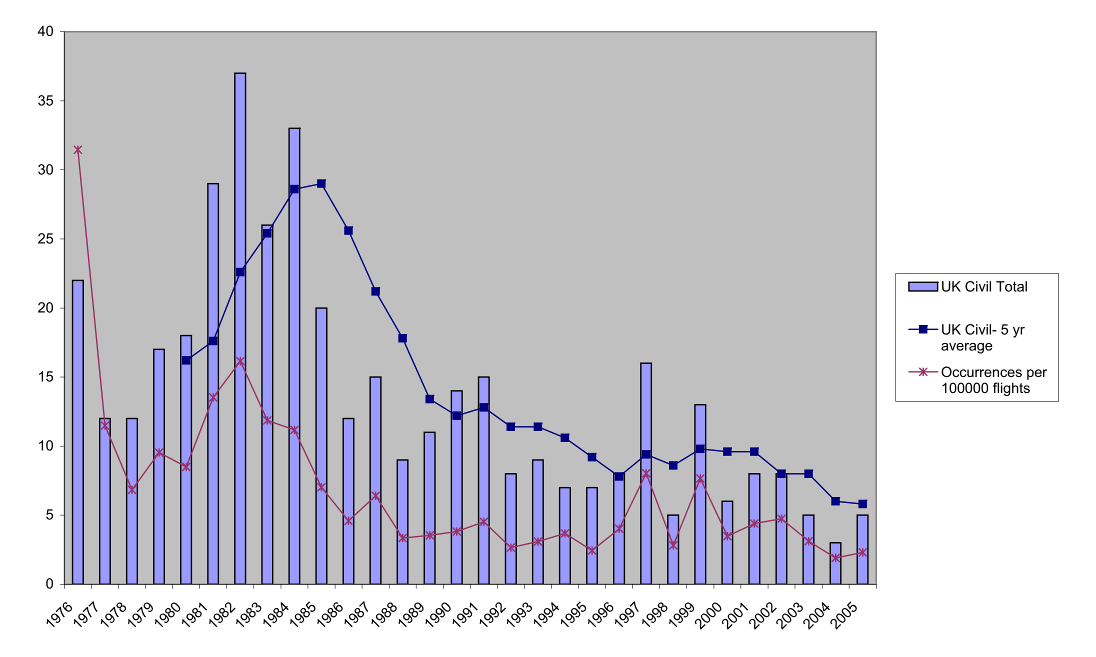

Figure 7 UK Civil Data by Year

In Figure 8, the trends show, for all 510 rotor fault occurrences, those occurrences which were accidents where there were fatalities or serious injuries compared to the rest of the data. These accidents were also concentrated in the 1980s and early 1990s, and there has been only one in the last 13 years.

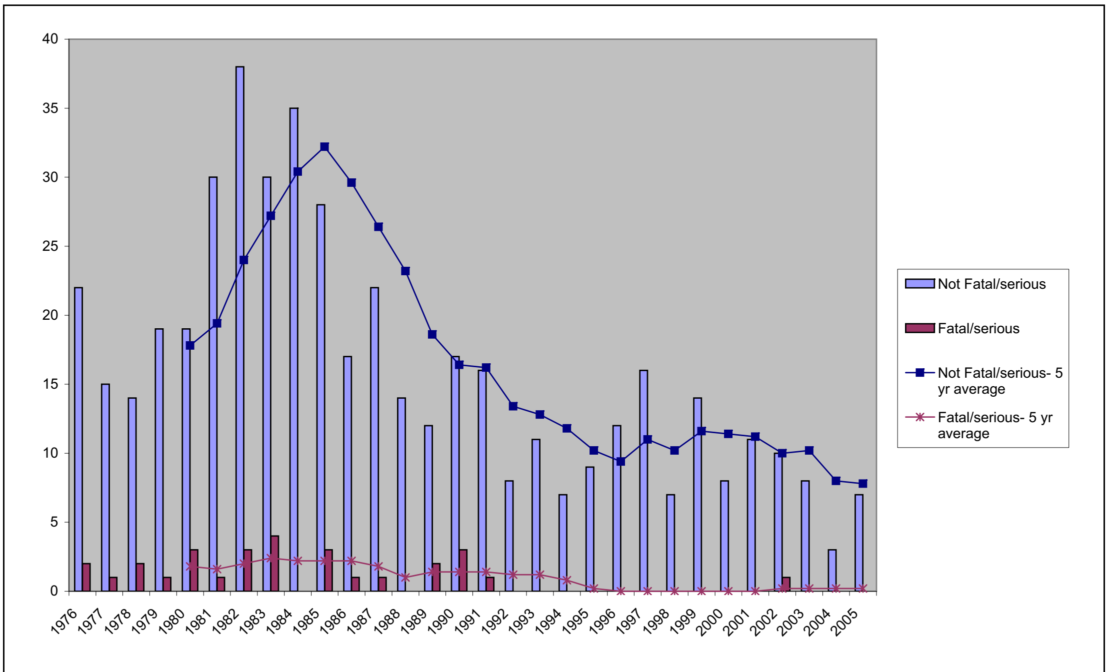

Figure 8 All Rotor Occurrences Showing Proportion of Fatal/Serious Cases

Figure 9a shows the location of the faults for the occurrences which were non-fatal and did not cause serious injuries. Figure 9b shows the same data for those associated with either fatalities or serious injuries. The largest single category in both data sets is the main rotor hub. In addition, hub faults exceed blade faults in both data sets for both main and tail rotors.

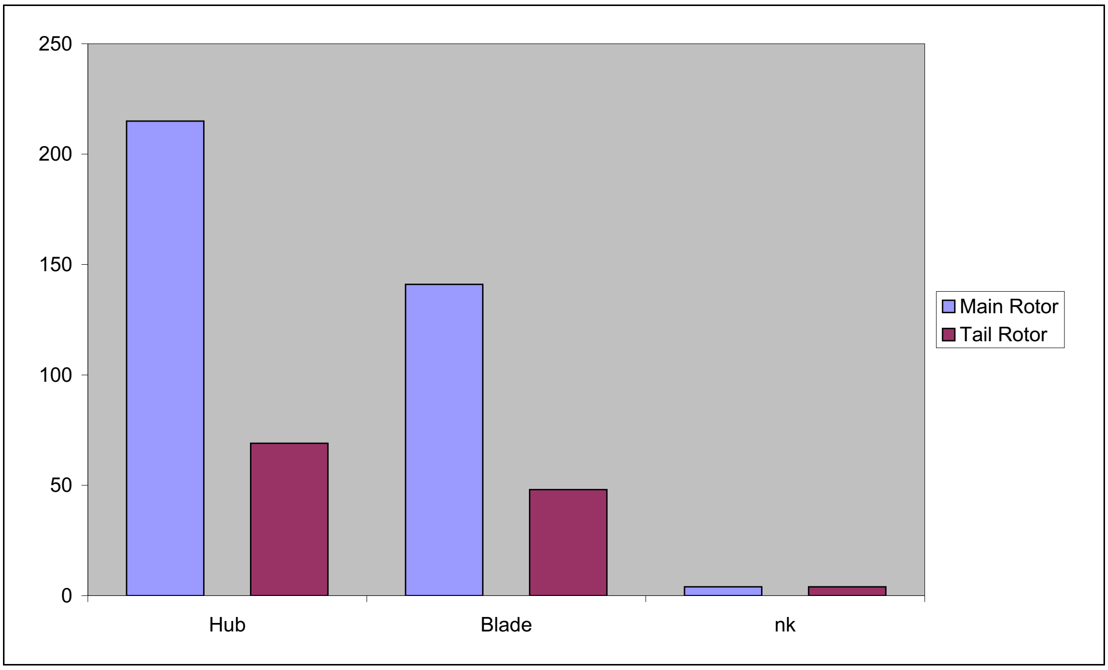

Figure 9a All Non-fatal/Serious Injury Occurrences by Fault Location

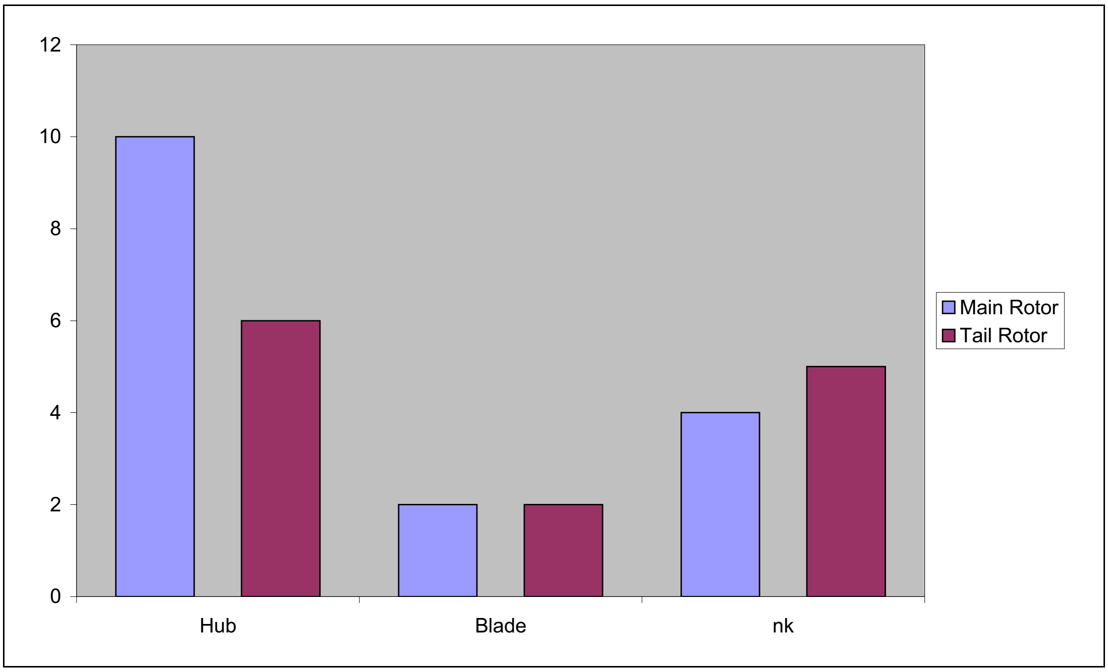

Figure 9b All Fatal/Serious Injury Occurrences by Fault Location

Main Rotor Data

The trend in main rotor hub and blade faults is shown in Figure 10. It can be seen that the peak in the 1980s was mainly caused by hub faults, which decreased rapidly, whilst the blade faults decreased more slowly.

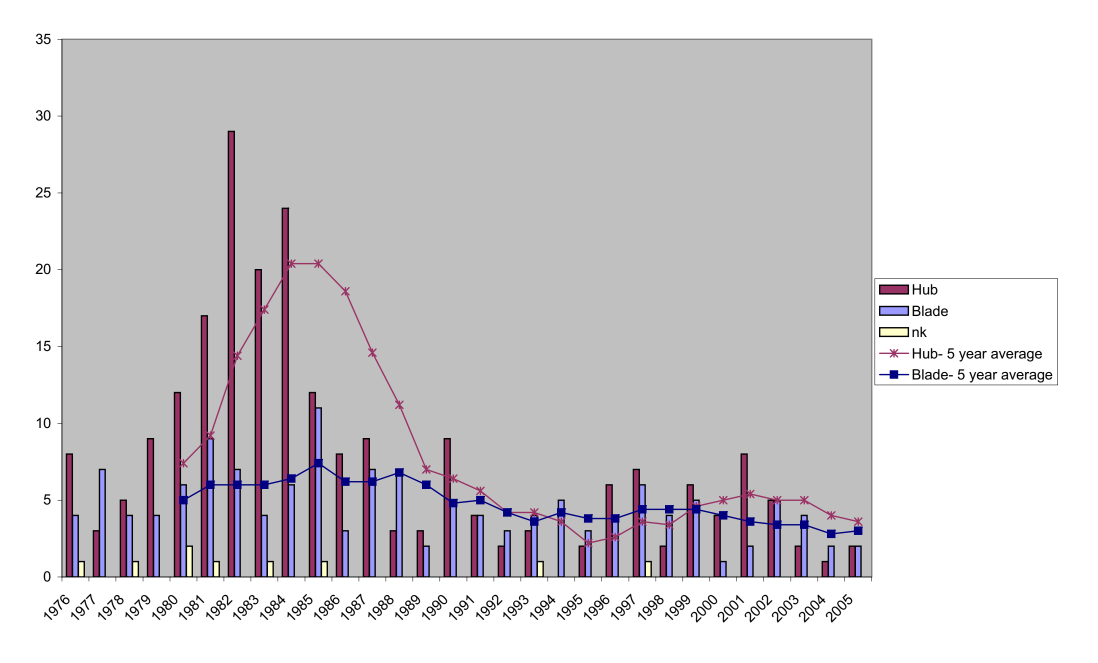

Figure 10 Trends in Main Rotor Faults

Figures 11 and 12 show the breakdown of the main rotor hub and blade faults, respectively. The hub faults were dominated by damper problems, then pitch control, pitch link and spindle issues, whilst most of the blade faults were caused by cracks or delamination, pocket or tip cap problems. It should be noted that the type of blade faults predominantly experienced depends on whether the aircraft have metal or composite blades.

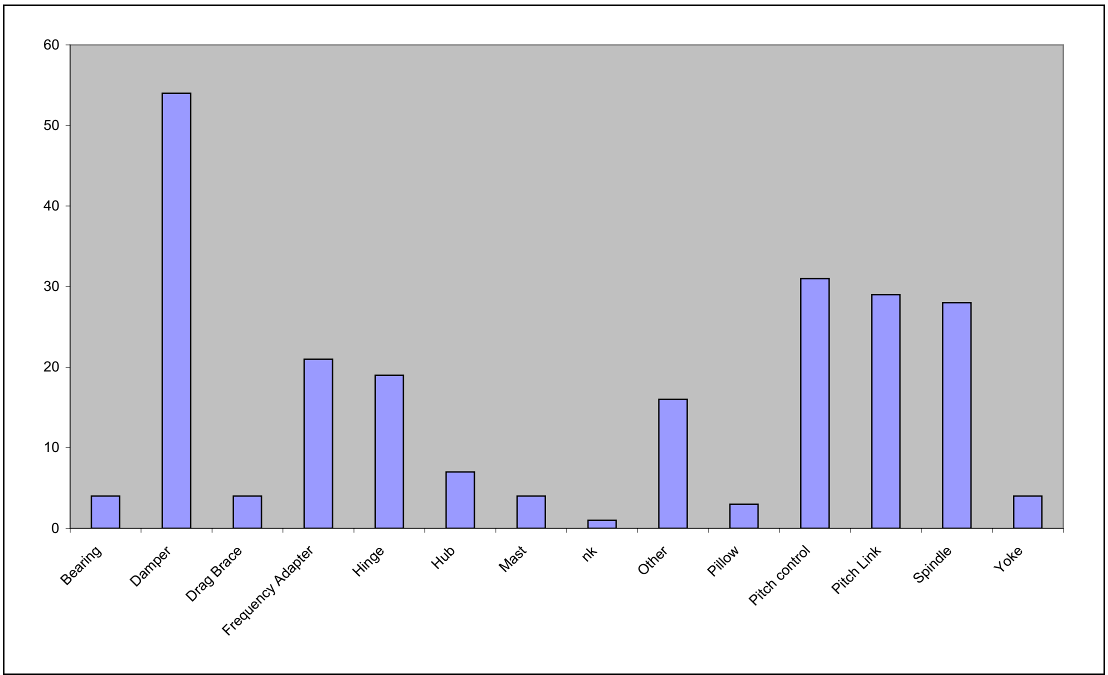

Figure 11 Categorisation of Main Rotor Hub Faults

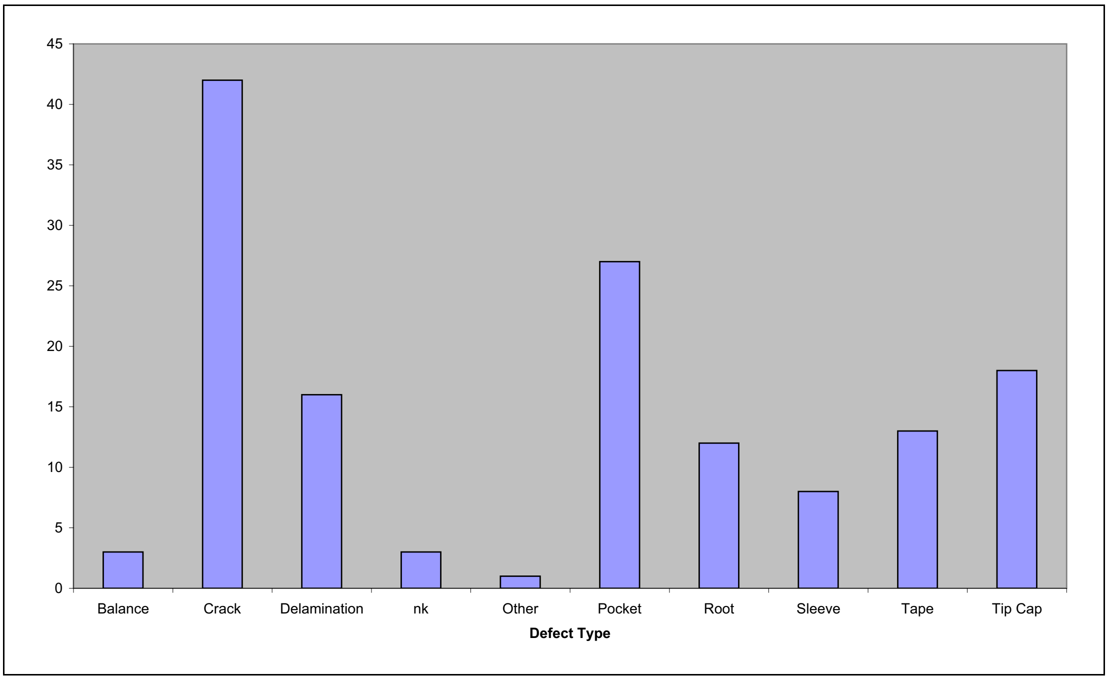

Figure 12 Categorisation of Main Rotor Blade Faults

Tail Rotor Data

The trend in tail rotor hub and blade faults is shown in Figure 13. The data is more scattered than for the main rotor, but both hub and blade faults have declined over the past 10 years.

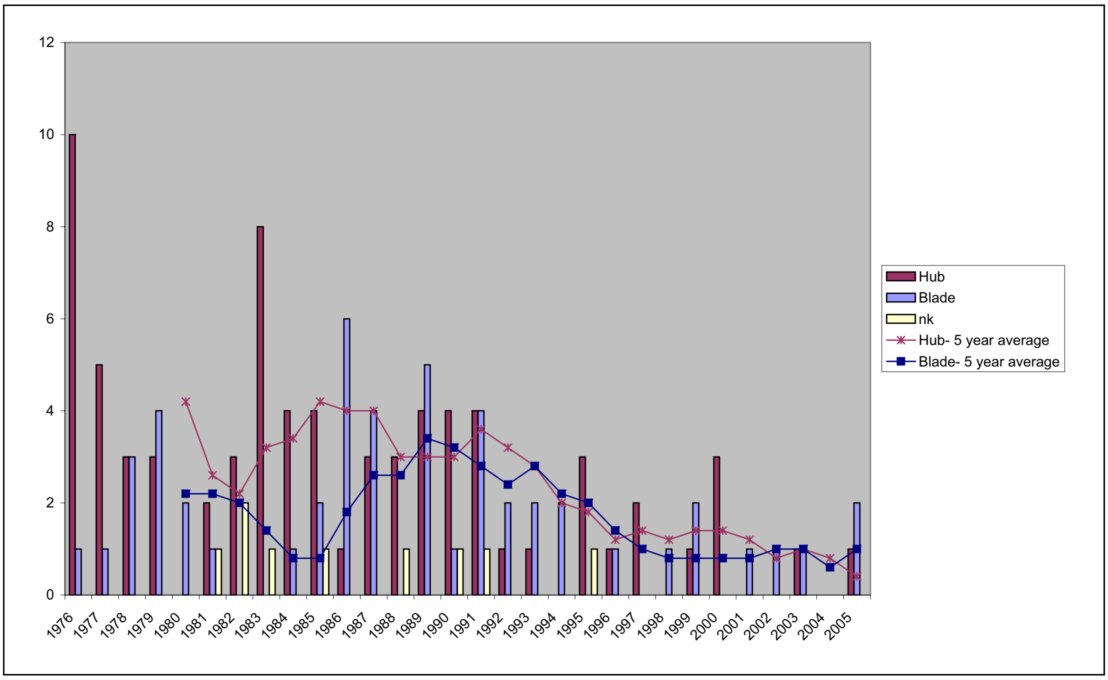

Figure 13 Trends in Tail Rotor Faults

Figures 14 and 15 show the breakdown of the tail rotor hub and blade faults, respectively. Pitch links were the largest single cause of tail rotor faults.

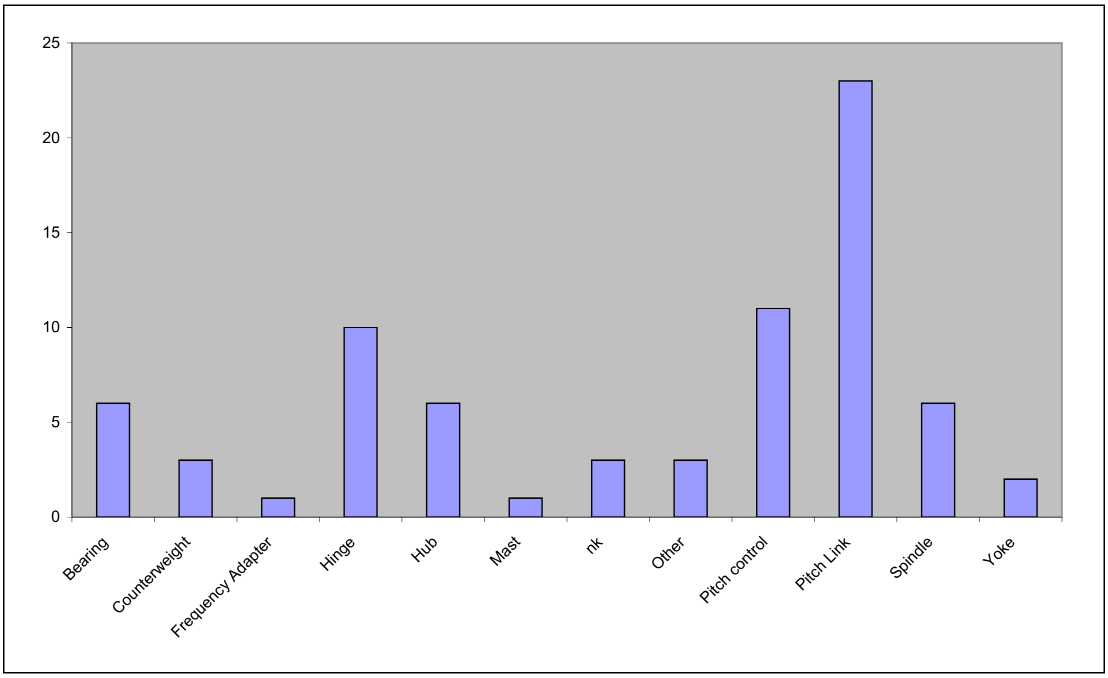

Figure 14 Categorisation of Tail Rotor Hub Faults

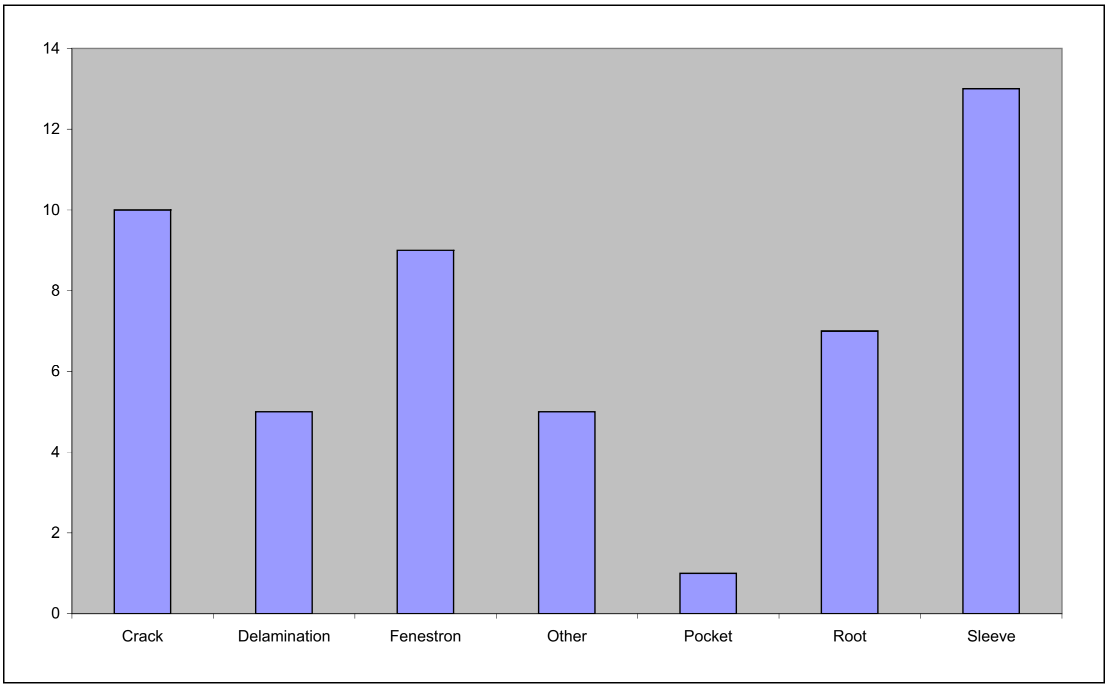

Figure 15 Categorisation of Tail Rotor Blade Faults

Benefit Rating

Figures 16 and 17 show the faults for the main and tail rotors scaled by their benefit rating (for a ground-based system). The derivation of benefit rating is detailed in Section 3.1.1.2. This is a speculative analysis, but it uses all the occurrence and accident data to give a rough indication of the number of accidents which could potentially be prevented by a rotor health monitoring system. The benefit is greater for the main rotor than for the tail rotor, but by less of a margin in recent years. This is due to the main rotor (blade plus hub) occurrence rate falling from a peak of over 4 accidents per year to less than 1.

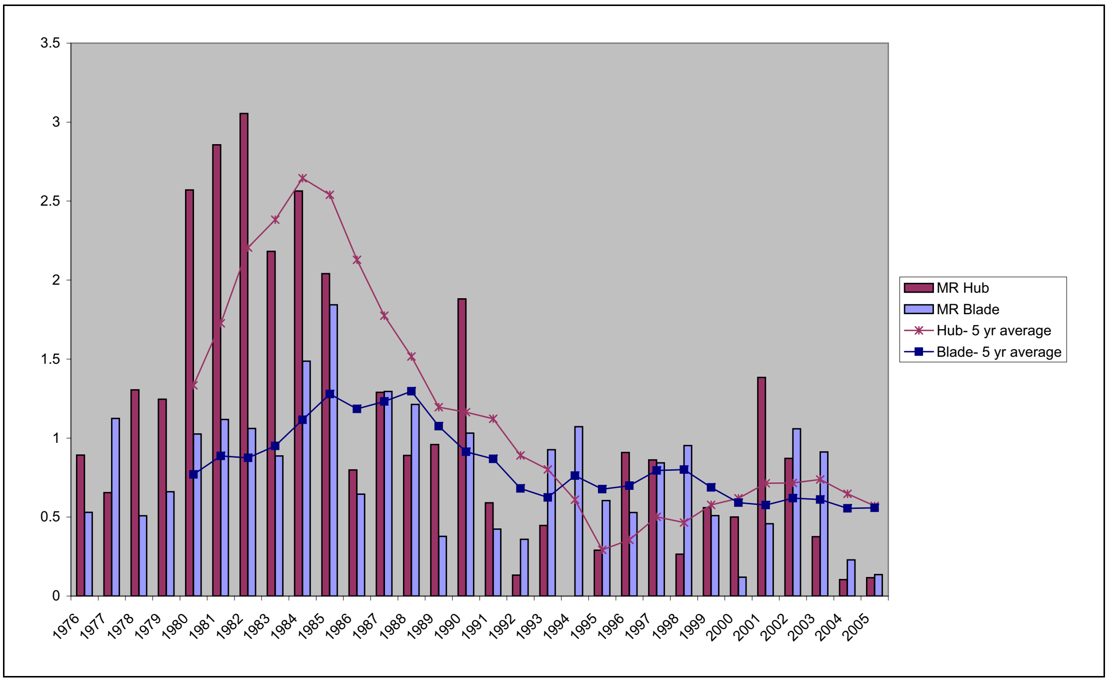

Figure 16 Main Rotor Benefit Rating by Year

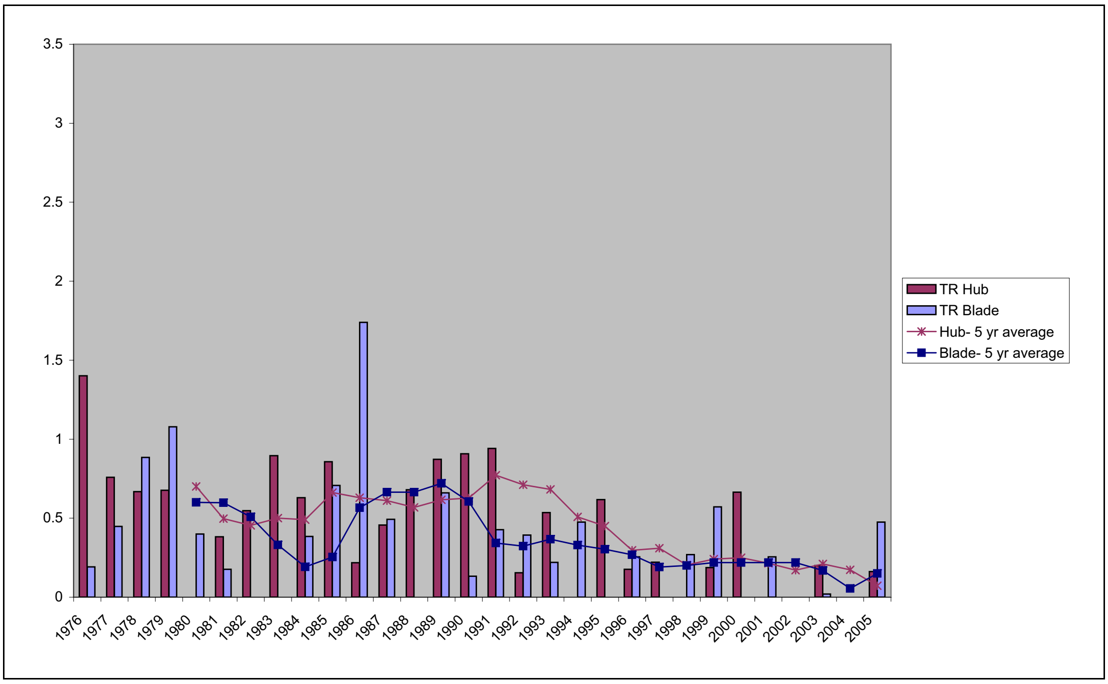

Figure 17 Tail Rotor Benefit Rating by Year

Aircraft Type Data

Further detailed analysis of the rotor fault data, including the breakdown of faults by aircraft type, is presented in Appendix C. This data should be used for assessing where it would be best to apply any further research effort and where a monitoring system would be most useful, and not for analysis of aircraft reliability. In many cases a high number of faults on an aircraft type is the result of a high flying rate of that aircraft, rather than any particular problem with the reliability of the rotor components.

3.1.3 **Discussion**

Most of the data in the database (approximately 84%) is from UK based aircraft and consists of occurrences as well as accidents. Within the UK data set there are very few accidents in relation to the number of occurrences, as would be expected. The remaining data (approximately 16%) relates to non-UK based aircraft and consists mostly of accidents with a few occurrences included as well. Since there have been many more accidents worldwide than there have been in the UK, the overall accident statistics are dominated by non-UK aircraft (approximately 79%). Though most of the data is sourced from civil operations, only 81% is from UK-based civil operations and approximately 5% is from military aircraft. This should be noted when viewing the graphs which are factored by aircraft usage, Figure 7 and Appendix C, Figure 1.

There were more main rotor faults than tail rotor, presumably due to the greater complexity of the main rotor. This indicates that there may be more benefit in targeting a monitoring system at main rotor faults. However, it should also be noted that the numbers of fatal/serious injury accidents caused by main and tail rotor faults were quite similar.

There was a peak in the main rotor faults in the 1980s, caused mainly by hub faults, and a decline thereafter. The peak did not coincide with the peak helicopter usage, which was in 1991, so there must be some other explanation. Analysis of the individual aircraft type data in Appendix C shows that the Chinook, Super Puma and S76 aircraft faults were the major contributors, and these aircraft were being introduced into service around this time. The Chinook aircraft was withdrawn from UK civil operations in the 1980s, which accounts for the drop it its annual fault rate. For the other types it is probable that this data represents the high initial fault rates that commonly occur when introducing any new helicopter type into service. However, a smaller rise in fault rates occurred in the late 1990s, which does not coincide with the introduction of any new aircraft type.

For the main rotor, there were more hub faults than blade faults during the high fault rate period, but the two are roughly equal in recent years. This does not indicate a specific area on which to concentrate a monitoring system. For the tail rotor, there was a decline in fault rate through the 1990s.

It is assumed that the general decline on the main and tail rotors is due to several factors:

- Improvements in materials and design of rotor components.
- Increased and refined inspection techniques as a result of rotor failures.
- Improved rotor monitoring and balancing technology (e.g. Blade Inspection Method (BIM), NDT, HUMS vibration trending, other periodic vibration checks).

There was a decline in numbers of helicopter flights, numbers of rotor-related accidents, and accident rate over time; in fact only 1 accident caused by a rotor failure was recorded in the last 13 years. This may suggest that there is limited benefit to be gained from additional rotor monitoring. However, there are several other factors to consider:

- The introduction of new helicopter types may increase the fault rate, though the design improvements on modern helicopter types may mean that this is not a significant factor.
- A monitoring system could reduce the need for intrusive and time- consuming inspections of rotor components, though a degree of maturity would be required before this could be achieved.
- A mature monitoring system could reduce the number of precautionary/ unscheduled landings by distinguishing between causes of vibration.

The benefit rating data is more speculative, since it is based on estimates of likely vibration and probability of detection, as well as an accident probability for the occurrences. For the tail rotor, there is a decreasing benefit with time, but the main rotor data is more scattered, with a slightly higher number of faults between 1996 and 2003.

3.1.4 Summary

The analysis shows that the frequency of rotor faults and accident rates are decreasing, the faults/flight rate for 2005 being only 1/7 of the 1982 value. The estimated benefit rating for a rotor monitoring system is also lower in recent years than it was in the 1980s. This shows that it would have been more advantageous to have a system in the past than it is now, but does not necessarily imply that it would not be worthwhile now or in the future.

The data presented here and in Appendix C does give some guidance on where rotor health monitoring should be aimed for maximum benefit (main rotor vs tail rotor; hub vs blade), and the fault types (e.g. pitch controls/links, blade cracks) that rotor monitoring needs to be able to detect.

3.2 Helicopter Accident and Incident Case Studies

This section contains a description of particular helicopter rotor failure cases where more detail is available from accident reports or bulletins than can be found in the MOR database. In addition, there are some notable rotor failures which are not included in the database, and details of these are also given below.

3.2.1 Main rotor

3.2.1.1 S76A Blade Failure (Reference [1])

Fault History

Date:	July, 2002
Fatalities/injuries:	11
Occurrence Type:	Accident
Report/ Bulletin:	AAIB Formal Report 1/2005
Included in MOR database:	Yes

The aircraft was on the fifth sector of a flight in the southern North Sea when a main rotor blade failed, causing the main rotor to separate from the fuselage, which crashed into the sea. The failure was caused by a fatigue crack originating from microstructural damage in a repaired portion of the blade spar after the blade had been struck by lightning. This crack had been propagating for approximately 100 hours, and had caused a sympathetic crack in the blade skin at least 7.3 hours before the failure. This crack was not detected by inspection of the blades since it was hidden, for much of the time, beneath an opaque protective patch and was in a location difficult to inspect during pre-flight checks.

During the last flight, the pilots noticed an increase in vibration levels about seven minutes before the accident, which they attributed to an out of track blade, confirmed by observation of the rotor disc.

The HUM system fitted to the aircraft was fully operational, and there was a good historical record of the track, lag and vibration as well as on the day of the accident. There had been a single vibration exceedance after the fourth sector, recording excessive 1R vertical vibration on the ground, but no action was prompted since spurious exceedances of this type were common, and the vibration levels in cruise were low. Examination of the track data showed that the cracked blade was flying low, and getting lower throughout the day, but that the track split was not unusual compared to previous times in the history of the aircraft. The vibration harmonics were not examined or reported on. It was concluded that there were no HUMS indications which should have indicated a developing fault.

Discussion

There were clear indications of the impending fault in the rotor vibration during the last flight, but these were too late to have enabled any fault detection system to have averted the accident. Even if the blade crack had been diagnosed at that time, there was no time to divert or even ditch in the sea.

There were developing HUMS indications at an earlier stage but these were not unusual compared to previous vibration or track split levels where, for instance, adjustments were necessary. The accident investigation did not include examination of all the HUMS vibration data, in particular the higher harmonics. It is possible that a sophisticated system looking at all harmonics of vibration across different flight regimes could have spotted characteristic fault pattern, but there is no supporting evidence from this case.

This case demonstrates the similarity of fault induced vibration as measured by current HUMS to that caused by common rotor tracking errors.

3.2.1.2 **S76A Spindle (Reference [47])**

Fault History

Date:	March, 1981
Fatalities/injuries:	4
Occurrence Type:	Accident
Report/ Bulletin:	AAIB Aircraft Incident Report 9/83
Included in MOR database:	Yes

On a training flight, in straight and level cruise at 3000 ft, a main rotor blade detached and the helicopter broke up and crashed. The blade spindle had suffered a fatigue failure in the threaded portion next to the blade retaining nut. The failure was induced by the increased spindle loading caused by wear in a dry lubricated bearing. In addition, the strength of the spindle was compromised by manufacturing errors. This failure was similar to a previous in-flight spindle failure on the same aircraft type.

Metallurgical analysis gave a probable crack propagation time of 50 flight hours. It is not certain how the bearing wear which contributed to the failure progressed, but it is thought likely that most of the wear occurred rapidly in a short time period before the accident.

Discussion

There was no vibration or CVFDR data recording on this aircraft, so it is hard to tell what indications would have been given. It can be speculated that the increasing bearing clearance, which led to the fault, could have produced vibration trends in a similar way to the tail rotor flapping hinge cases (see Section 3.2.2). If this were the case, early detection of the fault using an anomaly detection system may have been possible.

3.2.1.3 **AS332L1 Hub Failure**

Fault History

Date:	October, 2006
Fatalities/injuries:	0
Occurrence Type:	Incident
Report/ Bulletin:	None
Included in MOR database:	No

As the pilot was raising the collective to take off, there was a bang followed by vibration, so the takeoff was aborted and the passengers safely evacuated. Examination of the rotor head revealed a fractured main rotor spindle in the yoke area. Initial indications were of a fatigue type crack propagation mechanism.

The aircraft was equipped with HUMS, which had given no abnormal indications prior to the incident. Initial inspection of the data showed that the track and lag data for the previous flight were normal, the vibration on ground just prior to the failure was at low levels and no trends could be seen in the relevant transmission data.

Discussion

It is difficult to imagine any vibration anomalies being produced by the development of the fatigue crack, since any displacements caused would have been small compared to normal variations in blade position. The available HUMS data showed no abnormalities, but it may be worthwhile to examine trends in the harmonics across all regimes.

3.2.1.4 **AS332L1 Frequency Adaptor (Reference [48])**

Fault History

Date:	July, 1987
Fatalities/injuries:	0
Occurrence Type:	Incident
Report/ Bulletin:	None
Included in MOR database:	Yes

The aircraft was on a return flight to Aberdeen, carrying 18 passengers, when three thumps were felt. The aircraft was turned to divert to Sumburgh, and severe vibration immediately developed, making control of the aircraft extremely difficult. Ditching was considered, but the pilot managed to find control positions which reduced the vibration whilst maintaining progress towards the land. A landing was achieved, at which point examination of the rotor head revealed a detached frequency adaptor. The failure had occurred through wear of the liner on the spherical bearing, which had last been inspected 39 hours previously.

Discussion

This could have been a more serious accident under other circumstances. The excessive vibration could have led to ditching (inherently hazardous), or failure of other components.

The bearings were regularly checked, and these checks have been made more stringent since the failure. Modifications have also been made to improve the bearing installation design.

Again, there is no vibration evidence, but the gradual deterioration of the component could indicate the possibility of trends in vibration characteristics.

3.2.1.5 Bell 214ST Drag Brace Failure (Reference [49])

Fault History

Date:	July, 1987
Fatalities/injuries:	0
Occurrence Type:	Incident
Report/ Bulletin:	None
Included in MOR database:	Yes

The aircraft was heavily loaded, and in cruise at 3000 ft when there was a sudden onset of vibration making speech, control and reading the instruments very difficult. The aircraft was turned to land as quickly as possible, resulting in a high rotor rpm, rate of descent and landing speed. As the helicopter came to a halt the vibration became more severe, but the flight crew managed to stop the rotors safely and all passengers and crew were evacuated.

The cause of the vibration was a broken main rotor blade drag brace, which had failed by fatigue in the threaded portion of the barrel section, and induced a similar failure in the stiff nut at the same location. The breakage allowed the blade to pivot in a lead/ lag sense about its attachment pin, resulting in severe imbalance of the main rotor

Discussion

The nature and location of the fault means that, although the crack was probably propagating for some time, it is unlikely that it would have produced detectable vibration or blade position anomalies. The displacement and imbalance caused by the opening of the fatigue crack would be small compared to normal variations in the system caused by, for example, differential blade erosion or dirt accretion.

Like the last fault (Section 3.2.1.4), this failure could have led to a more serious accident had it occurred over the sea, for example, or in bad weather.

3.2.2 Tail Rotor

3.2.2.1 AS332L1 Hub Failure (Reference [50])

Fault History

Date:	June, 2005
Fatalities/injuries:	0
Occurrence Type:	Accident
Report/ Bulletin:	None
Included in MOR database:	No

The MHS Super Puma was ditched in the sea following increasing vibration. Inspection of the tail rotor revealed a broken arm on the pitch change spider. Investigation of the HUMS data concluded that there was rapid fault development and increase in vibration to abnormal levels on the last day's flying, with significant increases on the flight prior to the last. The vibration changes occurred in the 1T, 4T, 6T and 9T harmonics, and disruption could be seen in the vibration signal averages. Some less significant vibration increases in the trend for the last 50 hours of the 2T Radial data may have been indicative of earlier fault development.

Discussion

The fault development was fairly rapid, but symptoms were present prior to the last flight, so there is the potential for prevention of the accident. The fault manifested itself in the vibration signature as modulation of the blade pass harmonic (i.e. increases in 4T and 6T), rather than an increase in 1T vibration.

3.2.2.2 **AS332L1 Flapping Hinge Failure (Reference [51])**

Fault History

Date:	September, 1995
Fatalities/injuries:	0
Occurrence Type:	Incident
Report/Bulletin:	AAIB Aircraft Incident Report 2/98
Included in MOR database:	Yes

Whilst in cruise at 120 kt en-route to a platform in the North Sea, there was a sudden onset of airframe vibration. The cause of the vibration was not obvious to the crew, who initially diagnosed a main rotor fault. The aircraft was diverted and landed safely, evacuating all passengers. Upon inspection it was apparent that the vibration was caused by the fatigue cracking of a tail rotor flapping hinge retainer, initiated by fretting and corrosion of the retainer bore, attributed to a defective bearing.

Review of the HUMS data showed a clear trend in the 1T vibration, beginning 50 hours earlier, which had resulted in an exceedance. The maintenance undertaken as a result of this exceedance failed to identify the cracked component, therefore the tail rotor gearbox was replaced and the rotor rebalanced. This reduced the vibration levels and the aircraft was returned to service. The failure occurred approximately 5 hours later.

There have been several other documented cases of tail rotor flapping hinge bearing defects which have shown rising trends in vibration levels.

Discussion

This shows a developing tail rotor fault which can be detected via changes in the vibration signature, well before actual failure occurs. A sufficiently advanced rotor HUMS could have alerted the maintainers to the abnormal nature of the rapidly rising trend in 1T, avoiding an incorrect maintenance action.

3.2.2.3 **Bell 412 Blade Failure (Reference [52])**

Fault History

Date:	July, 2002
Fatalities/injuries:	4
Occurrence Type:	Accident
Report/ Bulletin:	Canadian Forces FSIR 1010-CH146420
Included in MOR database:	No

When returning from a search and rescue mission, the tail rotor departed the aircraft while in normal cruise flight. The aircraft crashed into hilly, tree-covered terrain about 400 metres down track from the tail rotor departure. The two pilots were killed instantly when the aircraft struck the ground with high vertical speed. Another crew member was very seriously injured while the fourth crew member sustained serious injuries. Investigation revealed that the failure was due to a fatigue crack initiating from a small damage site on the skin of the rotor blade about 18.5 inches from the tip of one blade. That section of one blade then flew off; the resulting imbalance of this dynamic component caused the T/R input shaft to fail instantly.

The aircraft was equipped with a HUM system. Investigation of the HUMS data showed that tail rotor vibration started to increase during a 2 h 12 min flight, with the most significant rise occurring during the next 57 min flight, and a final increase occurring during a last 29 min flight which ended with the accident. There were rising trends in the 1T axial and 1T radial measurements in normal cruise, and also in the tail rotor gearbox output shaft 1/rev vibration. Each of these indicators showed a 50% to 100% increase from quasi steady-state values over a 2 to 3 hour period. By the end of the flight prior to the accident the HUMS had recorded one exceedance for 1T axial vibration and two exceedances for 1T radial vibration. The aircraft did not return to base during this time, therefore the HUMS data had not been downloaded to a ground station.

Discussion

Although the potential warning time was very limited, it may have been possible to generate a HUMS tail rotor alert prior to the final flight. Fusing the combined 1T axial and 1T radial data, and also possibly the tail rotor gearbox 1/rev data, may have given a much clearer picture of the abnormal nature of the combined developing trends in the different measurements from different sensors.

3.2.3 Overview of Accident Case Studies

Review of the accident data in the MORS database and the potential benefit to be gained from a rotor HUMS (Figures 16 and 17) may give the impression that rotor monitoring would be of little benefit due to the low accident rate in recent years. However, the addition of the more recent case studies not included in the database shows that rotor failures still have the potential to cause accidents.

The tail rotor cases above, in particular, give hope that it would be possible to prevent at least some of these failures by applying an effective anomaly detection system, and that the data currently being gathered for RTB purposes may be suitable for input into such a system.

For the main rotor cases, there is no evidence from any of the fault cases of fault induced vibration at an early enough stage in the fault development to avoid the failure; this is partly because there was no RTB monitoring in many cases. However, from comparison with the tail rotor cases, it seems possible that such vibration may be detectable in some fault scenarios. There is no evidence that it would be possible to detect a developing crack in a main rotor blade or hub component, such as the AS332L1 hub failure (Section 3.2.1.3). However, where the fault is caused by component wear or degradation, such as in the frequency adaptor case (Section 3.2.1.4), it may be possible to detect it before the point of failure.

In the case of the tail rotor flapping hinge retainer failure (Section 3.2.2.2), there were clear indications of the developing fault which were noted by the maintainers but misdiagnosed. In this case, a rotor monitoring system which could distinguish the abnormal rotor fault symptoms could have avoided the incorrect maintenance and prevented the in-flight failure.

3.3 Outcomes from Accident Analysis

Although the overall statistics point to a declining accident rate, the more recent rotor related accident cases suggest that the potential safety benefits to be gained from further research into rotor heath monitoring would justify the effort required.

Although the vibration data available was very limited, review of the circumstances of the accidents suggests agreement with the earlier research that not all faults would produce detectable vibration characteristics at a sufficiently early stage in their development.

However, it was apparent in the accident case histories that the early stages of development of some faults would be detectable via existing vibration measurements, and that the trending and health monitoring potential of the data currently gathered for RTB purposes is not being fully exploited.

In some cases, faults have been mis-diagnosed, and the vibration due to a developing fault has been balanced out by adjusting the rotor. A system that could correctly diagnose the fault in these cases could be highly beneficial in accident prevention.

4 Fault Detection Strategy and Near-Term Future Development

4.1 Fault Detection Strategy

The previous research indicated potentially promising methods for rotor fault detection utilising pre-processing, unsupervised and supervised analysis techniques and using vibration measured during both steady and unsteady operational conditions. The research did not, however, conclusively determine whether comprehensive fault detection would be possible, or identify a clear way forward. In addition, notwithstanding the recent fault cases, analysis of the accident data does not support a speculative large scale research programme that would be very costly and time consuming. However, the outcomes of both the review of previous research and the accident review indicate possible areas for near term investigation that may be cost effective, and could be accomplished relatively easily.

Based on the previous work, five areas of investigation can be defined for future rotor HUMS research:

i) The extension, and trial using HUMS data, of the methods proposed by MJAD and partially demonstrated through the work commissioned by CAA in 1991, which included pre-processing, unsupervised and supervised data analysis as well as mathematically generating fault symptoms.

ii) The investigation of the potential use of vibration data acquired during unsteady flight conditions to aid the detection of rotor system catastrophic faults.

iii) The investigation of the emergent rotating-frame sensing technologies including data transfer from the rotor system to the non-rotating fuselage equipment.

iv) The investigation of the application of anomaly detection techniques to existing operational HUMS rotor vibration data. This could give an experimental validation of the feasibility of rotor system HUMS.

v) Combined with the analysis of HUMS operational data, the use of current rotorcraft modelling techniques to further investigate the rotor induced vibration, endorse the theory that certain types of rotor fault are manifested in the vibration, determine how the faults might be detected and investigate how various detection techniques could be fused together to obtain more robust diagnostic results.

In order to realise an effective rotor HUMS capable of detecting rotor faults, all of the above five investigation activities should be carried out where feasible, however the following research items should be given initial priority.

4.2 Near-Term Future Development

4.2.1 Application of Unsupervised Techniques to Existing HUMS Data

The outcomes of both the accident review and the review of previous research indicate that, at least for certain rotor faults, there are detectable changes in some of the vibration measurements currently being recorded on most HUMS equipped helicopters (monitoring thresholds are currently not set on many of these measurements). The possibilities for fault detection using unsupervised learning could most easily be investigated by the application of the anomaly detection techniques developed on the CAA transmission HUMS research (References [53] and [54]) to existing rotor vibration data. This could give an experimental validation of the feasibility of rotor system HUMS.

It is therefore proposed that a practical, low risk, investigation is performed to determine what more information can be extracted from existing HUMS data through the application of the advanced anomaly detection technique that has been successfully trialled on transmission data from a fleet of Bristow AS332L aircraft.

Ideally, the aircraft sensors, measurements to be taken, and range of operating conditions would be determined from a theoretical understanding of the vibration produced by rotor faults. However, it should be possible to fulfil some of the criteria for a suitable dataset by making use of existing vibration and blade position data already gathered for RTB purposes. While the sensor locations and measurement techniques may not necessarily be what would be indicated by a theoretical rotor modelling exercise, there should be enough archived data available to provide worthwhile insight.

The available vibration data could be used to develop and evaluate anomaly models using multiple rotor measurements (e.g. different vibration harmonics, multiple flight states) to describe the behaviour of the rotor data in no-fault conditions. It will be necessary to select a subset of the available measurements to incorporate into the models, and determine what pre-processing to apply. This would best be achieved by considering, from a theoretical basis, which measurements are most likely to indicate that a fault is present. Unlike the transmission case, it is likely that rotor fault detection will require analysis of data from several different operating conditions, as well as careful consideration of any maintenance inputs.

Once the rotor data has been modelled, fault case data could be used to determine which faults may be detected by comparison with the model. This fault data could be identified as anomalous cases within the existing data, sourced from known fault histories (if the data was available), or estimated from theoretical modelling of faults.

The MORS database analysis suggests that, of the aircraft types currently flying, the AS332L aircraft would be most relevant to this study. The high flying rate for this aircraft means that there should be a large amount of data available, and many of the relevant fault cases occurred on this aircraft type. One possible source for such data is the Bristows IHUMS database, which stores rotor measurement results from the North Sea operations of its fleet of around 15 aircraft. A preliminary investigation of the relevant files from one server showed that the following data is available:

i) Rotor measurement records accumulate for AS332L aircraft at an approximate rate of 900 records per month, with each record representing a flight.

ii) Each record contains a set of data for one or more regimes. There are sizeable data sets for each of the regimes of MPOG, Hover, 115kts, 125kts, 135kts, Climb, Descent.

iii) Each data set contains vibration amplitude and phase data from two accelerometers for harmonics 1R, 2R, 3R, 4R and 8R (main rotor) and 1R, 2R, 3R, 5R and 10R (tail rotor). In many cases there is also blade track, lag and velocity data for each main rotor blade.

The anomaly detection work should accomplish the following tasks:

i) Data warehousing - decoding the data and storing this in a database in a suitable format for data mining.

ii) Obtaining rotor maintenance data, including records of adjustments carried out, component replacements and any faults identified. Support will be required from the aircraft operator for this work.

iii) Selection of data to model from the main rotor and the tail rotor. There should be an input to this process from the previous theoretical work to determine where and how faults are most likely to appear within the available data. This selection should include:

a) vibration harmonics and/or blade data;

b) spread of regimes from the same flight; and

c) pre-processing to apply (e.g. trending, smoothing, normalisation).

iv) Development of models using the same anomaly modelling techniques as the transmission modelling work, but incorporating the particular rotor features such as adjustments and measurements in different regimes.

v) Use of the 'models of normality' to:

a) produce a fleet view of AS332L rotor behaviour;

b) detect anomalies in the historical data and correlate these with maintenance information; and

c) quantify expected vibration levels for comparison with any model outputs of expected fault levels.

4.2.1.1 **Expected Benefits**

A large percentage of the rotor vibration and blade position data that is currently recorded on HUMS is not used for trending or health monitoring purposes. Anomaly modelling would mean that all this data could be monitored and any significant deviations from normal behaviour could be alerted to the operator.

The modelling would give a better understanding of the characteristics of rotor vibration and blade position measurements and their variability with aircraft registration, time, adjustment settings, flight regime, etc.

There are indications that certain tail rotor failure modes can produce detectable changes in different vibration harmonics that currently do not have thresholds, and this may be true for the main rotor as well. If so, an anomaly detection exercise would identify where this is the case and provide a mechanism for detection. Any detection methods thus produced could be relatively easily implemented on existing aircraft using existing hardware.

Application of the techniques should give a clearer understanding of whether any fault indications are distinguishable from the normal variations in vibration characteristics.

4.2.2 **Further Theoretical Analysis**

Combined with the analysis of HUMS operational data, it would be beneficial to perform further theoretical analysis, building on the results obtained from the MJAD comprehensive mathematical models.

The general theory underpinning the rotorcraft modelling would be re-visited to review, from a theoretical basis, which rotor faults are manifested in the vibration and how these might be detected. This would include a general consideration of rotor faults in terms of the loading that they induce, and how components of rotor loading transfer into the fixed frame. There would then be a systematic determination of the extent to which fixed frame measurements are capable of differentiating any particular fault-induced loading of interest from any of the other various types of loading that might exist within the rotating frame.

Such an analysis would re-validate the existence of, and classify, specific fault 'signatures' in the fixed-frame vibration signal generated by the rotor. The results of the theoretical review would answer conclusively the question as to whether individual rotor faults possess clearly differentiable characteristics that could be used for their identification by a rotor HUM system. Of particular interest is the feasibility of differentiating adjustable faults that should be corrected by balance weight, pitch link and tab adjustments, from non-adjustable faults with airworthiness implications.

The rotor modelling would be used to systematically review the characteristics of rotor induced vibration, endorse the theory that rotor faults are manifested in the vibration, determine how the faults might be detected and investigate how various detection techniques could be fused together to obtain more robust diagnostic results.

4.2.2.1 **Expected Benefits**

A systematic analysis as described above would yield a clearer assessment of the fundamental feasibility of rotor HUMS than is available at present, and also the potential airworthiness benefits in terms of which faults could be detected, and which are likely to be undetectable.

The proposed theoretical work would provide further direction as to how any theoretically feasible system might be implemented in practice, and should give a clearer indication as to where future research would be best directed.

5 Conclusions

5.1 This report presents the results of a helicopter rotor health monitoring status review commissioned by the CAA. The work consisted of a review of previous research in the field and an analysis of helicopter accident and incident statistics. Based on the outcomes from these exercises, a strategy for future rotor HUMS research has been defined and outlines given for a proposed near-term programme of work.

5.2 A review of previous research, conducted by MJA Dynamics and the University of Maryland, determined that the realisation of an effective HUMS capable of detecting rotor faults requires further work in some or all of the following areas:

i) The extension, and trial using HUMS data, of the methods proposed by MJAD and partially demonstrated through the work commissioned by CAA in 1991, which included pre-processing, unsupervised and supervised data analysis as well as mathematically generating fault symptoms.

ii) The investigation of the potential use of vibration data acquired during unsteady flight conditions to aid the detection of rotor system catastrophic faults.

iii) The investigation of the emergent rotating-frame sensing technologies including data transfer from the rotor system to the non-rotating fuselage equipment.

iv) The investigation of the application of anomaly detection techniques to existing operational HUMS rotor vibration data. This could give an experimental validation of the feasibility of a rotor system HUMS.

v) Combined with the analysis of HUMS operational data, the use of current rotorcraft modelling techniques to further investigate the rotor induced vibration, endorse the theory that rotor faults are manifested in the vibration, determine how the faults might be detected and investigate how various detection techniques could be fused together to obtain more robust diagnostic results.

5.3 A review of rotor-related helicopter accident and incident data covering the last 28 years revealed that:

i) Although the accident rate is declining, a few recent high profile cases have demonstrated that there is still significant safety benefit in main and tail rotor fault detection.

ii) Some rotor fault mechanisms may be detectable through analysis of existing vibration measurements.

5.4 Both parts of the study indicated that, for some faults, it is unlikely that detection would be possible via vibration or blade position measurements at an early enough stage to provide meaningful safety benefits. Because of the likelihood that not all faults would be detectable, and because the fault rate appears to be generally declining, it is considered that a costly and speculative large-scale research programme into new technologies would be inappropriate.

However, two near term tasks have been identified to progress the rotor HUMS research in the most cost effective manner, and a potential programme of work has been outlined in this report:

i) It is proposed that an experimental investigation into the feasibility of a rotor system HUMS is conducted by the analysis of historical in-service rotor data, using the anomaly detection methods successfully developed and applied to HUMS transmission data (an unsupervised learning technique). This will provide a better understanding of the characteristics of in-service rotor vibration and blade position measurements, and determine the potential airworthiness benefits from improved rotor fault detection through the application of more advanced analysis techniques to existing HUMS data.

ii) Combined with the analysis of HUMS data, it would also be beneficial to build on the results obtained from the MJAD comprehensive mathematical models by performing a systematic theoretical review of how fault-related components of rotor loads are manifested in the fixed frame. This would establish which airworthiness-related rotor faults are potentially detectable, further investigate fault detection techniques, and define fusion methods that combine the results of various detection techniques to obtain more robust diagnostic results.

6 References

[1] Report on the accident to Sikorsky S-76A+, G-BJVX near the Leman 49/26 Foxtrot platform in the North Sea on 16 July 2002", AAIB, 2005 (AAIB Aircraft Accident Report No. 1/2005).

[2] Effects of Helicopter Main Rotor Faults on Fuselage Vibration Modes' (Azzam and Andrew, January 1991. (MJAD/R/60/91)).

[3] The Application of a Math-Dynamic Model to Characterise Blade Cracking/ Delamination Faults during Rotor Start-Up and Shut-Down' (Azzam and Andrew, March 1991. (MJAD/R/63/91)).

[4] An Appraisal of Theory-to-Test Transfer Analysis for Helicopter Vibration Prediction' (Azzam and Andrew, May 1991. (MJAD/R/69/91)).

[5] The Implementation of a Pattern Recognition Technique to Classify Main Rotor System Faults' (Azzam, Callan and Andrew, September 1991. (MJAD/R/75/91)).

[6] A Nonlinear Supervised Diagnostic Technique for the Recognition of Main Rotor System Primary and Compound Faults' (Azzam and Andrew, November 1991. (MJAD/R/86/91)).

[7] The Use of Infield Data and Mathematical Dynamic Models to Evaluate Diagnostic Strategies for Main Rotor Systems Faults' (Azzam and Andrew, December 1991. (MJAD/R/90/91)).

[8] Formulation of a Rotor-System Damage Detection Methodology," Ganguli, Chopra and Haas, 36th AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics and Materials Conference and Adaptive Structures Forum, New Orleans, LA, April 1995 (AIAA 95-1230-CP).

[9] Formulation of a Rotor-System Damage Detection Methodology", Ganguli, Chopra and Haas, Journal of the American Helicopter Society Vol.41 No.4, October 1996.

[10] Helicopter Rotor-System Damage Detection," Ganguli, Chopra and Haas, American Helicopter Society 2nd International Aeromechanics Specialists' Conference, Bridgeport, CT, October 1995.

[11] Detection of Helicopter Rotor System Simulated Faults using Neural Networks," Ganguli, Chopra and Haas, 36th AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics and Materials Conference and Adaptive Structures Forum, Salt Lake City, UT, April 1996 (AIAA-96-1646-CP).

[12] Detection of Helicopter Rotor System Simulated Faults using Neural Networks," Ganguli, Chopra and Haas, Journal of the American Helicopter Society, Vol.42 No.2, April 1997.

[13] A Physics-Based Model for Rotor System Health Monitoring," Ganguli, Chopra and Haas, 22nd European Rotorcraft Forum, Brighton, September 1996.

[14] Helicopter Rotor System Fault Detection using Physics-Based Model and Neural Networks," Ganguli, Chopra and Haas, AIAA Journal, Vol.36 No.6, June 1998.

[15] Helicopter Rotor System Health Monitoring using Numerical Simulation and Neural Networks," Ganguli, Chopra and Haas, 53rd American Helicopter Society Annual National Forum, Virginia Beach VA, April/May 1997.

[16] Simulation of Helicopter Rotor-System Structural Damage, Blade Mistracking, Friction and Freeplay," Ganguli, Chopra and Haas, AIAA Journal of Aircraft, Vol.35 No.4, July-August 1998.

[17] Rotor System Health Monitoring Using Coupled Rotor-Fuselage Vibration Analysis", Yang, Chopra and Haas, 56th American Helicopter Society Annual National Forum, Virginia Beach, VA, May 2000.

[18] Rotor-Fuselage Vibration Analysis for a Dissimilar Rotor with Single and Multiple Faults", Yang, Chopra and Haas, 57th American Helicopter Society Annual National Forum, Washington DC, May 2001.

[19] Vibration Prediction for Rotor System with Faults using Coupled Rotor Fuselage model," Yang, Chopra and Haas, Journal of Aircraft, Vol.41, No.2, March-April 2004.

[20] Sensitivity of Rotor-Fault-Induced Vibrations to Operational and Design Parameters," Yang, Chopra and Haas, 58th American Helicopter Society Annual National Forum, Montreal, June 2002.

[21] Sensitivity of Rotor-Fault-Induced Vibrations to Operational and Design Parameters," Yang, Chopra and Haas, Journal of the American Helicopter Society, Vol.49 No.3, July 2004.

[22] Configuration and Output of a Computer Mathematical Model Simulating Potentially Catastrophic Main Rotor Faults", Azzam, November 1989 (MJAD/R/28/89).

[23] The Identification of Potentially Catastrophic Helicopter Rotor Faults (Theoretical Study)", Hesham Azzam, January 1990 (MJAD/R/30/90)

[24] An assessment of the feasibility of simulating and early detecting potentially catastrophic main rotor faults", Hesham Azzam, 1988 (MJAD/R/17/88).

[25] "Investigation of Rotor Wing Interference," Hesham Azzam, MSc thesis, Southampton University, 1980.

[26] Hesham Azzam and Paul Taylor. "A Prescribed Wake Model for Helicopter Behaviour," 11th European Rotorcraft Forum, September 1985.

[27] Hesham Azzam and Paul Taylor. "A Robust General Helicopter Trim Program," 11th European Rotorcraft Forum, September 1985.

[28] Hesham Azzam and Paul Taylor. "Investigation of Helicopter Twin-Rotor Characteristics," 12th European Rotorcraft Forum, September 1986.

[29] Hesham Azzam and Paul Taylor. "A Improved Assumed Mode Method for Estimating the Motion of a Nonuniform Rotor Blade," 12th European Rotorcraft Forum, September 1986.

[30] Hesham Azzam. "Investigation of Single and Twin Rotor Behaviour," PhD thesis, University of Southampton, 1986.

[31] Hesham Azzam and Richard Talbot. "An Appraisal of Defective-Machine-Math-Dynamic Models for Machine Health Audits," Condition Monitoring and Diagnostic Engineering Management, COMADEM Proceedings, 1991.

[32] Hesham Azzam and Mike Andrew. "The Use of Math-Dynamic Models to Aid the Development of Integrated Health and Usage Monitoring Systems," Journal of Aerospace Engineering, Proc Instn Mech Engrs, Part G, Vol 206, pp 71-76, 1992.

[33] Hesham Azzam, and Mike Andrew. "Modal Testing and Machine Learning Techniques for Structural Health Monitoring," International conference, Structural dynamic modelling, test, analysis and correlation, Edited by NAFEMS, 1993.

[34] Hesham Azzam. "The Use of Mathematical Models and Artificial Intelligence Techniques to Improve HUMS Prediction Capabilities," The Royal Aeronautical Society, Proceedings of Innovation in Rotorcraft Technology, pp16.1-16.14, 1997.

[35] Malcolm Wallace and Hesham Azzam. "Development and Use of a Finite Element Model to Investigate the Relationship Between Helicopter Hub Loads and Airframe Stresses," Proceedings of 1999 MSC Users' Conference, June 1999.

[36] Alan Draper, "Fatigue Usage Monitoring In UK Military Helicopters" Workshop on Helicopter Health and Usage, Monitoring Systems, Melbourne, Australia, February 1999.

[37] Hesham Azzam, Jonathan Cook and Stuart Driver. "FUMSTM Technologies for Verifiable Affordable Prognostics Health Management (PHM)," 2004 IEEE Aerospace Conference Proceedings, April 2004.

[38] Hesham Azzam and Mike Andrew. "A Modular Intelligent Data Administration Approach for Helicopter Health and Usage Monitoring Systems," Journal of Aerospace Engineering, Proc Instn Mech Engrs Vol. 209, IMechE 1995.

[39] Hesham Azzam and Neil Harrison. "A Demonstration of the Feasibility and Performance of an Intelligent Management System Operating on HUMS In-Service Data," CAA Paper 99006, 1999

[40] Hesham Azzam, Iain Hebden, Frank Beaven and Malcolm Wallace. "Fusion and Decision Making Techniques for Certifiable, Affordable Structural Prognostic Health Management," 2005 IEEE Aerospace Conference Proceedings, 2005.

[41] Hesham Azzam and Jason Hazell. "Appraisal of Advanced Diagnostic Strategies Using A Suite of Helicopter Fault Simulations," COMADEM Proceedings, 1996.

[42] Hesham Azzam, Frank Beaven, Andrew Smith and Malcolm Wallace. "FUMSTM Fleet Management Applications and Algorithms," 10th Joint DoD/NASA/FAA Conference on Aging Aircraft 2007.

[43] Alan Draper, "The Operational Benefits of Health and Usage Monitoring Systems in UK Military Helicopters," Third International Conference on Health and Usage Monitoring - HUMS2003, DSTO Platforms Sciences Laboratory, Australia, December 2002

[44] A Coupled Rotor Fuselage Vibration Analysis for Helicopter Rotor System Fault Detection", Yang, 2002, PhD Thesis, University of Maryland.

[45] Aeroelastic Optimization of Advanced Geometry and Composite Helicopter Rotors", Ganguli, 2004, PhD Thesis, University of Maryland.

[46] Identification of Helicopter Dynamics using Recurrent Neural Networks and Flight Data", M. Vijaya Kumar, S. N. Omkar, R. Ganguli P. Sampath, and S. Suresh, Journal of American Helicopter Society, April 2006.

[47] Report on the Accident to Sikorsky S76A Spirit G-BGXY at South Kirkton, Aberdeenshire on 12th March, 1981", AAIB, 1983 (AAIB Aircraft Accident Report No. 9/83)

[48] AAIB Bulletin 1/88, AS332 Super Puma G-PUMD, 1988.

[49] AAIB Bulletin 10/85, Bell 214ST G-BKFN, 1985.

[50] MHS Aircraft 9M-STT Ditching- Analysis of HUMS Data", Salter, 2005. Smiths Aerospace Report REP1649(1).

[51] "Report on the Incident to Aerospatiale AS332L Super Puma, G-PUMH over North Sea on 27th September, 1995", AAIB, 1998 (AAIB Aircraft Incident Report No. 2/98)

[52] Canadian Forces Flight Safety Investigation Report number 1010-CH146420 (DFS 2-2), June, 2005.

[53] Intelligent Management of HUMS Data: Development and Demonstration of a Vibration Health Monitoring Anomaly Detection System", Larder, Callan and Salter, American Helicopter Society 62nd Annual Forum, May 2006.

[54] Intelligent Management of Helicopter VHM Data: Application of Advanced Analysis Techniques In-Service Interim Report on Phase 1 of the Research Project", Callan, 2007, Smiths Aerospace Report REP1697(2).

Appendix A Summaries of MJAD Reports

1 Introduction

Appendix A presents summaries of six reports produced for the CAA in 1991, containing the results of a study performed by MJAD to investigate diagnostic techniques using mathematical models and in-service data for the detection of faults in helicopter rotor systems. These reports are:

i) MJAD/R/60/91 dated 21 January 1991

ii) MJAD/R/63/91 dated 27 March 1991

iii) MJAD/R/69/91 dated 30 May 1991

iv) MJAD/R/75/91 dated 26 September 1991

v) MJAD/R/86/91 dated 15 November 1991

vi) MJAD/R/90/91 dated 18 December 1991

To give a better appreciation of the work documented in the above reports, summaries are included of three further reports that pre-date these:

vii) MJAD/R/17/88 dated 1988

viii) MJAD/R/28/89 dated 23 November 1989

ix) MJAD/R/30/90 dated 15 January 1990

2 Notation

For clarity, the definitions of terms that were introduced in MJAD/R/90/91 and used in this summary are presented in following paragraphs.

The term **relative displacement** refers to track or lag displacement of a faulty blade minus track or lag displacement of a master blade.

Frequency perturbation (or frequency change) refers to the difference between the natural frequency of an ideal blade and that of a faulty one.

Signal and system parameters: Statistical moments such as averages and standard deviations are parameters. Parameters also refer to natural characteristics such as frequencies, damping coefficients, etc.

A **measurement** refers to a time series, e.g. vibration signal, displacement signal, pressure trace.

N-dimensional measurement space is a set of records that are often simultaneously acquired. Each record describes a certain condition and consists of N measurements.

A **feature (or an observation)** is extracted from a measurement by appropriate operations. For example, the average and standard deviation of a displacement signal are features. Vibration harmonic contents are features. The autocorrelation or the probability density of a signal and the cross-correlation between two signals also provide a series of features.

N-dimensional feature space (state space) is a set of records that are often simultaneously acquired. Each record describes a certain condition and consists of N features.

A **vector (state vector)**: each record (e.g. a record of features) may be represented by a vector whose components are the record elements (e.g. the values of features).

A **mapped (pre-processed) feature** fm is an arbitrary defined function of the features f1, f2, f3, ..., fn that belong to a single record (e.g. fm = (f1+ f2)/f1).

A **primary fault** influences a single blade property (i.e. lift coefficient or flap-wise natural frequency).

A **compound fault** is a combination of primary faults.

A **linear fault** is a fault that induces vibration and displacement disparities in a linear manner, which is proportional to its severity.

A **non-linear fault** is a fault that induces vibration and displacement disparities that are non-linearly correlated to the fault severity.

For a **linear compound fault**, the rules of linear composition apply and, hence, the symptoms of the compound fault are the sum (or the vector sum) of the symptom values of the primary components. For a non-linear compound fault, the linear rules do not apply.

3 MJAD Report MJAD/R/17/88 Dated 1988 [24]

Title: An assessment of the feasibility of simulating and early detection of potentially catastrophic main rotor faults

Author: Hesham Azzam

The following summary focuses specifically on the content of the above document that is directly relevant to rotor health monitoring.

3.1 Introduction

The document presented a review of potential rotor system faults and associated health monitoring techniques. It argued that an important first step to establish the requirements of a successful health monitoring technique is to consider the aircraft design methodology and to assess, through simulations for example, the effects of various faults on the characteristics of helicopter components. The objective of the document was to assess the feasibility of providing early fault detection and discrimination based on the simulation of main rotor catastrophic faults, and to perform an extensive literature survey to establish a philosophy that could have potential benefits for main rotor health monitoring.

Traditionally, the rotor system has been assessed by a safe life procedure aided by health monitoring and non-destructive/ground tests as well as pilot reported events including impacts, exeedances of torque and r.p.m limits, etc. Modern rotor system designs have made increasing use of damage tolerant concepts, higher harmonic control, composite materials and elastomeric bearings which can improve the dynamic performance and markedly reduce the vibration levels and fracture propagation rates. Nevertheless, fracture propagation rates can be very high, suggesting that on-board fault detection and alerting capabilities would be advantageous. At the time of the review (1988), however, these capabilities were not considered to be practically achievable options.

It was noted that health monitoring systems should be capable of discriminating between abnormalities induced by faults and effects of manufacturing tolerances, maintenance or operational induced variability. An assessment of detection methods for rotor faults, as well as effects of these faults on structural properties and aeroelastic characteristics, would be required to aid the development of an efficient and reliable main rotor health monitoring system capable of the early detection of catastrophic failures.

3.2 Detection of Rotor System Faults

The rotor system includes the rotor blades, hub, dampers, rotating anti-vibration features, and rotating control systems. Generally any crack, defect or damage results in reduced stiffness and a consequent change in natural frequencies. The health monitoring technique must discover the fault before the residual strength of a component falls below an acceptable safe level.

3.2.1 Faults of Metal Components

Several methods have been developed to detect small spar cracks including: electrical systems (detecting an open circuit or capacitor discharge) and differential pressure systems (where the spar is sealed and then evacuated or pressurized) detecting a loss of pressure differential if a crack occurs. On-board strain measurement systems have been evaluated, and these rely on understanding metal fracture modes and quantifying failure rates by implementing fracture mechanics. Other methods include periodic component visual inspections, vibration monitoring, and using hollow components filled with a highly penetrating, highly visible dye that would bleed through a crack.

Generally, a gradual growth in crack length causes a gradual reduction in the values of material constants of initially homogeneous and isotropic materials. Test specimens have been used to measure and theoretically predict the degradation of material constants with increased crack length.

The increased roughness of rotor blades due to corrosion, erosion, any induced operational damages etc., causes an increase in drag and flow turbulence, and can cause premature blade stall and excessive oscillatory pitch link loads above certain airspeeds.

3.2.2 Faults of Composite Components

Composites provide the ability to selectively stiffen structural load paths, and show excellent damage tolerance characteristics because of slow defect growth - damage can be indicated by change in natural frequencies and decreasing stiffness before the structure's integrity is compromised.

Failure mechanisms in composites are different from those in homogeneous materials. While homogeneous materials fail as a result of the initiation and growth of a single dominant crack, composites can sustain many cracks in the weak phase before ultimate failure. Composites exhibit complex damage failure modes such as matrix cracking, fibre-matrix interface debonding, interfacial delamination, and fibre fracture. Damage tests have shown that composite strengths can remain almost the same up to a certain damage threshold (or impact energy), after which strengths fall as the damage area increases.

Stiffness reduction has been shown to be a valuable means of damage interpretation. For example, a composite main rotor blade of an AH-1S Cobra that had accumulated 200 flight hours in addition to 25 hours of whirl testing, was subjected to a loading spectrum generated from a flight load survey. After 22 million cycles, an ultra-sonic inspection of the blade root area was conducted because a root strain gauge yielded high strain readings, which were an indicator of about a 30% reduction in the value of the root stiffness from the value measured at 10.7 million cycles. Increased delamination was detected, and the test was resumed until 28 million cycles. Progressive reductions in the flap-wise stiffness and the torsion stiffness were observed at the root section.

Several non-destructive inspection techniques have been developed to evaluate composite structures. The X-ray and bore-scope inspections are usually used in experimental investigations. Thermography (heat pattern observation) is particularly useful for rapid scanning and the detection of manufacturing imperfections and in-service damage state. Acoustic emission has been used to detect matrix cracking, delamination, debonding and fibre fracture.

3.2.3 Faults of Bearings

Approaches to monitoring rolling bearing faults include oil debris monitoring and vibration analysis. The loss of stiffness may be regarded as a fault criterion for elastomeric bearings, however, visual inspections may provide an adequate means of detecting incipient failures. The objective of elastomeric rotor head bearings is to improve reliability while reducing maintenance; the elastomeric bearings eliminate periodic lubrication requirements as well as unscheduled down time caused by seal leakage. Alternate laminates of rubber and steel can be used to react blade centrifugal force while deflecting through pitch, flap, and lag motions.

3.3 Simulation of Main Rotor Faults

3.3.1 Theoretical Models for Health Monitoring Techniques

Mathematical models can be implemented to achieve health monitoring goals, minimize the required number of sensors, provide fault isolation, and improve the reliability of the system.

Helicopter rotor systems have an asymmetrical flow around the disc, and also a complicated wake that is not carried away by the free stream. The wake plays an important part in determining the dynamic response of the blades and, hence, the vibratory forces and moments transmitted to the fuselage. The behaviour of the rotor is also strongly influenced by the motion of the blade and its elastic response. An efficient prescribed wake model has been developed and implemented into a helicopter rotor program, together with overall trim programs, and the introduction of blade elastic deflection effects [25]-[30]. The adequacy of the wake model in helicopter rotor behaviour analysis has been established by comparing its results with a set of consistent and coherent experimental data collected from different sources. The reasonable qualitative and quantitative trends of the mathematical model suggest that this is a suitable candidate for the development of a rotor health monitoring capability. The first step to achieve this goal is to assess the feasibility of simulating main rotor faults.

3.3.2 Mathematical Models for Main Rotor Faults

The development of adequate mathematical rotor models, and the simulation of various main rotor faults, provides a valuable tool which could be implemented to discriminate between various faults with economy and efficiency. Such models provide information about the minimum number of sensors, as well as the best sensor locations and operating regimes required to identify each fault, avoiding trial-error procedures for fault diagnosis.

The review above suggests that main rotor faults can cause local changes in material properties. Therefore, the theory of generalized functions can be used to simulate most main rotor faults. For example, span-wise or chord-wise mass imbalance and stiffness reduction due to blade damage can be simulated by 'Dirac Delta' (the Dirac Delta, often referred to as the unit impulse function, can be informally thought of as a function d(x) that has the value of infinity for x = 0 and the value zero elsewhere). Then the effects of the faults on the natural frequencies and mode shapes, and on the dynamic and aerodynamic behaviour or the rotor, can be evaluated. Furthermore, the theory of generalized functions can be utilized to study the effects of impact loads due to loosening in control system components. Alternatively, tab errors could be simulated by introducing the tab aerodynamic characteristics into the mathematical models.

3.4 Conclusions and Recommendations

This study was initiated to assess the feasibility of simulating catastrophic main rotor faults to provide a mechanism for the early detection of these faults.

On the basis of experimental evidence, it was concluded that main rotor faults result in considerable reductions in structural properties.

Mathematical models that are capable of simulating fault effects and discriminating between faults were proposed. It was recommended that these models, describing the aerodynamic, dynamic, and elastic behaviour of a helicopter main rotor, be implemented for rotor health monitoring.

4 MJAD Report MJAD/R/28/89 Dated 1989 [22]

Title: Configuration and output of a computer mathematical model simulating potentially catastrophic main rotor faults

Author: Hesham Azzam

4.1 Introduction

An extensive data survey on potentially catastrophic main rotor faults (PCFs) was carried out which highlighted symptoms associated with certain PCFs. To obtain a wider appreciation of these, MJAD implemented a mathematical model on a PC to simulate a variety of fault conditions. This document discusses the mathematical model of the various PCFs and describes the general computer program.

4.2 Mathematical Model

While the comprehensive mathematical model of the helicopter requires considerable computational resources, the PCFs have been simply modelled at a macro-symptom level, i.e. as changes in structural stiffness, bearing clearances etc., without recourse to microscopic studies. Such an approach offers insight into the fault signature patterns in terms of measurable aerodynamic, aeroelastic and vibration trends.

The data survey indicated that a major group of main rotor defects result in detectable reductions in structural properties and residual strengths. Accordingly, the first category of PCFs were modelled by considering the reductions in associated stiffness. A blade crack or delamination was simulated by reducing the local stiffness using a Dirac Delta function at any arbitrary radial location. Dirac functions can be of any size in which case their 'strength' is defined by duration multiplied by amplitude. The coefficient of the Dirac Delta defines a damage severity, which has been calculated at arbitrary reference points that vary along the blade radius in sympathy with the moment carrying capability. The above model has also been applied to torsion and drag-wise elastic deflections. An alternative "fracture hinge" model was also implemented for comparison purposes, but not pursued further. These mathematical concepts were used to simulate the first category of faults, including:

i) composite blade delamination to fracture at blade root;

ii) composite blade delamination to fracture at any radial position;

iii) metal blade fracture at any radial position;

iv) swash-plate or control rod faults that result in changes in control system stiffness;

v) jammed flap-wise, drag-wise, or feathering hinges caused by bearing faults;

vi) jammed damper causing a rise in root stiffness; and

vii) crack development in lugs.

The second category of faults identified is caused by manufacturing and operational tolerances that may accumulate and result in dissimilarities of structural properties. They can give rise to induced vibration and/or elevated stresses on the helicopter leading to further damage and catastrophic failures. The individual blade concept has been used to simulate these faults by assigning different structural properties to the faulty blade.

The third category of PCFs is component faults. In this case a fault develops in one element of a component, then progressively affects the complete component.

Consequently, a staged mathematical model has to be applied. An example of this category is the blade root retention fault that caused a fatal accident of a S76A in 1981. A centrifugal retention component was designed to carry centrifugal forces only; the shear and moments were reacted by shear and elastomeric bearings. An increased clearance in the shear bearing modified the load paths. In particular, the retention element was subjected to a bending moment that resulted in catastrophic fracture. In the mathematical model, the effective stiffness of the faulty blade is dependent upon the clearance in the shear bearing. The distance between the shear bearing and the retention component has been used to re-evaluate the effective flapping hinge offset and update the frequencies of the faulty blade. A two- stiffness approach is utilised as the blade fluctuates above and below a calculated reference angle. The first stiffness is that of a blade restrained at the retention component, and the second stiffness is that of the blade restrained at the shear bearing. Finally, the model of the first category of faults has been applied to simulate the crack developing in the retention component.

The fourth category of faults relates to changes in damping and arises from inoperative dampers or loss of elastic characteristics caused by delamination or cracking. The simulation of these faults required the introduction of a modified damping coefficient into the blade response equation.

The final fault category covers aerodynamic related discrepancies. In this case, the fault modifies the aerodynamic characteristics of the faulty blade or its angle of attack. An example of this fault type is a leading edge strip lifting prior to detachment. Equally, excessive clearance in control system fittings and linkages combined with severe vibration of the rotor head may cause a random variation of the pitch angle of the blade. This fault has been simulated by generating a series of random numbers representing the variation in control angle with azimuth. The maximum amplitude of the random angle is a function of the damage severity.

4.3 Program Trim Description

A prescribed wake model was implemented which provides good correlation between prediction and measurement. A general trim algorithm was also utilised which can combine any arbitrary coupled trim equations with non-uniform induced velocity profiles, non-linear aerodynamic data, and arbitrary elastic deformation models.

The program accommodates several user requirements via the trim options. The first is used to investigate fault effects on frequencies and mode shapes only, and the trim analysis is not performed. The second trim option applies to ground runs with various collective angle and rotor speed settings. The third option provides a very quick look at control parameters in trimmed flight based on a simplified analysis, which does not include fault effects.

Considerable effort was spent on transforming the blade non-linear elastic response differential equations of the flap-wise, drag-wise, and torsion motions to a set of linear equations. This greatly improved computational efficiency. The fourth trim option makes use of this analysis and provides a reasonable and efficient solution of the coupled trim equations, the equations of elastic motions and the Glauert wake. (In forward flight, the Glauert effect is a fore-to-aft linear induced flow variation due to the wake skewing). The fifth option solves the equations of elastic motions using the same analysis, utilises the Glauert wake, and calculates the rotor forces and moments using detailed non-linear aerodynamic data. The sixth trim option evaluates forces, moments and elastic deformations using detailed non-linear aerodynamic data and the Glauert wake, or the representative non-linear non-uniform wake model. Trim options 4 to 6 can be invoked for helicopters with or without PCFs.

The precision of the calculations is chosen by setting parameters which define a precision for degrees of freedom calculations, a precision for the analytical trim procedure, and a precision for detailed trim evaluations respectively. The output of the program consists of the following:

i) brief list of blade parameters;

ii) detailed presentation of natural frequencies and mode shapes;

iii) detailed output of the harmonics of each degree of freedom and blade deflections at four radial locations; and

iv) detailed information about the helicopter hub vibratory loads, fixed frame.

4.4 Conclusions

A representative mathematical model that describes the helicopter behaviour adequately and simulates various PCFs was implemented on a PC. The program combines a representative non-uniform induced velocity wake, detailed non-linear aerodynamic data, a robust trim procedure, and a blade elastic response model along with several mathematical models for various faults. A preliminary, brief study on fault effects was carried out in order to demonstrate and test the capability of the program.

The output of the program is configured to aid discriminatory fault identification and early detection strategies. The measurable deflections at blade tip (track, lag and torsion), along with six vibratory forces and moments, lay the foundation for a comprehensive comparison between various fault effects. This represents an essential step toward constructing a concise and efficient diagnostic system.

5 MJAD Report MJAD/R/30/90 Date 1990 [23]

Title: The identification of potentially catastrophic helicopter rotor faults (theoretical study)

Author: Hesham Azzam; Date: 15 January, 1990

5.1 Introduction

This document presents the symptoms associated with a suite of primary faults, which were simulated in the mathematical model described in MJAD/R/28/89.

5.2 Classification of Faults

The following categories of faults were identified in MJAD/R/28/89:

i) Faults that induce structural stiffness changes.

ii) Structural dissimilarities caused by manufacture and operational tolerances.

iii) Component faults; a fault develops in one element of a component then progressively affects the complete component, e.g. the S76 blade spindle failure.

iv) Faults that induce changes in structural damping.

v) Faults that induce aerodynamic discrepancies.

Generally, a fault can be classified as either a primary fault or a combination of primary faults, i.e. a composite fault. In practice, it is more likely that a composite fault will be encountered. However, it is probable that a primary element of a composite fault will be dominant. For example, a crack or delamination may cause unequal reductions in flap-wise, drag-wise, and torsion stiffness. The analysis of primary faults provides an essential tool to understand the signatures of the composite faults, and establish a reliable foundation from which a powerful diagnostic strategy can be built.

A list of the primary faults, along with examples of defects containing these faults, is presented in Table 1.

Table 1 List of Primary Faults

Primary fault	Defect examples
Flap-wise stiffness reduction at certain radial station	composite blade delamination metal blade fracture and or lug cracks; categories 1 to 3.
torsion stiffness reduction at certain radial station	composite blade delamination metal blade fracture control system bearings faults, swash-plate, pitch link, or lug cracks; categories 1 to 3.
Drag-wise stiffness reduction at certain radial station	composite blade delamination metal blade' fracture, lug and blade retention faults; categories 1 to 3.
Flap-wise stiffness rise	jammed flapping hinge, structural dissimilarities; categories 1 to 3.
torsion stiffness 'rise	jammed feathering hinge, control system bearings faults, structural dissimilarities; categories 1 to 3.
Drag-wise stiffness rise	jammed lag hinge or damper, structural dissimilarities; categories 1 to 3.
a change in structural damping	Inoperative damper or loss of elastic characteristics caused by delamination; category 4.

Table 1 List of Primary Faults (Continued)

a change in lift coefficient of a portion of the blade	leading edge strip lifting prior to detachment, excessive clearance in control system, tab and pitch link adjustment errors; category 5.
a change in drag coefficient of a portion of the blade	
a change in pitching moment coefficient of a portion of the blade	
mass imbalance	rotor head mass imbalance
pitch link adjustment error	incorrectly adjusted pitch link

Thirteen groups of primary faults were analysed and, within each group, the severity of the defect was increased progressively through 5 stages. Accordingly, a total of 65 faults cases were studied.

The primary flap-wise, drag-wise and torsion damages cause reductions in flap-wise, drag-wise and torsion stiffness respectively. This in turn affects the associated natural frequencies. The damage severity percentage has been calculated at arbitrary reference points that vary along the blade radius in sympathy with the net moment carrying capability. The severity of the defect is best assessed by the percentage change in natural frequencies.

The primary "jammed flapping hinge" and "jammed lag hinge" faults cause a rise in the flap-wise, and drag-wise stiffness respectively, and a corresponding rise in the natural frequencies. The 100% defect severity case converts the articulated blade arrangement to a rigidly mounted blade arrangement. The defect severity is a relative measure and the faults' effects on frequencies are a clear indicator of the damage severity.

The control system primary faults cause changes in the control system stiffness and associated changes in the torsion frequencies. Mass imbalance, and pitch link adjustment errors are included to see whether it is possible to discriminate between catastrophic and adjustable faults.

5.3 Typical Fault Signatures of a 5-Bladed Articulated Rotor

Considering the natural mode shapes and normalised moment mode shapes of a rotating articulated blade, a fracture or delamination induced at a radial position at which the contribution of a mode to the net moment is high will generally cause a considerable reduction in the frequency of this mode. Conversely, a delamination or crack at a radial station where the value of the normalised moment is almost equal to zero will cause virtually no change in the mode frequency. Therefore the location of the damage determines the relative reductions of the natural frequencies of different blade mode shapes. Table 2 presents an extract from the data in MJAD/R/30/90 defining the changes in blade natural frequencies as a result of different fault severities for four fault groups (three groups being the same fault type, but at different locations).

Table 2 Effect of Faults on Blade Natural Frequencies

S No	fault description	relative intensity of defect	changes in the blade natural frequencies, % flapwise frequencies 1st	2nd	3rd	4th
1	flapwise damage at 0.25 R	20%	0.000	0.281	1.810	2.410
2	flapwise damage at 0.25 R	40%	0.000	0.700	4.361	5.262
3	flapwise damage at 0.25 R	60%	0.000	1.387	8.142	8.611
4	flapwise damage at 0.25 R	80%	0.001	2.711	14.08	12.44
5	flapwise damage at 0.25 R	100%	0.003	6.200	23.87	16.60
6	flapwise damage at 0.50 R	20%	0.000	2.254	0.001	6.211
7	flapwise damage at 0.50 R	40%	0.000	5.803	0.003	13.63
8	flapwise damage at 0.50 R	60%	0.001	12.15	0.007	22.59
9	flapwise damage at 0.50 R	80%	0.005	26.36	0.021	33.40
10	flapwise damage at 0.50 R	90%	0.014	42.63	0.070	39.44
11	flapwise damage at 0.70 R	20%	0.000	1.41	4.634	1.189
12	flapwise damage at 0.70 R	40%	0.000	3.70	10.38	2.452
13	flapwise damage at 0.70 R	60%	0.000	7.90	17.47	3.771
14	flapwise damage at 0.70 R	80%	0.001	17.21	25.78	5.124
15	flapwise damage at 0.70 R	100%	0.005	43.64	33.87	6.482
16	jammed flapping hinge	20%	−0.95	−0.98	−0.87	−0.95
17	jammed flapping hinge	40%	−2.49	−2.64	−2.50	−2.45
18	jammed flapping hinge	60%	−3.65	−3.94	−3.89	−3.75
19	jammed flapping hinge	80%	−5.51	−6.15	−6.44	−6.23
20	jammed flapping hinge	100%	−8.04	−9.38	−10.6	−10.5

The effect of a jammed hinge is to increase all the natural frequencies. However, the increased fundamental frequencies dominate the effects of other modes when the blade force response is evaluated.

Moderate defect severities result in negligible changes in mode shapes. High values of defect severity effect a change in mode shapes around the defect region, and hence, the blade deflection deviates from the deflection of the perfect blade. However, the effect of mode shape deviations on the force response is of a second order nature compared with the effect of deviations in natural frequencies.

5.4 Discrimination Between Faults

The signatures described in the previous section can be used to discriminate between faults. The discrimination strategy was demonstrated by analysing the force response parameters of an example trimmed, articulated 5 bladed helicopter at hover, 60 and 120 knots forward speed. The defined 'symptoms' cover 1R, 2R, 3R, and 4R vibration components of forces and moments along with the relative blade track and lag measurements and torsion displacements at 0, 90, 180, and 270 degrees of azimuth.

5.4.1 Fault Discrimination Table

The results of the analysis were used to construct a fault discrimination table. An extract from this is presented in Table 3.

Table 3 Fault Discrimination Table for 120 Knots Condition

fault group	1	2	3	4	5	6	7	8	9	10	11	12	13	14
1R X-force,		–	@x					**	–			x	x	
2R X–force		–	@x	–	–	@x	@*	*	X		–		–	–
3R X–force		–	–		@*	@x	X	–	–					–
4R X–force					–	@x	@*							
1R Y–force		–	@x					**			–	–	–	–
2R Y–force														
3R Y–force		–	–	–	@*	@x	–	–	–					
4R Y–force														
1R Z–force		–	X	*				*	@*	–	*		*	*
2R Z–force	–	**	X	–				–	@–					
3R Z-force	–	X	–				@x	–	@*	–	–			@x
4R Z-force	X	X	X	–	–	–	X		@*		@x			@*
1R X–moment	–	X	**	X					X		–		–	–
2R X–moment				**				–	–		–		–	–
3R X–moment		X	–	**										
4R X-moment														
1R Y-moment														
2R Y-moment														
3R Y-moment														
4R Y-moment														
1R Z–moment		–	@x	X				**	@*		X		X	X
2R Z–moment		–	@*	X	@x	@–	@–	**					X	
3R Z-moment			@*			@–	@x	X	X				–	–
4R Z-moment		*	*	X	X	X	X	–	@*	X	@*			x
track at 0.0		-x	-*	+*				-x	+-	+-	--		--	+x
track at 90.	+-	+-	+*	+*				+-	+x	+-	-x		-x	+x
track at 180		+x	+*	+x				--	+*	+-	-x		-*	+-
track at 270		--	-X	+-				+-	+*	+-	-x		-*	+x
lag at 0.0		+x	+*					-*	-x		+-		+x	--
lag at 90.		+x	+*					-*	-x		+-		+x	--
lag at 180		+x	+*					-*	-x		+-		+x	--
lag at 270		+x	+*					-*	-x		+-		+x	--
twist at 0.0														
twist at 90.									+x	+x	-*			+*
twist at 180.									+*	+-	-*			+x
twist at 270									+*		-X			+-

** : first order high force or moment
* : first order low force or moment
x : second order
\- : third order
-* and +~ : negative and positive first order relative displacement
-x and +x : negative and positive second order relative displacement
–– and +- : negative and positive third order relative displacement
@ : peak for one fault within a group

As an example of the interpretation of the discrimination table, a jammed damper fault can be considered (fault group 8 in Table 3), which is characterised by a high 1R X-force. However, as the fault severity reduces the level of confidence associated with its correct identification reduces and the probability of the existence of fault groups 12 and 13 representing typical 'tuning' adjustments increases, raising the possibility of a fault-masking problem. However, if attention is paid to blade track and lag trends from hover to 120 kts, further discriminatory information may be obtained. First, a jammed damper and mass imbalance (groups 8 and 12) have little effect on the track at 180 degrees of azimuth, whilst the pitch link adjustment (group 13) effects a noticeable track perturbation. Second, considering lag, the jammed damper has a dominant affect whilst the adjustable faults have moderate or negligible effects. Finer inspections of the track and lag trends could be pursued by considering the relative magnitudes between the displacements, both in terms of track/lag ratios and as functions of rotor azimuth. Finally, the information on the effects of faults on blade natural frequencies may be exploited which, for the example, indicates that the jammed damper affects the fundamental blade lag frequency, whilst the adjustable faults have negligible effect.

By analysing the information in the table in this way a preliminary diagnostic strategy may be constructed. Consideration then needs to be given as to whether this information can be used in a practical system.

5.4.2 Sensor Fit and Test Envelope

Clearly, accelerometers would form part of any sensor fit to supply information on the low frequency vibration orders, i.e. 1R, 2R etc. To make an assessment of the rotor induced forces implicit in the fuselage motion, it may be prudent to consider a minimum of six such accelerometers. Their output could be processed with the underlying assumption of rigid body motion, to yield three translations and three rotations. The only extra proviso is that the accelerometers would have to be placed in a minimum of three physical locations on the fuselage (two tri-axial accelerometers cannot supply the desired data), and preferably on major frames. The second sensor type would measure blade positions, both in track and lag. While one such device would be imperative and most ideally used to record information at a rotor azimuth of 180 degrees (i.e. at the nose of the aircraft), a second such sensor may be necessary to fully discriminate between faults. It is postulated that sufficient promise has been indicated in the discrimination tables as to not warrant placing sensors in the rotating frame.

While the test envelope would inevitably include steady state conditions (hover, 120kts etc), it would also be prudent to exploit unsteady conditions. The rationale for this is that a number of potentially catastrophic faults affect the natural frequencies of the individual blades. Transient conditions would excite such natural frequencies and, accordingly, may offer considerable discriminatory information. One useful test would be to collect vibration and blade positional information whilst the main rotor is running up or down on the ground. During this transient, various natural blade frequencies temporarily coincide with forcing frequencies which are an integer multiple of the speed of the rotor. Accordingly, for short periods, a given blade mode may be in resonance. Subtle differences in blades would then become maximised and may highlight modal disparities.

The above requirements are not inconsistent with current HUMS technology, however, currently installed systems do not have the sensor fits to provide all the data needed.

5.5 Conclusions

The PC program implementing the mathematical model described in MJAD/R/28/89 was utilised to analyse 65 fault cases in all. The faults included blade fracture development, binding or jammed dampers/hinges and control system irregularities along with two 'tuning' faults, namely pitch link and mass adjustments. Discrimination tables have been presented and used to illustrate discriminatory fault identification strategies. These strategies are based on measurements that can be realised with today's technology.

It was argued that a minimum of six accelerometers and one (with possibly the need for a second) blade positional measurement sensor would be required for a health monitoring system capable of discriminating between rotor system faults.

Measurements from both steady and transient helicopter operating conditions would also be required. The latter was considered necessary to highlight changes in blade natural frequencies, which are symptomatic of a number of potentially catastrophic faults.

6 MJAD Report MJAD/R/60/91, 1991 [2]

Title: Effects of helicopter main rotor faults on fuselage vibration modes

Authors: Hesham Azzam and Mike Andrew

6.1 Introduction

MJAD/R/28/89 presented a representative mathematical model that describes the helicopter behaviour adequately and simulates various main rotor catastrophic faults. The program combines a representative non-uniform induced velocity wake, detailed non-linear aerodynamic data, a general trim procedure, and a blade elastic response model along with several mathematical models for various faults. MJAD/R/30/90 described how the model had been utilised to analyse a range of different faults to generate a set of fault discrimination tables, which were then used to illustrate a discriminatory fault identification strategy.

The next step was to evaluate the helicopter response at various non-rotating locations to minimize the number of sensors and determine the optimum sensor locations for a HUMS capability targeted at detecting main rotor faults. The report presented a helicopter fuselage response mathematical model, which was then used to carry out a preliminary investigation of the signatures of main rotor adjustments and catastrophic faults as measured by sensors mounted at various fuselage positions.

6.2 The Fuselage Response Model

The comprehensive computer program described previously was utilized to calculate the hub loads induced by main rotor adjustments and catastrophic faults. The fuselage rigid and elastic motions are then considered to determine the helicopter response to the induced hub loads.

The rigid body response of the helicopter fuselage gives rise to six out-of-trim vibration disturbances; three translations and three rotations. The six disturbances are related through six non-linear, coupled differential equations. The differential equations are linearised by neglecting the second order terms of disturbances and, combined with a harmonic analysis technique, are used to evaluate the required rigid body response.

The helicopter fuselage is a very complicated structure that consists of several elements having certain masses, and stiffness and damping characteristics. The exact prediction of the fuselage elastic response requires an extensive finite element analysis. Nevertheless, the dominant components of the elastic response can be estimated from the transverse, lateral and torsional motions of a simple structure. The associated differential equations of the free vibration are used to calculate the fuselage natural frequencies and mode shapes. The forces and moments acting at the hub are represented by Dirac Delta functions. Modal analysis is applied to derive the differential equations of the elastic degrees of freedom. A harmonic balance technique is used to solve the differential equations. Six elastic modes and six 'rigid body' degrees of freedom are considered. Based on the above analysis, a PC-based fuselage response program was developed and implemented.

6.3 Theory Versus Test

Validation of a mathematical model is a major task that requires a large set of coherent, consistent test data along with accurate geometric, structural and aerodynamic information about the associated helicopters. Although this was not feasible, it was considered that a 'quick look comparison' was desirable to attach a degree of confidence to the conclusions and findings extracted from the mathematical model. Therefore, a five bladed articulated rotor vehicle resembling the S61 helicopter was considered. The vibrations along the fuselage caused by a one kilogram tip mass and a 0.8 degree pitch link change were then predicted and compared with the available test data.

For a sensor fitted near to the helicopter nose (assumed to be located on the elastic axis of the fuselage at a distance of 17 metres from the tail), the difference between theoretical and maintenance manual values of the 1R lateral sensitivity to tip mass adjustment in the hover was 1.3%. Considering the sensitivity of the 1R vertical vibration to pitch link adjustment at 110 knots forward speed, the differences between the theory and tests were 5.8% (maintenance manual) and 4.1% (measured). The results indicated that the mathematical model provides appropriate qualitative results and reasonable trends for the vibration-displacement disparities. Furthermore, just over half the test cases indicated reasonable quantitative predictions. While not intended to provide a full validation, the 'quick look comparison' demonstrated that the model was adequate for the purpose of establishing diagnostic strategies.

6.4 The Importance of the Fuselage Elastic Motion

Typical fuselage mode shapes for a four bladed Puma helicopter were derived. The results indicated that the contribution of the elastic response to the vibration is high at the rear part of the helicopter and is low at locations between 0.6 to 0.8 of the distance from the tail to the nose of the aircraft. The hub vibratory loads produced by a main rotor mass imbalance of 2.5 kilograms located at a distance of 0.64 metres from the shaft centre were predicted by the model. The vibrations induced by these loads at various fuselage locations were evaluated and a comparison between the 'rigid body' response and the 'rigid body + elastic structure' response was undertaken.

Mass imbalance generates 1R longitudinal and lateral forces at the hub centre. The longitudinal 1R force causes a 1R pitching moment about the helicopter c.g. and, hence, induces a vertical vibration along the fuselage. Considering vibration measured along the elastic axis of the fuselage, the inclusion of the elastic motion has little effect on longitudinal and vertical vibration, but influences the lateral vibration considerably at the rear and the front of the fuselage.

A helicopter fuselage is designed such that its natural frequencies do not coincide with the main rotor frequencies. Accordingly, the vibration caused by the elastic motion near to the helicopter c.g. is small. Sometimes however, fuselage structural modifications are introduced in order to satisfy specific operational requirements. The structural modifications may cause considerable changes in natural frequencies and may result in appreciable changes in the vibration signatures. To illustrate this, the hub loads produced by a pitch link discrepancy of 0.2 degrees at 100 knots forward speed were considered. The helicopter fuselage was stiffened such that the frequency of the first elastic mode became very close to the main rotor 2R. The impact of this modification was that the 1R signature at the rear and the front of the fuselage was altered, and a high 2R vibration was induced. This illustrated the importance of the fuselage response model.

6.5 Vibration Modes of Main Rotor Faults

The objective of the report was to present the helicopter fuselage model and not to conduct a thorough investigation of the vibration signatures of catastrophic main rotor faults as measured by sensors located at various fuselage locations. Nevertheless, a preliminary discussion was presented in the report and is summarised in the following paragraphs.

A five-bladed helicopter was considered and the vibration and displacement disparities induced by a pitch link change, a tab maladjustment and a flap-wise blade crack were calculated. The discrimination between various faults can be based on the 1R or 2R vibration 'shapes' as measured by sensors located along the fuselage. Unfortunately, these vibration shapes depend on the damage (or adjustment) severity. This problem can be overcome by evaluating normalised vibration modes (shapes), which can provide a valuable diagnostic tool. The normalised vibration modes provide an invariant property for the fault whose vibration signature varies in a linear manner with the severity. Thus a set of vibration modes could describe a linear fault group. The group may contain an infinite number of cases that possess the same damage type and have different damage severities. For non-linear behaviour, a larger number of vibration mode sets is required. The number of sets is proportional to the order of the non-linearity, e.g. for a second order non-linear fault group, two sets are required. Another invariant fault property is the ratio between the vibration modes, e.g. the ratio between the normalised 1R and 2R vibration modes. It should be noted that at least two sensors are required to initiate a diagnostic process based on vibration modes.

To illustrate this philosophy, the vibration modes associated with the pitch link, tab and blade crack faults were evaluated by normalising the vibration to unity at the helicopter nose. The vibration modes are shown in Figure 1. The normalisation factors are displayed with the title of the y-axis, which indicate the severity of the vibration. The ratio between the normalisation factors for the 2R and 1R modes is an invariant property (for linear effects). Considering the lateral and vertical vibration, the ratio values are presented in Table 4.

Table 4 Ratios of 1R and 2R Modes

	Lateral	Vertical
Flap-wise damage	7.431	9.9071
Pitch link adjustment	1.832	0.6186
Tab adjustment	1.752	0.7211

The above table suggests that discriminatory diagnostic systems would be expected to benefit from this strategy. However, it is important to state that the first step in the discrimination procedure is to test the magnitude of the normalisation factors (or the level of vibration) in order to obtain the invariant properties from the vibration modes which are not contaminated by noise. Considering the relative amplitude of lateral and vertical vibration modes at different fuselage longitudinal stations provides further diagnostic features.

The determination of the optimum number of sensors requires an extensive study to evaluate the linear and non-linear fault properties deduced from the normalised vibration modes.

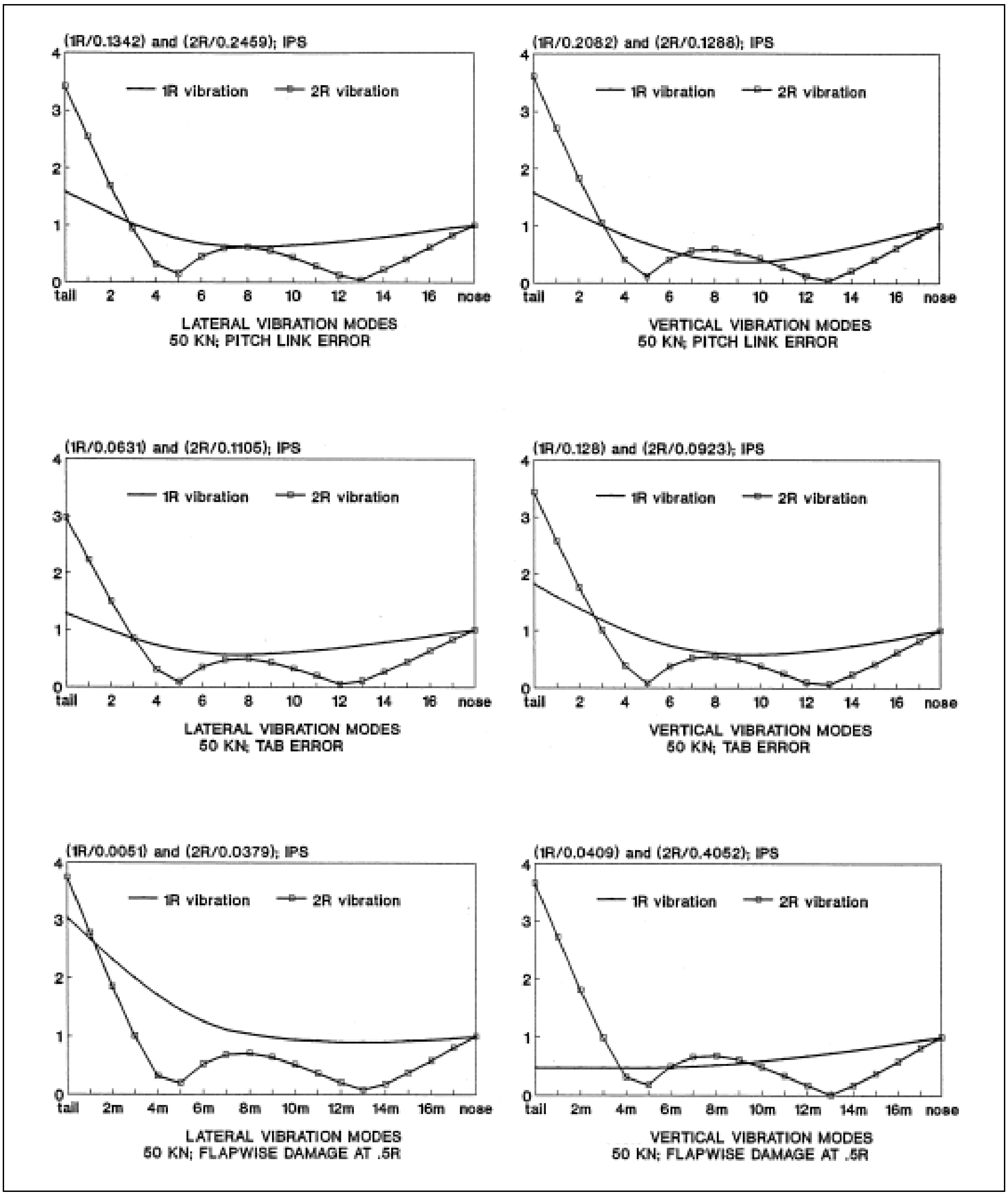

Figure 1 1R and 2R Vibration Modes

6.6 Conclusions

A mathematical model was developed and implemented on a PC to evaluate the response of the helicopter fuselage to the vibratory hub loads induced by main rotor adjustments and catastrophic faults. Comparisons between measured vibration sensitivities and theoretical results indicated a good agreement between test and theory. It was found that the vibration signature is considerably influenced by the fuselage elastic response.

Fuselage vibration modes have been established for a number of faults for potential use within a discrimination strategy. It is expected that the analysis of vibration modes will provide a valuable health monitoring capability. The value and ratio of normalised vibration modes are considered particularly applicable to fault diagnostics.

7 MJAD Report MJAD/R/63/91, 1991 [3]

Title: The application of a math-dynamic model to characterise blade cracking/ delamination faults during rotor start-up and shut-down

Authors: Hesham Azzam and Mike Andrew

7.1 Introduction

MJAD developed a math-dynamic model that could be used to establish diagnostic strategies based on data collected during rotor start-up and shutdown. These two dynamic operations can excite a hidden fault allowing its early detection. A preliminary investigation of the detection methods is presented.

7.2 The Math-Dynamic Model

It can be shown that a natural vibration mode may be modified by the presence of a structural defect in the rotor blade or hub assembly. Accordingly, it is appropriate to isolate helicopter operating conditions where this modified mode can be sufficiently excited so that its early detection may be realised. Practical dynamic manoeuvres are rotor start-up and shutdown. MJAD developed a helicopter math-dynamic model to establish a diagnostic strategy based on data collected during these operations.

The model allows each of the helicopter controls to be specified by up to eight functions. The model considers all of the helicopter five controls; namely, collective pitch, longitudinal cyclic, lateral cyclic, tail rotor thrust (or tail rotor pitch angle) and rotor speed. The rotor speed option, for example, is used on the ground in rotor run-up or run-down operations. The dynamic-elastic model of the main rotor considers full inertia and aerodynamic couplings as well as hub acceleration effects.

7.3 Preliminary Investigation of a Fault Detection Strategy

Throughout the investigation of rotor start-up and shutdown operations, a five bladed articulated rotor vehicle resembling the S61 helicopter was considered. The variation of the articulated blade natural frequencies with rotor speed was plotted. To investigate the effects of blade cracking or delamination, flap-wise damage was simulated on one blade at 50% radius.

The first shut down operation was conducted at minimum collective pitch angle. In the model, the cyclic controls were not applied and there was no wind. An FFT analysis of long vibration records consisting of 1024 data points or more (about 6 seconds) was then performed. Inspection of the results showed a 1R component only present for the damaged condition, suggesting the possibility of using such analysis to identify the 1R component caused by the fault. However, the level of the 1R vibration excited by this operation was small and hence, in a practical environment, the signal may well be lost in the noise.

The level of the vibration caused by the fault can be increased to a detectable level by applying cyclic control and/or operating in moderate wind. However, FFTs of long vibration records calculated with 20kts of wind and cyclic control activation showed that other low frequency energy could easily hide the 1R vibration peak. Additional harmonics, such as 2R and 4R offer no further practical discrimination capability.

The results suggested that the FFT (or power spectral) analysis of long records has a very limited use for fault detection during start-up and shutdown operations. An examination of the time history of the vibration records indicated that the long record FFT failed to offer fault discrimination because of the strong trends in the blade tip track/lag and rotor hub force and moment data with time during the rotor run-down.

In order to overcome the limitation of the 'long' FFT approach, the FFT analysis was repeatedly applied to much shorter records (64 data point constant size blocks of the run-down data). Main rotor harmonics ranging from 1R to 5R were then extracted from the FFT results and plotted against a central time coordinate. The comparison between no-fault and fault conditions presented in Figure 2 for the 1R X-moment data set shows that the refined FFT technique clearly identified a distinguishing pattern. Accordingly, it is recommended that this technique be further investigated and, where possible, benchmarked against in-service data.

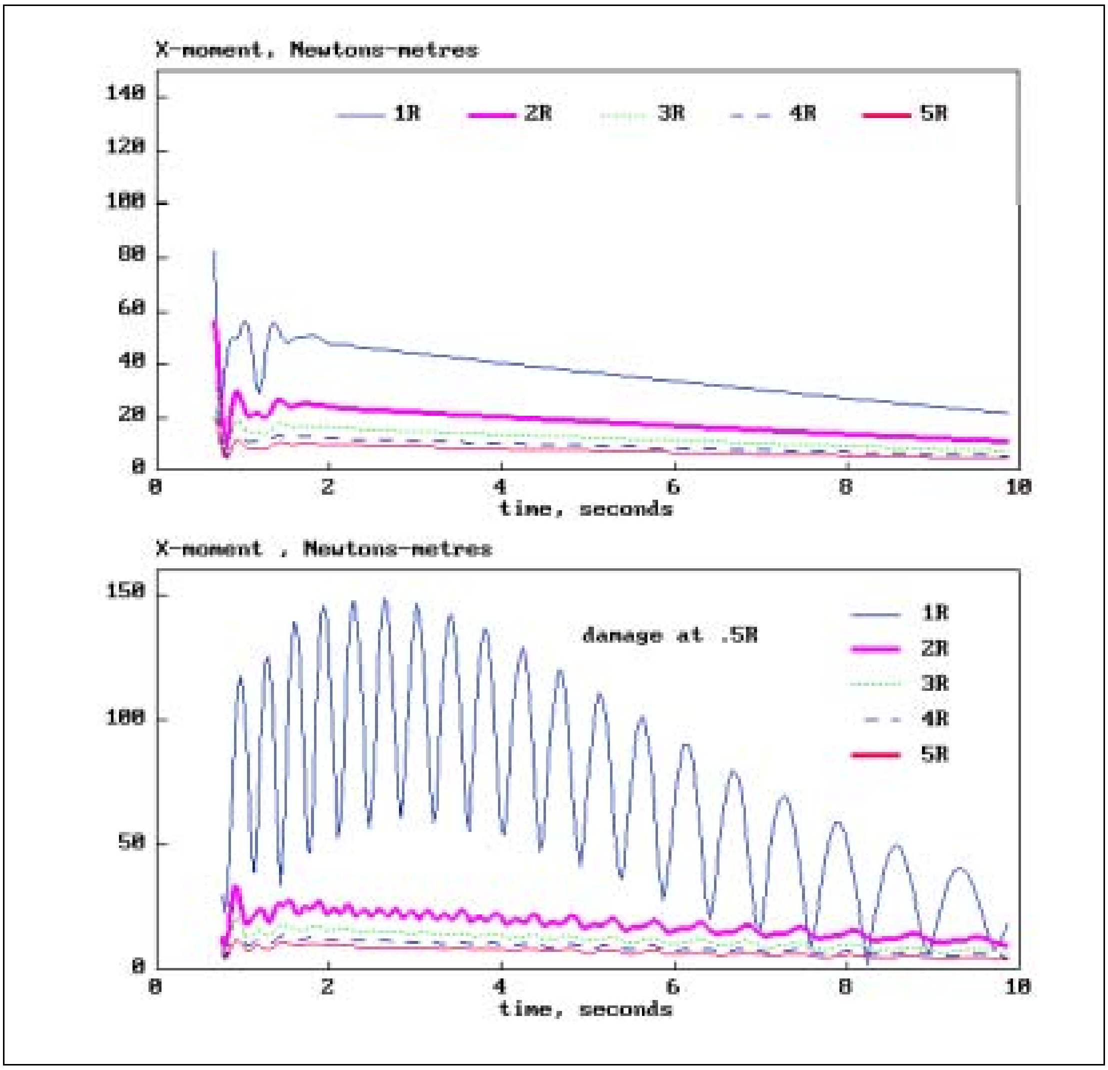

Figure 2 X-Moment Vibration During Rotor Run Down

7.4 Conclusions

A math-dynamic model capable of simulating helicopter operations such as rotor start-up and shutdown was used to carry out an investigation into the detection of cracking/ delamination in main rotor blades during rotor start-up and shutdown operations. Discriminatory patterns in the run-down data were identified once an appropriate data processing technique was established. The model could be utilised in future to search for/define 'acceptable pilot control actions' that excite the natural modes, which may be modified by rotor system faults.

8 MJAD Report MJAD/R/69/91, 1991 [4]

Title: An appraisal of theory-to-test transfer analysis for helicopter vibration prediction

Authors: Hesham Azzam and Mike Andrew

8.1 Introduction

Certain fault detection strategies can be based on differences between predicted normal vibration levels and actual measured vibration levels. Such a strategy requires accurate and efficient (fast) mathematical models. However, the prediction capability of a helicopter mathematical model is often restricted due to limitations of the modelling theories used, lack of detailed data on various helicopter components, effects of aging etc. Therefore, there is a need to establish a prediction strategy that can overcome these difficulties to increase the level of confidence in the forecasted vibration levels.

This report describes a technique that uses MJAD math-dynamic models together with the output of helicopter fuselage sensors to perform theory-to-test transfer analysis. The transfer analysis can provide calibration (or empirical correction) factors to correct the theoretical expectation so that this agrees with the actual measurement. These calibration factors automatically compensate for any simplifying assumptions or the absence of a component model. Thus, the math-dynamic models along with the theory-to-test transfer analysis can be used to accurately predict the expected normal vibration associated with certain operational conditions.

It should be noted that the approach of theory-to-test transfer analysis is a refinement approach that requires a representative mathematical model. The fidelity of the model has to be as close to reality as possible, otherwise the prediction may be very inaccurate. It is essential therefore to demonstrate that the fidelity of the model is adequate prior to the evaluation of transfer functions.

8.2 Theory Versus Test

The results of MJAD/R/60/91 were obtained by invoking the Glauert wake option. The Glauert wake is a simple rotor induced inflow profile that provides adequate qualitative results and requires low computational capacity. It is well known that this profile underestimates the mean value of the rotor induced velocity, especially in hover and at low forward speed. MJAD math-dynamic models can also provide a representative non-uniform induced velocity profile derived from a prescribed wake analysis. In this case, the wake of the rotor is considered to consist of strong tip vortices and near interactive blade vortex sheets. The tip vortices are allowed to contract below the rotor and then expand. The strengths of the tip vortices are related to the flapping response of the blades and the strength of the near interactive wake is determined from the blade loading. The geometry of the tip vortex is established from available flow visualization results and extending these results to cover the speed range using appropriate mathematical models.

In MJAD/R/60/91, ground and high altitude effects in hover and forward flight respectively were not considered in the analysis (for example, at the ground, it is expected that the induced velocity would be less than in free air). It was felt that neglecting these effects might compensate for the low induced velocity predicted by the Glauert wake. In this report, ground effects have been considered in hover and high altitude atmospheric conditions have been incorporated in forward flight. The prescribed wake option has also been utilised.

The results of the detailed analysis were compared with those from flight tests. The difference between theoretical and maintenance manual values of the 1R lateral sensitivity to tip mass adjustment in hover was less than 1.0%. Considering the sensitivity of the 1R longitudinal vibration to pitch link adjustment, the differences between the theory and test were 12.7% in hover and less than 10.8% at forward flight. The prediction of the sensitivities of the 1R lateral vibration to pitch link adjustment was 90.4% accurate at 110 knots; otherwise, the accuracy was better than 96.3%. Regarding the vertical vibration, the prediction accuracy was 84.2% at 70 knots; otherwise, the accuracy was better than 89.5%. The differences between the predicted relative track values and the measured values were 13.5% at 110 knots and less than 4% otherwise. The accuracy of the calculated values of relative lag was better than 87%.

Due to operational variables, the vibration measured by the fuselage sensors exhibits a degree of variability between different flights. The ratio between the standard deviation and the mean value of a vibration component is a good measure of the degree of variability, provided that the mean value is not very small in comparison with the noise level. Typical values of this ratio were 10% to 20%. The differences between the theory and test lie within this range and, therefore, the quantitative agreement between the theory and test is considered to be good.

The results presented in MJAD/R/60/91 based on the Glauert wake suggested that the fidelity of the mathematical model is satisfactory when 1R vibration (or vibration less than bR) is considered. Using the prescribed wake further improves the fidelity of the model. Therefore, the validity of the theory-to-test transfer analysis can be demonstrated by examining the predicted and measured 1R data associated with a pitch link error on the S61 helicopter.

8.3 Theory-to-Test Transfer Analysis

As explained, the prediction capability of a helicopter mathematical model is often restricted due to a number of practical limitations. The theory-to-test transfer analysis is a technique that provides a matrix of calibration coefficients that can be used to correct theoretical predictions. The elements of the matrix are composed of ratios between estimated and measured vibration components (or relative displacements).

A theory-to-test transfer analysis providing usable numeric results requires high fidelity math-dynamic models that can predict the measured values directly. This is achievable for 1R vibration, and for vibration less than bR. For the cases described previously, and for a sensor fitted near the helicopter nose, the transfer analysis provides the calibrations shown in Table 5. The transfer matrix columns associated with 1R vibration (longitudinal, lateral and vertical), relative track displacement and relative lag displacement were evaluated. The elements of the 1R column are composed of the ratio between the measured and predicted vibration vectors. Each element is represented by gain factor and phase factor. The simulated system is correctly identified when the values of the transfer gain factors are close to unity and the difference between the phase factors of a vibration component is very small.

Inspection of the results indicated that the helicopter system is adequately identified for forward flight. In hover, the values of the gain factors are reasonable, but a phase shift of about 62 degrees was noticed (lateral and vertical 1R vibration).

Table 5 Theory-to-Test Transfer Analysis

Flight Regime	1R vibration: gain and phase transfer factors						Tip relative displacements: transfer gains	
	longitudinal		lateral		vertical			
	gain	phase	gain	phase	gain	phase	track	lag
hover	0.887	269.0	0.965	132.6	1.117	64.8	0.963	1.079
70 kn	0.8922	261.6	0.983	66.9	1.187	5.5	1.037	1.156
90 kn	1.057	262.7	0.981	75.2	1.114	3.0	0.981	1.141
110 kn	0.9099	269.1	0.913	67.0	0.957	1.8	0.881	1.004

A theory-to-test transfer analysis has limited use where theoretical predictions do not necessarily correspond to the measured values directly, for example, in the prediction of bR vibration. In this case, accurate structural and operational data are required to correctly estimate the bR vibration.

8.4 Conclusions

Comparisons between the theoretical results from the MJAD mathematical computer models and measured vibration sensitivities indicated a good agreement between test and theory.

An effective HUM system should determine acceptable vibration levels associated with the helicopter operational conditions and judge whether the measured vibration is compatible with the expected levels or not in order to avoid false alarms triggered by high levels of vibration that can be theoretically attributed to excessive, temporary operational conditions. One of the methods to attain such effectiveness can be based on evaluating the differences between predicted normal vibration levels and actual measured vibration levels; such a strategy requires accurate and efficient (fast) mathematical models. Therefore, a theory-to-test transfer analysis has been presented to provide a matrix of calibration coefficients that can be used to improve the mathematical model capability. For example test data, the transfer analysis has been utilized to correct the theoretical predictions for 1R vibration results and blade track and lag.

9 MJAD Report MJAD/R/75/91, 1991 [5]

Title: The implementation of a pattern recognition technique to classify main rotor system faults

Authors: Hesham Azzam, Mike Andrew and Robert Callan

9.1 Introduction

Data modelling and machine learning techniques have a role to play in the efficient processing of HUMS data to extract useful health information. Stochastic and deterministic data modelling methods can be utilized to reduce large amounts of data to a small set of significant parameters and allow the state of the helicopter to be forecasted adequately. The methods provide a means of reducing the multidimensional space that describes the helicopter condition to a space of optimum principal dimensions. Viewed from the origin of the principal space, the data stability is usually improved and, therefore, the level of confidence in the forecasted state is significantly increased.

Supervised and unsupervised machine learning techniques can be used to classify the helicopter conditions and, if supervised, attach expert knowledge to the identified classes. In this report, an unsupervised pattern recognition technique was applied to theoretically generated vibration data in order to indicate the potential benefits of machine learning procedures. The applicability of supervised machine learning was also considered. Preliminary findings regarding the optimum methods, including data pre-processing, which can effectively aid the establishment of powerful diagnostic strategies are presented.

9.2 Machine Learning Techniques for Fault Diagnosis

Machine learning techniques have been developed to extract knowledge from a set of observations. The knowledge is a function of a machine state vector in N-dimensional space. The components of the machine state vector are the values of the observations and N is the number of the observations. In the helicopter HUMS case, the observations include vibration components from fuselage sensors, relative displacement measurements and performance parameters. The knowledge is needed to evaluate the vehicle health and guide maintenance actions.

9.2.1 Supervised Learning Techniques

The supervised methods attach defined knowledge to the learning system during training sessions. Then, the system can identify the state of the machine by matching the processed observations with the acquired knowledge.

Supervised diagnostics can be based on neural networks. Observations are presented to the network as the input, and the network produces an output (or several outputs) using 'weights' learned during training. To illustrate this, consider the vibration components induced by an impact load at a particular radial position of a blade. These components may be presented as input to a network. The weights are adjusted in training sessions such that the output of the network indicates the position at which the impact load is applied. The defined knowledge in this example is the position of the impact associated with the vibration pattern. Following the training period, the network will use the determined weights and identify the position of the impact from the vibration signature. The supervised methods can be used over a range corresponding to the range of the training set (interpolation), however extrapolation is usually unreliable.

The direct application of the supervised methods to HUMS data is likely to always be restricted because of the absence of a comprehensive data set. An appropriate training data set must contain a range of helicopter state vectors associated with each fault in order to describe the possible range of the fault severity. The data set must also contain a range of observations associated with multiple faults; two faults may be induced together in the same blade or in more than one blade. A huge training set is therefore required to cover all the possible main rotor system faults.

Defined faults and fault groups along with a theoretically generated data set could be utilised in order to establish a practical supervised system. In such a system, the large number of possible faults are reduced to a limited number of primary components. The system would be trained to identify the primary component faults from the theoretically generated data. In addition, the system could also be trained to compose a compound fault and predict the matrix of observations associated with the fault. Then, the weights of the system could be corrected by using a relatively small training set of measured data and theory-to-test transfer functions.

9.2.2 Unsupervised Learning Techniques

In this case no defined knowledge is required to train the system. The system classifies the state of the helicopter according to the values of the state vectors (the measured observations) unsupervised. The significance of the classification is purely geometric or statistical. For example, the state vectors of a severely damaged helicopter are expected to point at a location far away from the origin of the N-dimensional state space whilst the state vectors of an ideal helicopter would reside near to the origin. These two positions define the centres of two distinct clusters. The unsupervised system identifies the clusters (patterns) and it is up to the user to attach an engineering description to these patterns (if any).

The use of unsupervised techniques for fault diagnosis requires expert knowledge to signify the centre of each cluster (pattern) and its variance. The centre of a cluster may define a fault group and the variance may define the range of the possible fault severities. Unlike the pre-knowledge of the supervised system, the cluster knowledge is not used to train the system.

For this report, unsupervised learning was investigated, employing the MJAD Pattern Learning Algorithm TOolkit (PLATO). The objective of the investigation was to indicate the potential benefits of machine learning procedures and indicate preliminary ideas on the optimum methods of data pre-processing which can effectively aid the establishment of diagnostic strategies.

9.3 The Rotor System Faults

A helicopter resembling the S61 was considered and the MJAD mathematical model was used to establish the vibration and relative displacement signatures of 5 fault groups. The fault groups simulated a pitch link error (coded P), mass imbalance (coded M), a tab error (coded T), a damper fault (coded D) and a blade flap-wise crack (coded C). The severity of the fault within each group was varied such that ranges of low to severe vibrations were produced. A total of 27 records were produced for the various faults, with each record containing 34 observations. The observations consisted of the following:

i) Vibration components measured at a location near to the helicopter nose; sensor 1.

ii) Vibration components measured at a location aft of the main rotor shaft (about 8 metres from the tail; sensor 2).

iii) Relative tip track and lag displacements measured at 180 and 270 degrees of azimuth (i.e. at the nose and at the port side of the aircraft).

In order to generalize the findings of this study, two helicopters were considered. The fuselage of the first helicopter was relatively stiff and possessed a natural frequency close to 5R. The fuselage of the second helicopter possessed a natural frequency close to 2R. Thus, the vibration induced by a fault in the first vehicle differed from the vibration induced in the second.

The blade crack was induced at 0.25 radius. The severity of the crack was progressively increased to give changes of 1.5%, 3%, 4.1% and 6.5% in the third flapping frequency. The changes in the other natural frequencies were less than these values. The severity of the damper fault was increased by applying a stepped reduction of 22.5% from the original viscous resistance.

9.4 Pattern Recognition of the Rotor Faults

As indicated in the introduction, data modelling techniques can be used to reduce the multidimensional space that describes the helicopter state to a space of optimum principal dimensions. In the absence of data modelling, a manual selection process can be undertaken. This was the method adopted here, where data sub-sets were selected and processed by the PLATO system.

9.4.1 Classification of Vibration Components

Five groups were obtained by clustering the vertical vibration components (1R to 5R) measured at a location near to the helicopter nose (sensor 1). Two severe cases of pitch link error were isolated in group 1 and three (out of four) damper faults were isolated in group 3. The other groups contained more than one fault type. Nevertheless, the common feature between the different faults in the same group was found to be the level of vibration. For example, group 5 contained all faults which induced a low level of vibration. Three (out of four) crack faults were assigned to this group. An increase in the number of vibration components to 15 (1R to 5R vertical, lateral and longitudinal) had an insignificant effect on the recognition capability in this particular case, producing the result shown in Figure 3.

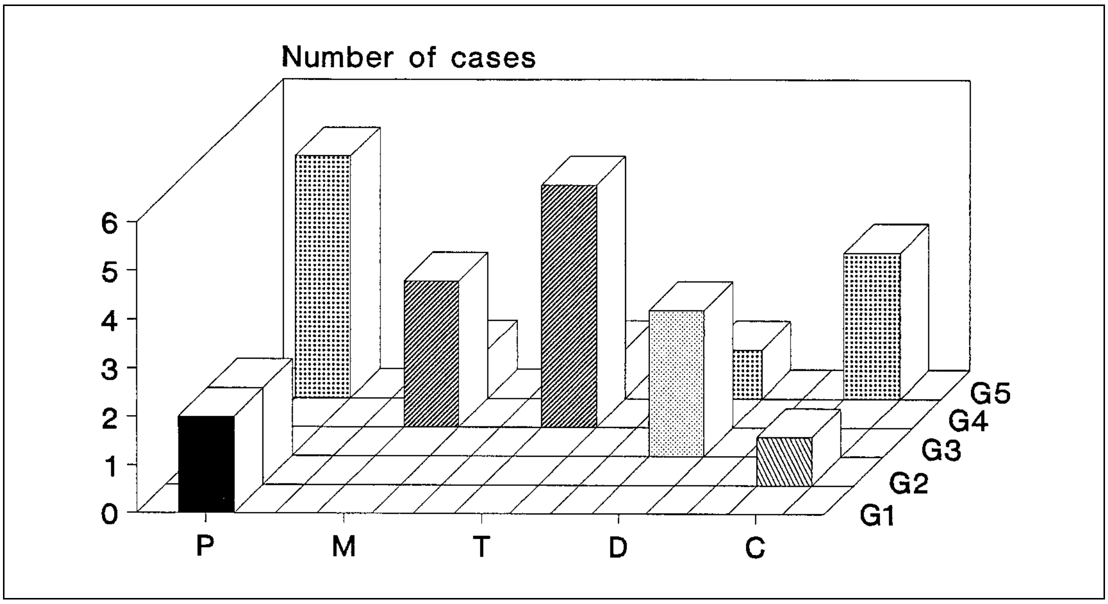

Figure 3 Fault Classification Based on 15 Vibration Components from One Sensor

A similar analysis was performed for the second helicopter. Again, classification based on a five vertical vibration components was found to identify the various faults on the basis of severity of induced vibration and not on the fault type. In this case the results indicated that an increase in the number of vibration components does improve the recognition capability; with 15 components one group now contained damper fault cases only and another group was dominated by tab error cases.

The analysis demonstrated that the classification of a small number of vibration components can be used to set alarm levels and identify severe faults, and the classification of a large number of vibration components can aid the recognition of particular faults. However, isolating each fault type in a separate group is very difficult and it was therefore concluded that the recognition of each fault type by classifying raw data is not practical.

9.4.2 Classification of Vibration Modes

As discussed in MJAD/R/60/91, the discrimination between various faults may be based on normalised vibration modes (calculated as ratios of vibration measurements at different sensors to a reference sensor). The normalised modes provide an invariant property for the fault whose vibration signature varies in a linear manner with the severity. Thus a set of vibration modes could describe a linear fault group. For nonlinear behaviour, a larger number of vibration mode sets is required (proportional to the order of the nonlinearity - i.e. two sets are required for a second order nonlinear fault group). Another invariant fault property is the ratio between the vibration modes, e.g. the ratio between the normalised 1R and 2R vibration modes. A vibration mode consists of a series of transfer ratios which partially describe the transfer functions of fuselage components and partially describe the various loads input to each element. The vibration modes can therefore be utilized to detect component structural faults which induce transfer function disparities, and also rotor system faults that change the load pattern.

The main rotor shaft lies between the locations of sensor 1 and sensor 2 and, hence, vibration modes obtained from measurements at these locations are expected to aid fault diagnosis. In the following analysis, the normalising measurement was chosen to be the 1R vertical vibration of sensor 1 because the 1R vibration is fault dependent. However, the nbR vibration components are helicopter and fault dependent. Therefore, the 5R vibration components of sensor 1 were normalised by the corresponding components of sensor 2.

Four groups were obtained by classifying two vibration mode components (the 1R longitudinal and 1R lateral components). Remarkable recognition was achieved; the damper fault cases, the cracked blade cases and the mass imbalance cases were now all isolated in separate groups. The fourth group contained tab error and pitch link error cases. Using a clustering option to pre-specify the number of data groups, a five-group classification of the same two vibration mode components characterised the damper fault cases into two groups on the basis of the damage severity. Full recognition of each fault type was achieved by classifying three mode components (1R vertical, lateral and longitudinal) (Figure 4).

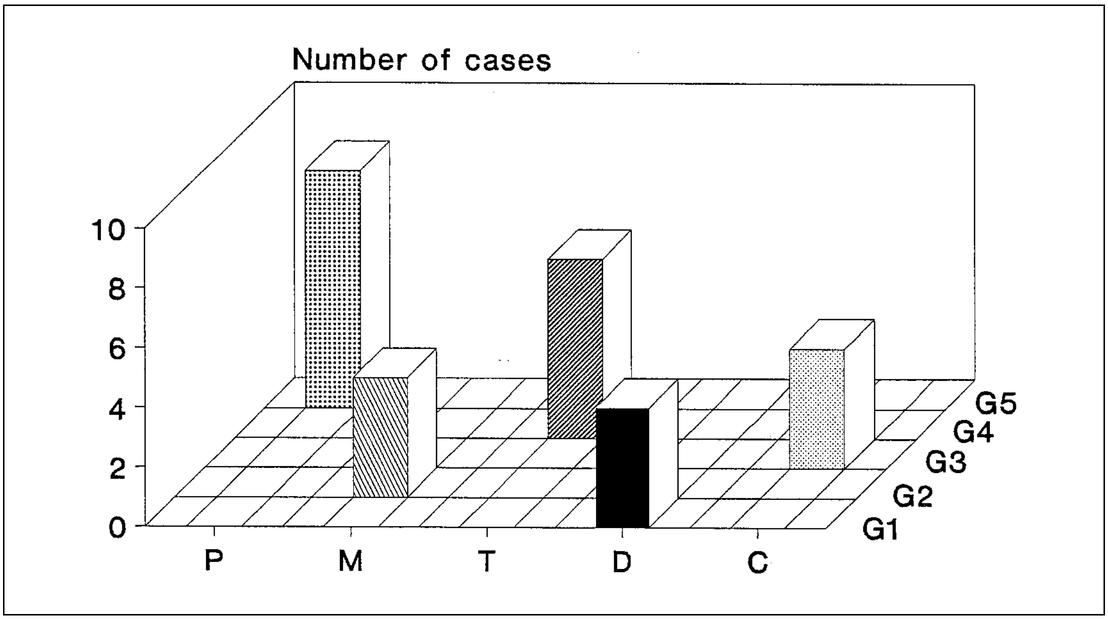

Figure 4 Fault Classification Based on Three Normalised Vibration Components from Two Sensors

Fifteen vibration mode components were analysed with 4, 6 and 7 groups. With four groups, the mass, tab and pitch link errors cases occupied the same group. Six group classification allowed the mass imbalance cases to be isolated. Seven group classification allowed the full recognition of fault types. The marked effect of a large number of dimensions was to characterise the non-linear fault group of cracked blade. The above findings were also shown to apply to the second helicopter.

The analysis demonstrated that classification of a small number of vibration mode components can provide a clear recognition of various main rotor system faults. Classification of a large number of vibration mode components characterise the non-linear fault groups. Expert knowledge is required to attach a fault name or description to various group centres and assign a range of fault severities to the variance of each group. This knowledge can be obtained from the mathematical model along with the available measured data.

9.5 **Conclusions**

Mathematical computer models were used to evaluate the vibration and displacement disparities induced at two fuselage-mounted sensor locations by five main rotor system faults. A pattern learning algorithm was used to classify these disparities, producing the following findings:

i) Classification of a small number of vibration components can be used to set alarm levels and identify severe main rotor system faults.

ii) Classification of a large number of vibration components can aid the recognition of particular faults.

iii) The recognition of each fault type by classifying raw data (vibration components and relative displacements) is not practical.

iv) Classification of a small number of vibration mode components can provide a clear recognition of various main rotor system faults.

v) Classification of a large number of vibration mode components can characterise the non-linear fault groups.

10 MJAD Report MJAD/R/86/91, 1991 [6]

Title: A non-linear supervised diagnostic technique for the recognition of main rotor system primary and compound faults

Author: Hesham Azzam

10.1 Introduction

The purpose of this report was to investigate early fault detection strategies for main rotor system primary and compound faults based on appropriate pre-processing and a non-linear supervised diagnostic technique.

10.1.1 Data Pre-Processing

Diagnosis of main rotor system faults is often based on the analysis of vibration components induced by faults. Unfortunately, the possible significant influence of operational effects on these components may lead to a false interpretation. Corrupted data records can also lead to a misleading diagnosis. Another source of fault discrimination errors is associated with limitations of pattern recognition techniques. It was apparent that most of the diagnostic difficulties arise from the direct use of raw data in the discrimination process. It was therefore postulated that data pre-processing could overcome the diagnostic difficulties by performing the following tasks:

i) **Data integrity checking:** The objective of this task is to check data validity and separate corrupted data records. The malfunction of a system sensor can be identified by analysing sensor signals and comparing the results with a reasonableness threshold and a matrix of cross-validation criteria between system sensors.

ii) **Operational effects modelling:** The objective of this task is the implementation of models to mitigate operational effects and map the HUMS database into another database of reduced fields (the fields associated with the operational conditions are removed). The vibration components of the new database reflect the mechanical state of the vehicle and not the operational state.

iii) **Pre-process mapping:** The pre-process mapping is a series of transformations applied to the vector components that describe the machine state (i.e. vibration components). The transformation models may be tailored to remove the difficulties associated with a particular pattern recognition technique. For example, in MJAD/R/75/91 initial attempts at separating mechanical fault classes by grouping vibration data highlighted the difficulty with data clustering techniques. The major problem was that a fault from a given class could migrate from one data group to another because of its severity. By pre-process mapping of vibration using a simple linear severity model, fault types were successfully separated.

10.1.2 Machine Learning Techniques

Machine learning techniques are either supervised or unsupervised. Supervised methods attach defined knowledge to the learning system during training sessions. The system can then identify the state of the machine by matching the processed observations with the acquired knowledge. Unsupervised methods classify the state of the helicopter according to the values of the state vectors (the measured observations) unsupervised.

A supervised technique (e.g. a neural network) can only be used over a range corresponding to the range of the training set, therefore an extremely large training data set would be required. However, if appropriate data pre-processing techniques can help to overcome this difficulty, neural networks offer a potentially promising tool.

10.2 Symptoms of Rotor System Faults

A helicopter resembling the S61 was considered and the math-model described in MJAD/R/28/89 was used to establish the vibration and relative displacement symptoms of six fault groups. The fault groups simulated a pitch link error (coded P), mass imbalance (coded M), a tab error (coded T), a damper fault (coded D), a blade flap-wise crack (coded C) and a jammed flapping hinge (coded J). The severity of the fault within each group was varied such that ranges of low to severe vibrations were produced. A total of 53 records were generated for various faults, with each record containing 94 observations. The observations consisted of the following:

i) Vibration components measured at the following locations: (1) near to the helicopter nose; sensor 1, (2) one metre aft of sensor 1; sensor 2, and (3) aft of the rotor shaft; sensor 3 (about 8 metres from the tail).

ii) Relative tip track and lag displacements measured at 180 and 270 degrees of azimuth

The vibration components 1R to 5R were considered and each component was described by an amplitude value and a phase angle. The mass imbalance was introduced at 0.15 radius. The tab symptoms were obtained by considering a tab of 0.04 metres in length located at 0.7 radius. The blade crack was induced at 0.5 radius. The severity of the crack was progressively increased such that the associated changes in the second flapping frequency corresponded to values in the range of 0.94 % to 22%. The damage severity of the jammed flapping hinge was progressively increased, with changes in the first flap-wise natural frequency ranging from 0.45% to 6.6%. The severity of the damper fault was increased by considering a stepped reduction from the original viscous resistance. The fault severity was described by the percentage reduction of damping.

The 53 estimated observation records were used to train the non-linear supervised system. The mathematical model was also used to generate symptoms for a number of primary and compound faults in order to examine the diagnostic capability of the system.

10.3 A Non-linear Supervised Technique

Neural networks can provide implicit non-linear models that describe the relationships between faults and induced vibration. Unlike neural networks, in this report a mathematical model-based non-linear supervised technique was used, in which non-linear models (relationships) are explicitly expressed (i.e. as a set of fault models); an example of a non-linear model is as follows:

> The real component of the vertical vibration induced by a crack of severity x at radial location r at 70kts = (ax+bx2+cx3) cos[p(0.5R-r)/R].

A least square error minimisation method is then used to determine the weights of the models; therefore, extrapolation capabilities can be achieved if the non-linear models are adequately representative; the reported method can be regarded as an advanced non-linear extension of the well known Rotor Track and Balance (RTB) methods.

In the training session, model weights for each observation were calculated using an error minimisation algorithm. The best degree of non-linearity was determined from values of correlation coefficients, error functions or ratios between standard deviations and averages of mapped observations. The system started from a simple linear model (if not instructed to do otherwise), and progressively increased the degree of non-linearity until specified model accuracy was reached. A total of 94 non-linear models (one for each observation) were automatically generated. Nevertheless, models supplied by an expert user may improve the diagnosis and reduce the computational capacity required.

The system produced a "N-expected fault model" if the number of the expected faults in the training set was N. However, the user could request another expected fault model of order less than N. The necessary number of observations required to diagnose a fault and determine its severity using a particular expected fault model was equal to the order of this model. A number of observations less than the order of the expected fault model may suffice for the recognition of fault types only (i.e. not severities). For example, in MJAD/R/75/91, three pre-processed observations were used to identify five fault groups. The non-linear models were tagged with significance numbers by scanning the associated training set (identifying the most significant effects of faults from the models); for example, a crack-induced vibration could have less significance in hover than in forward flight. The most significant set of non-linear models (the fundamental set) was selected for fundamental diagnostics. The other sets of models were used to cross-examine the diagnosis and define confidence limits. Ideally, a data set should be mapped into a space where effects of operational conditions such as AUW, temperature, and pressure are removed. The observations obtained at different speeds (e.g. hover and high forward speed) could be incorporated into one record. The optimum diagnosis was then obtained by a least square minimisation approach applied over a wide range of observations.

The diagnosis of a fault is achieved by presenting the associated set of observations to the trained system. Quite often, the measured vibration is composed of fault and machine dependent components. The machine dependent vibration can be felt even if the machine is free of faults. The non-linear models identify this vibration. An important pre-processing step is to remove the machine dependent vibration in order to facilitate the diagnosis of a compound fault using the symptoms of its primary components.

10.4 Diagnosis of Main Rotor Tuning Faults

Mass imbalance, pitch link and tab errors were considered and an associated three-expected fault model was implemented. Symptoms of primary faults were used to train the system. Although the tab fault could be regarded as a compound fault because it influences the lift, drag and pitching moment, it is regarded as a primary fault because it is related to the deflection of a single lifting surface.

10.4.1 Primary Faults

The mass fault was found to be a linear fault. The pitch link and tab errors were found to be close to linear (i.e. the degree of non-linearity is relatively low). Both linear and non-linear models were investigated. The two models were found to predict the mass imbalance accurately even if the mass value was well outside of the training range (i.e. successful extrapolation to out-of-range faults).

Regarding a pitch link fault, both models accurately identified an out-of-range error, but incorrectly diagnosed small residual values of mass and tab faults. The residual values predicted by the non-linear model (1.0 grams and 0.054 degrees) were less than those predicted by the linear model (102 grams and .145 degrees). Increasing the degree of non-linearity was found to reduce the residual errors.

Considering an out-of-range tab fault, the diagnostic accuracy of the non-linear and linear models was better than 0.1% and 5.0% respectively. The residual mass and tab errors given by the non-linear model were 6 grams and 0.002 degrees. The residual errors of the linear model were 100 grams and 0.121 degrees.

The analysis demonstrated that the mass fault is a linear fault, and the pitch link and tab faults are close-to-linear faults. Both the linear and non-linear models fully recognize primary tuning faults (type and severity) even if a fault severity lies out of the training range. The non-linear model provides excellent diagnostics and accurate extrapolation capability.

10.4.2 Compound Faults

The vibration induced by a compound mass imbalance and pitch link fault was introduced to the linear and non-linear models. The diagnostic accuracy of the two models was better than 0.5%. With a compound mass imbalance, pitch link and tab fault, for the linear model the diagnostic accuracies of the mass, pitch link, and tab components were better than 2.1%, 2.73% and 19% respectively. The corresponding diagnostic accuracies of the non-linear model were better than 1.5%, 1.26% and 8.3% respectively.

The analysis showed that both the linear and non-linear models fully recognize the primary contents of compound tuning faults (type and severity), however the non-linear model is more accurate than the linear model. A tuning compound fault is composed of its primary components in a weak non-linear manner (i.e. almost linear) and, hence, the rules of linear composition can be applied to decompose the fault with a reasonable degree of accuracy. In other words, the symptoms of a compound tuning fault can be considered to be the sum (or the vector sum) of the symptom values of its primary components.

10.4.3 A Low Order Expected-Fault Model

Modelling was performed to demonstrate that maintenance actions that are made with the view that the vibration is induced by a number of expected faults 'Ne' less than the actual number of faults will never totally remedy the fault symptoms. These actions can remove a number of independent vibration components equal to Ne-1. If a vibration minimisation approach is used (for example when performing RTB), a specified number of vibration components can be reduced.

The optimum order of the expected-fault model (i.e. matching the actual number of faults present so that there are no masked faults) can be determined by an appropriate implementation of unsupervised learning techniques. The choice of the model order is important not only to reduce the level of vibration (e.g. when performing RTB) but also to avoid masking catastrophic faults, which could occur if a low order expected-fault model is used.

10.5 Main Rotor Compound Faults

The capability of the non-linear supervised technique to predict any primary fault including a cracked blade, a jammed hinge or inoperative damper was found to be excellent (perfect interpolation and accurate extrapolation). Regarding a compound fault, the diagnostic accuracy depended on how the fault was composed from its primary components.

To demonstrate the recognition capability of the non-linear system, Figure 5 shows the diagnosis of a compound fault composed of a mass imbalance, a pitch link error, a tab error and a blade crack fault (MPTC compound fault) using a six-expected fault model. Figure 6 shows that a five component compound fault (MPTDJ = mass, pitch link, tab, damper and jammed flapping hinge faults) is also reasonably estimated by a six-expected fault model.

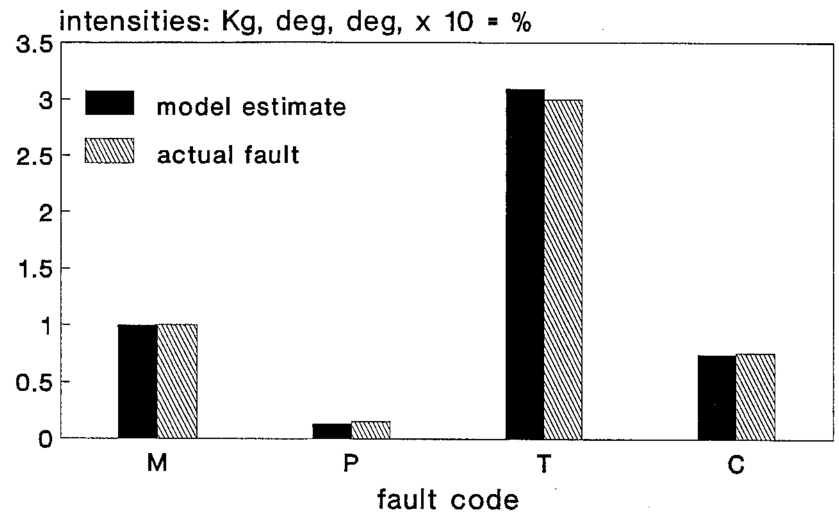

Figure 5 MTPC Compound Fault Diagnosed by a Six-expected Fault Model

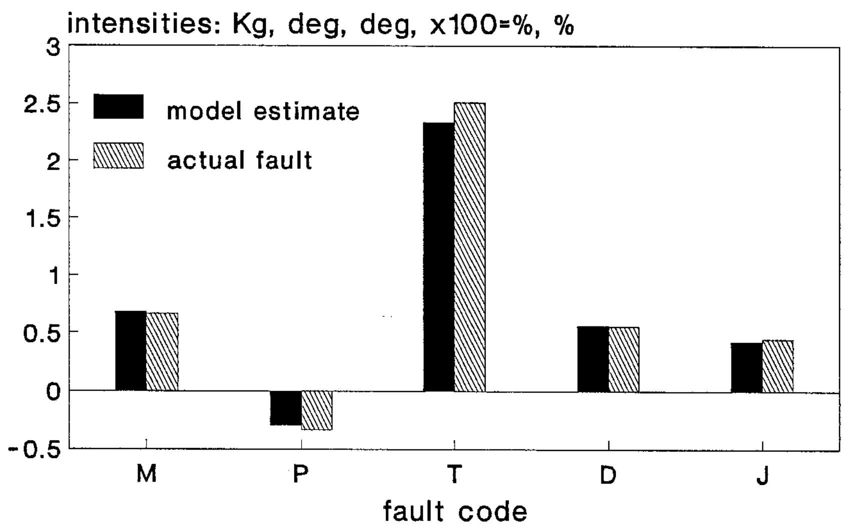

Figure 6 MTPDJ Compound Fault Diagnosed by a Six-expected Fault Model

The diagnostic results are greatly influenced by the chosen observations for diagnosis (i.e. choosing observations that are most effected by the faults). A fundamental set of observations has been used to diagnose the compound faults. The fundamental sets of observations were selected by inspecting the associated non-linear models (94 models). Then, the most significant observations were chosen such that solution singularities, ill condition formulations and low signal to noise were avoided.

The analysis demonstrated that an important pre-processing step is to scan the training data set and determine the fundamental set of observations. This step can be also be implemented to define the optimum number and location of sensors that provide a high identification resolution of a specified number of faults. The non-linear supervised diagnostic technique, which is based on linear decomposition, identifies the components of compound faults with reasonable degree of engineering accuracy and, hence, a substantial reduction in the level of vibration can be achieved by introducing maintenance corrections in sympathy with the obtained results. A second stage diagnosis is required to remove the vibration residues. Diagnosis based on non-linear decomposition is an alternative method that can improve the recognition accuracy and may eliminate the need for the second stage diagnosis.

10.6 Non-linear Decomposition

In order to train a neural network, a huge dataset would be required to cover the possible large number of primary and compound faults. However, it has been demonstrated that a non-linear supervised system can be successfully trained using a small dataset that contains primary faults only. The system is used to accurately diagnose primary faults, and implements linear decomposition rules to successfully identify compound faults. The accuracy of identification can be refined further by including a limited number of compound faults in the training set. These faults allow the rules of non-linear decomposition to be established. A full first order non-linear decomposition can be achieved by including in the training set a number of compound faults equal to the number of primary faults. A second order non-linear decomposition requires twice the number. In practice, a reasonable accuracy can be achieved by considering a partial first order non-linear decomposition. In this case, the non-linear coupling is considered for a number of faults less than the number of the primary faults.

10.7 Conclusions

A non-linear supervised diagnostic technique was developed and trained using mathematically generated vibration induced by six main rotor system primary faults. The non-linear system was used to diagnose primary and compound faults.

The capability of the non-linear supervised system to predict primary faults including a cracked blade, a jammed hinge or an inoperative damper was found to be excellent even if the value of the fault severity was outside of the training range. The non-linear supervised system fully recognized the primary contents of compound tuning faults (type and severity).

Diagnosis based on a low order expected-fault model can mask catastrophic faults. The correct order of the expected-fault model could be estimated by unsupervised learning techniques.

For a group of faults and observations, there is a fundamental set of observations, which provides the best diagnostic accuracy. An important pre-processing step is to scan the training data set and determine the fundamental set of observations. This step can be also implemented to define the optimum number and location of sensors that provide a high identification resolution.

A non-linear supervised diagnostic technique based on linear decomposition identifies the components of compound faults (linearly or non-linearly composed) with reasonable degree of engineering accuracy and, hence, substantial reduction in the level of vibration can be achieved by introducing maintenance corrections based on the obtained results. For non-linearly composed faults, a second stage diagnosis is required to remove the vibration residues. A technique based on non-linear decomposition improves the fault recognition accuracy and may eliminate the need for a second stage diagnosis.

11 MJAD Report MJAD/R/90/91, 1991 [7]

Title: The use of infield data and mathematical dynamic models to evaluate diagnostic strategies for main rotor system faults

Author: Hesham Azzam

11.1 Introduction

This report summarises several elements of the work described in the previous documents. HUM Systems provide a wide range of sensor measurements on a flight by flight basis, and offer considerable potential for the construction of fault diagnostic strategies by efficiently extracting useful information from the large quantity of resulting data. Mathematical dynamic models, pre-processing methods and machine learning techniques can be used to realize these strategies. Therefore, representative models were developed and implemented along with in-service data in order to evaluate appropriate diagnostic strategies for helicopter main rotor system faults.

11.2 Mathematical Dynamic Models

The role of math-dynamic models is widely recognized to substantially support the various prediction procedures and facilitate the extraction of representative information from system signals. The following activities were carried out to provide math-dynamic models for a rotor health monitoring system.

11.2.1 Simulation of Main Rotor System Faults

This task evaluated induced effects of rotor system faults (in the rotating frame) to define fault discrimination strategies. For example, previous work including the analysis of 120 fault cases indicated the following three conceivable fault discrimination strategies:

i) Identifying the vibration and relative blade displacement disparities induced by faults.

ii) Monitoring a matrix of blade natural frequency perturbations to measure the possible defect severity and determine the defect location.

iii) Estimating the disparities of the system parameters induced by faults, ageing, and/or environmental disturbances.

Sufficient promise has been indicated in the analysis to suggest that there is no need to place sensors in the rotating frame.

11.2.2 Simulation of the Fuselage Response

This task facilitated the practical implementation of the first discrimination strategy. By evaluating fuselage vibration at various sensor locations, it is feasible to define the minimum number of sensors and determine the optimum sensor locations for effective monitoring (MJAD/R/60/91). Where raw vibration features do not yield a successful diagnosis, the use of appropriate mapped features in the diagnostic process is desirable. In MJAD/R/60/91, fuselage vibration was normalised (mapped) to define vibration modes. Such a mapping reduced the sensitivity of vibration features needed for fault isolation to fault severity and attenuated non-stationary characteristics of measurements caused by ageing and noise.

Vibration modes associated with a pitch link error, a tab error and a blade crack were evaluated by normalising the vibration to unity at the helicopter nose. For the example faults, each possessed a distinct pattern in the combined behaviour of the 1R and 2R vibration modes. Therefore, the ratio between the normalisation factors of the 2R and 1R modes provided a diagnostic feature (an invariant property for linear faults). Thus, the knowledge learned from a fuselage response model can be implemented to improve fault detection techniques using fuselage sensors.

11.2.3 Simulation of Helicopter Manoeuvres

The second discrimination strategy requires a periodic modal examination of the rotor blades and hub assembly. This implies mounting components on a test stand and applying appropriate excitation loads. For example, the relationship between crack radial locations and blade modal frequency changes shown in Figure 7 (calculated using a fracture hinge model and Dirac Delta model) indicates that cracks can be identified by monitoring a matrix of blade natural frequency perturbations.

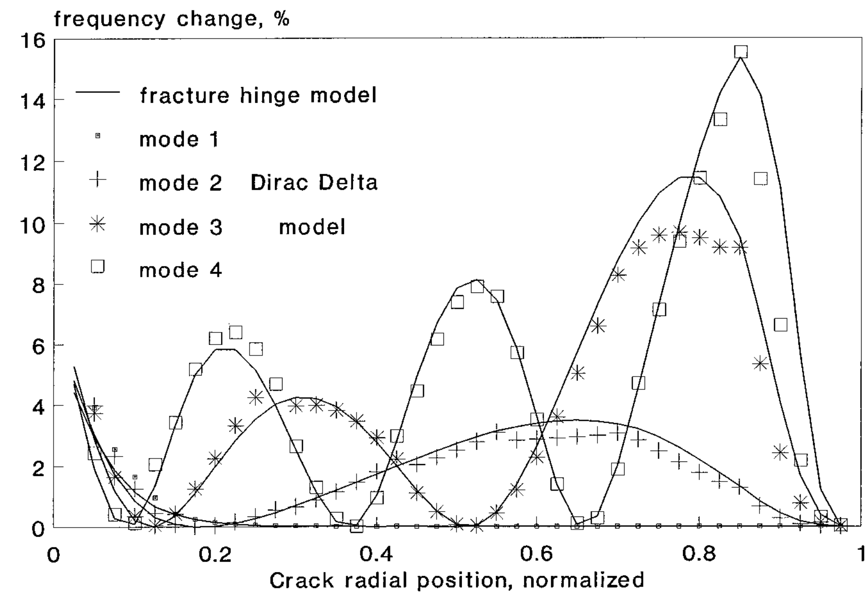

Figure 7 Effects of the Location of a Crack on the Natural Frequencies of a Hingeless Blade

The second discrimination strategy can also be realized during appropriate helicopter operating conditions where the natural vibration modes influenced by blade defects are sufficiently excited. In MJAD/R/63/91, a math-dynamic model for helicopter manoeuvres was used to carry out an investigation to detect cracking/delamination in main rotor blades during rotor start-up and shutdown operations. Discriminatory patterns in the shutdown data were identified after an appropriate data processing technique was established. The model has therefore been used to distinguish an effective operation and choose an appropriate data analysis technique, which will magnify fault disparities. It could be utilised in future to define acceptable pilot control actions to be applied to excite the natural vibration modes that may be modified by rotor system faults.

11.2.4 Theory-to Test Transfer Analysis

Whilst it has been demonstrated that helicopter math-dynamic models can be of substantial benefit for rotor health monitoring, the prediction capability of these models is often restricted because of the vehicle complexity and the associated formidable dynamic, aerodynamic and structural problems. It is therefore required to establish a prediction strategy that can overcome any possible shortcoming of available math-dynamic models.

Theory-to-test transfer analysis (MJAD/R/69/91) is a technique that provides a matrix of calibration coefficients, which can be used to correct theoretical predictions. For example, a transfer matrix was produced for the 1R vibration induced by a pitch link error.

A narrow range of measured data can be used to identify system parameters through the use of appropriate theory-to-test transfer analysis. In MJAD/R/69/91, it was proposed that the parameters required to establish rigid body and elastic motions can be defined through the use of model equations and sensor measurements. Then, the model can forecast vibration signatures associated with a wider range of faults and, hence, support supervised diagnostic techniques that require such a range of training data.

11.2.5 **In-Service Signatures of Faults**

i) **Mass imbalance:** Regarding the 1R lateral vibration induced by main rotor blade mass imbalance, it has been shown that the theory and test agrees reasonably well.

ii) **Pitch link error:** Considering the measured and predicted vibration induced by a pitch link error, results show that the predictive accuracy of the 1R vibration components in hover and at forward flight is better than 85%. The predictive accuracy of the relative track and lag is better than 87%.

iii) **Lag damper fault:** Inspection of measured characteristics of a Sea King helicopter lag damper indicated that a damper may operate initially as a lightly damped system until the lead-lag speed reaches a critical value at which a full viscous damping becomes effective, then, a "blow off" condition may occur whereupon the damping characteristic is more friction in nature. If a blade damper looses its damping properties, it may be expected that the blade will lag the other blades since the resistance of the faulty blade to the aerodynamic drag forces is reduced. In contrast, data acquired on the Sea King helicopter with defective lag damper at a rotor azimuth between 180 and 200 degrees produced a different trend. Nevertheless, the mathematical dynamic model predicted the measured trend correctly. The results referred to as theory 1 were obtained by considering the measured viscous and Coulomb friction characteristics of the dampers. Theory 1 was found to describe the unexpected trend in a qualitative manner. When the possible effect of the friction of the damper and the attachment hinges on the lag natural frequencies was taken into account (effecting a 0.31% rise in the fundamental lag frequency of the blade with the faulty damper), the model also produced the desired quantitative results (theory 2). The actual lag displacement of the blade with a defective damper was seen to be a function of blade azimuth, helicopter speed and all up weight (AUW).

iv) **Tail rotor mass imbalance:** Evaluation of vibration components measured at locations near the tail rotor requires thorough information about the structure of the tail rotor assembly and pylon. At these locations the elastic response is dominant and the contribution of the rigid body motion is very small. The measured ratio between the 1R lateral and 1R vertical vibration could be reasonably predicted by appropriate choice of structural property distributions.

v) **Tail rotor brinelled flap bearing:** The vibration induced by the brinelling of flap bearings in the AS332L tail rotor was measured during a ground run. The wind state during the ground run was unknown. The theoretical trend of the 1R-5R vibration was similar to the measured trend. As mentioned above, accurate evaluation of vibration signatures near the tail rotor requires detailed structural information as well as a model that considers the coupling between the lateral and longitudinal motions.

vi) **Main rotor blade crack:** A controlled test undertaken on an AH-1S Cobra composite main rotor blade detected a 30% increase in calibrated blade bending loads during testing, which equated to a reduction in root stiffness of 30%. An inspection showed some delamination of the blade. Analysis of the strain gauge data revealed a progressive increase in strain levels in both flap-wise and torsional directions. The test indicated that blade cracking/delamination causes a stiffness reduction. By simulating this reduction, vibration and relative displacement disparities induced by a blade crack have been evaluated.

11.3 Data Pre-Processing

It was postulated that data pre-processing can overcome difficulties arising from the direct use of raw data in fault discrimination processes through data integrity checking, operational effects modelling, and pre-process mapping.

11.4 Data Integrity Checking

The objective of this task is to check data validity and separate corrupted data records. The malfunction of a system sensor can be identified by analysing signals of sensors and comparing the results with a reasonableness threshold and a matrix of cross validation criteria between system sensors.

11.5 Operational Effects Modelling

Models are implemented to mitigate operational effects so that vibration components reflect the mechanical state of the vehicle and not the operational state. The inputs to the models are required to be effective parameters that reasonably correlate with airframe vibration.

11.5.1 Effective Parameters

In order to attenuate the operational influences on vibration, effective helicopter parameters are introduced (e.g. effective weight, power, r.p.m. and speed). The effective parameters are obtained from non-dimensional parameters through the use of appropriate reference values of temperature (15 degrees centigrade), air density (1.225 kg/m^3), air pressure (1 bar) and altitude (sea level). The effective parameters are more closely linked to airframe vibration. To illustrate this, in-service data from three S61 helicopters were analysed. The degree to which the 5R vibration is related to ordinary and effective parameters was indicated through the evaluation of correlation coefficients. It was shown that the effective parameters correlate to vibration more than the ordinary parameters.

11.5.2 Models to Provide Corrected Vibration Amplitudes

In many cases, sensor measurement interpretation is based on periodically checking amplitudes against a predefined threshold. A shortcoming with the traditional threshold criteria is that an aircraft may move in and out of a serviceability state purely as a result of prevailing operating conditions and not because of any mechanical deterioration. It is therefore beneficial to mitigate effects of the perturbations in weight, speed, rotor r.p.m and settings of control stick and pedals. (Perturbations in the latter are often required to compensate for other perturbations including changes in c.g position).

Using the effective parameters described previously, a theoretical approach was used to provide a model to relate hub vibratory loads to weight, speed and r.p.m. This can be used to forecast vibration amplitudes at logged operational conditions. Differences between measured and forecasted amplitudes should then primarily depend on the helicopter health state and weakly depend on the modelled operational conditions (e.g. weight, speed and r.p.m.). Hence, fault diagnosis can be based on these differences. The outputs from the model are corrected vibration amplitudes. The corrected amplitude consists of two parts; the difference between the measured and forecasted amplitudes, which depends on the health state of the machine, and a constant part that describes a typical operating state. A corrected amplitude therefore contains diagnostic information and is also a representative engineering indicator of vibration levels.

The modelling technique described above was implemented on in-service data from two S61 helicopters. A comparison between corrected and raw 5R amplitudes showed that the model attenuated effects of weight, r.p.m and speed perturbations (Figure 8). The erratic fluctuation of amplitudes between flights is reduced. There are two peaks in the figure that have not been attenuated. These can be attributed to either the onset of a fault or an operational condition that is not included in the model (e.g. the position of the control stick).

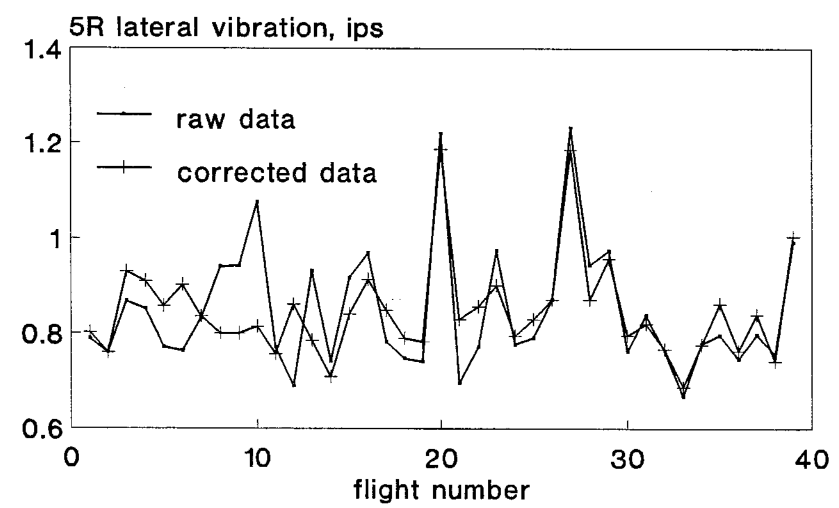

Figure 8 S61 Measured at Corrected Airframe Vibration in Cruise (RTB Sensor Location)

For a full evaluation of the model coefficients, more in-service data would be required from serviceable helicopters that cover the complete range of operational conditions.

11.5.3 **Pre-Process Mapping**

Pre-process mapping is a series of transformations applied to vibration components to aid fault discrimination by removing difficulties that can be experienced with the raw components. Theoretically generated data has been mapped through the use of vibration modes.

i) Vibration Modes

A vibration mode partially describes dynamic characteristics of fuselage components between sensors and partially describes component loads. Thus, vibration modes can be utilized to detect rotor system faults that change the load pattern as demonstrated in MJAD/R/60/91. In the following section the term normalised vibration refers to a local value of a vibration mode.

ii) Vibration Modes of In-Service Data

Using S61 in-service data, a comparison was performed between raw and normalised vibration amplitudes at four fuselage locations. The normalisation was based on measurements at the Rotor Track and Balance (RTB) location. The normalised vibration is multiplied by the mean value of the RTB amplitudes in order to regain appropriate engineering values. The normalisation attenuated the erratic (fluctuating) behaviour of vibration at the station 290 and engine front support (EFS) locations. The normalised vibration at station 493, however, exhibited a higher degree of variability in comparison with raw amplitudes. For a sensor at the tail gearbox (TGB) location, the normalisation hardly influenced the erratic behaviour.

The results indicated that the normalisation could produce favourable or adverse effects. In other words, operational effects and noise may be attenuated or magnified depending on relative sensor positions and correlation between sensors. It should be noted that operational conditions modelling has either a favourable effect or no effect and has no adverse effects. However, the modelling mitigates operational effects only and does not mitigate noise.

The final analysis investigated the effect of normalisation in the presence of a fault. Data was obtained from an S61 helicopter before and after two maintenance actions to rectify high vibration levels. In the first action, an error (suspected to be an adjustment in the wrong direction) was unwittingly introduced, causing higher vibration levels. After four flights, the maintenance procedure was successfully repeated and the error corrected. The 1R vertical vibration amplitudes of a group of sensors provided discrimination between the cases (represented by three groups of data). The vibration amplitudes were then normalised using different sensors. For most of the normalising sensors (8 out of 10 sensors), the normalised amplitudes identified the distinct signature after the correct maintenance, but did not provide discrimination for the first two datasets, which was attributed to the fact that vibration modes are less sensitive to defect severities. It was noted that that the diagnostic resolution would depend on the choice of the normalising sensor; dividing by the amplitude of a noisy sensor would reduce the quality of results.

11.6 Machine Learning Techniques

Both supervised and unsupervised machine learning techniques have been investigated.

11.6.1 Unsupervised Learning Technique

In MJAD/R/75/91, the capability of an unsupervised data clustering and a pre-processing technique was investigated. Initial attempts at separating fault classes by grouping vibration data using a clustering technique highlighted the problem that a fault from a given class could migrate from one data group to another because of its severity. However, by pre-process mapping of vibration to fuselage vibration modes, fault types were separated. It was concluded that unsupervised learning is important because:

i) It provides unbiased classification of patterns (associated with faults). The feature space is divided into regions, each region relates to a fault, an operational condition or both. Knowledge about these regions can be used to set an appropriate expected fault model or establish training set requirements for supervised methods (see the following section).

ii) A fault may display a disconnected pattern and occupy two separated regions. Supervised identification of such a fault is laborious, however a series of supervised diagnoses applied on regions defined by unsupervised classification could rectify such a problem.

11.6.2 Supervised Learning Techniques

As indicated in MJAD/R/86/91, a supervised neural network can be used to cover a range corresponding to the range of the training set (interpolation), however, extrapolation can not happen in any predictable way. Therefore an extremely large training set is required, containing a range of helicopter state vectors associated with each fault in order to describe the possible range of fault severity, and also a range of observations associated with multiple faults. Nevertheless, neural networks are potentially promising tools if it is possible to overcome the problem of initially limited a-priori knowledge about faults. This is can be achieved through further implementation of mathematical dynamic models, which can be used to generate fault vibration signatures. Training sessions can be initiated by theoretically generated features and then updated by measured features as they become available. This step requires further validation of the mathematical dynamic models.

A second step is to develop a supervised method that can cope with limited a-priori knowledge about faults. In MJAD/R/86/91, a non-linear supervised diagnostic technique was proposed and it was found that the technique reasonably predicts primary and compound faults even if the value of fault severity lies out of the training range. In other words, the technique can cope with a limited training set. The non-linear supervised method can be used to extrapolate measured features and hence extend training sets for neural network applications.

One issue to be addressed is that supervised techniques require prior knowledge about the number of expected faults. Fault diagnosis should be based on an N-expected fault model, where the number of the expected faults in the training set is N. Maintenance actions made with the view that the vibration is induced by a number of expected faults less than the actual number will never remedy all the fault symptoms and can mask serious faults. Unsupervised methods, which divide the feature space into a number of regions, can be used to check whether the correct number of expected faults is chosen. By attaching engineering knowledge to each region, it is possible to determine a number of expected faults. It is also feasible to observe the formation of new regions and update the number of expected patterns accordingly.

11.7 Conclusions

Representative mathematical dynamic models, pre-processing methods and machine learning techniques have been developed and utilized along with in-service data to evaluate diagnostic strategies for helicopter main rotor system faults.

By implementing a fuselage response model, the discrimination effectiveness of fuselage sensors, appropriate mapping procedures, and feature selection methods was scrutinised. A fuselage mathematical dynamic model was developed and implemented to evaluate vibration induced by main rotor system faults along the helicopter fuselage, indicating positions and measurement components that significantly reflect fault signatures. The fuselage vibration was normalised (mapped) to define vibration modes. Such a mapping reduces the sensitivity of vibration to fault severity and attenuates the non-stationary characteristics of measurements caused by ageing and noise. Vibration modes can provide important diagnostic features. Further implementation and validation of the mathematical dynamic models could be targeted at investigating feature selection and mapping procedures that can magnify fault signatures.

Signatures of a suite of rotor system faults were scrutinized through the use of mathematical dynamic models and in-service data. Generally, the agreement between the theory and test was found to be acceptable.

By developing a model for helicopter manoeuvres and implementing an appropriate data analysis technique, vibration disparities caused by blade cracking/delamination during rotor shutdown were detected. The model could be utilised to define acceptable pilot control actions in order to excite the natural vibration modes, which might be modified by rotor system faults.

By developing a theory-to-test transfer analysis, the theoretical predicative capability of fault signatures was improved. In-service data was used to provide a matrix of calibration coefficients, which can be used to correct theoretical predictions. A narrow range of measured data could be used to identify system parameters. More specifically, the parameters required to establish rigid body and elastic motions can be defined through the use of model equations and sensor measurements.

Models were developed to mitigate operational effects on vibration. By applying the models to in-service data, vibration peaks caused by operational conditions were reduced. By pre-process mapping (normalisation) of in-service vibration, operational and ageing effects or noise can be attenuated, meanwhile fault features can potentially be made clearer. The effect of fault severity on vibration was found to be reduced, supporting conclusions drawn earlier from theoretically generated data. Data pre-processing methods require further investigation. The methods could be improved, for example, by incorporating the control stick position into the models of "operational effects". Also, mapping methods could be further investigated and tested using in-service data.

An unsupervised pattern learning algorithm was used and it was concluded that separating mechanical fault classes by classifying raw data is not practical. Classification of normalised vibration was found to separate fault types and provide a clear recognition of various faults. Unsupervised learning techniques could be further investigated in order to make use of appropriate mapping methods, determine the number of expected faults, and define the datasets, which, if treated separately by a supervised system, would enhance fault diagnosis.

A non-linear supervised diagnostic technique was developed. The capability of the model to identify primary and compound faults was shown to be good even if the value of fault severity lies out of the training range. This suggests that the technique can use a narrow range of in-service data to realize signatures of a wider range of faults. The validity of the non-linear supervised technique could be examined by appropriate in-service data.

Appendix B University of Glasgow Independent Commentary on the MJAD Reports and University of Maryland Publications

Appendix B presents an independent review of both the MJAD reports and the UoM publications that has been produced by the University of Glasgow (UoG).

Whilst presenting the UoG's review in full, GE Aviation wishes to draw the reader's attention to the fact that the original author of the MJAD work under review does not agree with all the comments written by UoG. However, most of the differences can be addressed by comparing the concise summary of the original work (Section 3.3) with the UoG comments (Sections B1 to B6), and by considering the following clarifications, written by one of the authors of the MJAD reports:

i) The UoG's independent review was of limited duration and, at the behest of GE Aviation, focussed on those MJAD documents that were produced for the CAA over approximately a one year period (1990/91). This precluded a detailed investigation of the origin of the MJAD mathematical models and their validation results. These rotor aeroelastic/dynamic models had been described, and in GE Aviation's opinion, been sufficiently validated in public domain papers ([25] to [30]) prior to the CAA work.

ii) Throughout their reports, MJAD indicated that the primary purpose of the mathematical models should be to aid the establishment of fault detection strategies/algorithms for rotor HUMS, not to improve helicopter theory. In this context, it is believed that the focus should be on improved fault detection techniques rather than mathematical model refinements.

iii) Modelling an aircraft with high fidelity is an ongoing challenge that has not been fully achieved and is not necessarily required to develop diagnostics and prognostics. Therefore, MJAD acknowledgment of the limitations of the mathematical models should not be understood to be limited to their comprehensive models; the acknowledgment applies to all existing models.

iv) The technique whereby FFTs of sub-intervals of the full time-record are used as in MJAD/R/63/91 [3] is well known and is applied to date to non-stationery engine vibration data. The wavelet technique was emerging at the time of research and is currently applied by GE Aviation to non-stationary acoustic emission signals; for example, see Reference [42].

v) The MJAD study considered fault detection based on hub loads; see Section A5.4 and Reference [23]. However, the technologies required to measure and transfer data from the rotating frame have not matured to enable practical HUMS implementation. To date, the author is not aware of an in-service HUMS system that uses hub loads for fault detection.

vi) For a perfect helicopter with similar blades and without any fault, the fuselage vibration induced by the rotor will only contain nbR harmonics, n =1, 2, 3, .. and b is the number of blades. For example, for a five bladed rotor, 1R to 4R and 6R to 9R will not exist; the vibration harmonics will only be 5R, 10R, 15R, etc. Most of the rotor system faults induce fuselage vibration dominated by 1R components when the faults are excited by steady or unsteady operations; they do not affect the nbR vibration significantly.

vii) At the time of research, and perhaps to date, available in-service rotor induced vibration measurements did not indicate reliable/repeatable higher harmonics (e.g. 2R) that could be used to conclusively validate fault simulations. The reason is that most of the in-service faults are RTB faults, which mostly induce 1R vibration. Other faults are often rectified early before they cause catastrophic damage. MJAD proposed diagnostic strategies based on measured 2R after applying appropriate pre-processing techniques.

viii) According to MJAD, a mass imbalance fault can be classified as linear since its induced vibration is caused by the mass centrifugal force, which is linearly proportional to the mass; a pitch link fault can be classified as weakly non-linear since its induced vibration is caused by the lift force, which is, almost, linearly proportional to the angle of attack before the stall region; examination of the MJAD crack simulation reveals non-linear characteristics. The classification is very important and would enable the development of powerful diagnostic techniques that allow fault-model based normalisation to remove the effects of fault severities. This classification combined with the fault algebra introduced by MJAD would allow the development of non-linear fault detection deterministic techniques (an extension to the RTB algorithm that could detect more than 3 fault types and compute their severities, see Section 3.3.7.6).

ix) One of the most important MJAD conclusions is that some faults require unsteady operational conditions to manifest themselves. Civil and military helicopter operations contain a number of unsteady conditions such as rotor start-up, rotor shut-down, control reversals and transitions to steady conditions. These unsteady operations do not pose any airworthiness risks.

x) Pre-processing and normalisation should be implemented to (a) remove rogue data, (b) remove operational effects, (c) attenuate variability caused by, for example, age (d) attenuate any undesirable effects of fault severities on features used for fault isolation, etc.

xi) For all existing HUM systems, the vibration measurements are acquired at specified test points, for example, at steady forward speed between 70 to 75 knots. Over such test points, the dominant effect of operational conditions on vibration can be represented by simple equations. Correcting, normalising or pre-processing the vibration over these test points by such equations could be performed in an airborne system; comprehensive mathematical models cannot be used because the limitations of the airborne computing resources.

xii) For some time, GE Aviation has used neural networks to simulate the normal behaviour of subsystems; the expected normal behaviour (e.g. an actuator extension) given by the network is compared with the actual behaviour (e.g. the measured extension) to diagnose abnormalities.

1 UOG Review of the MJAD Reports Produced for the CAA

1.1 Effects of Helicopter Main Rotor Faults on Fuselage Vibration Modes' (Azzam and Andrew, January 1991. (MJAD/R/60/91)[2])

This report describes the results from a brief validation exercise whereby the predictions of the MJAD proprietary helicopter fuselage dynamics model HELFUS for the response to excitation by two different rotor 'faults' (an additional tip mass applied to a single blade, and an asymmetric change to a single blade pitch link) are compared against measured data from experiments conducted on G-BCLC, a Sikorsky S-61 aircraft (MJAD/N/4/89). Few details of HELFUS are available in this report, but it appears to rely on a comprehensive code, ROBUST, to supply its input in the form of the hub loads that are generated in the presence of the rotor faults. The authors conclude that the matching between experimental data and the model amplitude-response is reasonable in most cases. From a superficial perspective it is hard to disagree with this conclusion but, as a point of rigour, such a conclusion needs to be qualified as it depends on the objective or contextual basis for the assessment. The following comment takes the report somewhat out of context, but may be relevant depending on how this software might be exploited in the future. It is not clear from the material presented in this report whether the HELFUS/ROBUST suite of codes yields adequate fidelity for the training, for instance, of any machine-learning based HUMS that might be reliant on high levels of realism for their proper operation, and further in-depth verification of its properties would be required.

Considering how the model is used later in this series of reports, it is a possible concern that the presented validation data extends only to the once-per-rev response of the system: no data is presented to validate the ability of the model to capture the higher-frequency dynamics of the system. Detection of the presence of multiple faults, or differentiation between faults of different type, for instance potentially catastrophic faults from more routine track-and-balance errors, as is acknowledged later in the same report, is certain to require accurate matching of the dynamics over a broader portion of the response spectrum.

It is unclear to what extent the structural model within HELFUS was tuned specifically to the dynamics of G-BCLC for the purposes of the validation. It is often the case that fuselage vibrational characteristics (especially in terms of measured response at a small number of localised sensor positions) show significant cross-fleet variability as a result of manufacturing tolerances, specific equipment installation and internal load distribution. This factor likely has important implications for the application of model-based techniques to the implementation of a uniform, robust, fleet-wide HUMS. The consequences of airframe variability are shown in the results presented in Figs. 19 through 23, where the variation of low-frequency vibrational response (presumably calculated using HELFUS) for a 'typical' Puma fuselage is contrasted with that for a 'stiffened' fuselage.

The major contribution of this report in the present context, though, was to suggest that certain rotor faults might have characteristic 'signatures' that might be identified by suitable measurement of fuselage vibration. In the ideal case, a signature would be unique to a particular type of fault, and would persist in the vibration measurement despite any variation in the damage severity associated with any particular manifestation of the fault. Identification of these signatures might thus form the basis of a viable HUMS for rotor fault detection.

1.2 **The Application of a Math-Dynamic Model to Characterise Blade Cracking/ Delamination Faults during Rotor Start-Up and Shut-Down' (Azzam and Andrew, March 1991. (MJAD/R/63/91)[3])**

This brief report summarises, in simple terms, some of the characteristics of the MJAD math dynamic model for rotorcraft simulation, without presenting the detail beyond the generic features of the model needed to give real insight into this.

The report details the results of simulations of start-up and shut-down of the Sikorsky S-61 model with simulated 'flap-wise damage' on one of its blades. The analysis is interesting for the difficulty it exposes in distinguishing the characteristic signal produced by the damaged blade from the background signal unless the damage is sufficiently severe. The technique whereby FFTs of sub-intervals of the full time-record are used to differentiate the presence of damage from the background signal is interesting. Another possibility though is that, during rotor start-up or shut-down, the measurable effects of the damage are contained within only a small window of the data, in which case the basis functions of the Fourier Transform (because of their non-compact nature) will struggle to differentiate the damage-induced component of the signal from the background signal. In this case a more appropriate transform using a compact basis (e.g. wavelets) may yield better results.

Whether or not pilots would find the requirement for any control actions to excite the natural modes of the system during start-up or shut-down 'acceptable' is debatable. Consideration needs to be given the dangers of ground resonance, blade sailing and similar consequences of excited natural modes when the rotor is operating away from its design speed.

1.3 **An Appraisal of Theory-to-Test Transfer Analysis for Helicopter Vibration Prediction' (Azzam and Andrew, May 1991. (MJAD/R/69/91)[4])**

This is an interesting report that attempts to address the problem of differentiating expected levels of vibration from anomalous readings, from the point of view of implementing a practical rotor HUMS. To a certain extent this approach defines a tangential philosophy to that alluded to in the earlier reports in this series - instead of attempting to identify the signature of anomalous events within a signal that might, to all intents and purposes, appear to the HUMS as an arbitrary variation of some parameter with time, the process suggested in this report involves the direct comparison between the measured signal, and a model-generated reference signal, to determine the presence of anomalous behaviour within the system. This second approach would appear to set very high demands on the fidelity of the underpinning numerical model that is used to generate the reference signal. Arguably the level of fidelity would need to be much higher than in a model that was used simply to identify fault signatures, following the earlier approach. Yet, in terms of robustness, the second approach exhibits the greatest potential from a purely logical point of view since it merely requires detection of a deviation from normality, even though the source of the deviation might possibly be unknown. The first approach, on the other hand, is dependent on identifying the occurrence of a particular, known type of fault (through detecting its signature) and hence a HUMS designed along these lines may not respond to the development of a previously unclassified fault in the system. Although not remarked upon in the report, which of the two fault detection philosophies that is employed will have significant impact on the relative performance of the two machine learning based techniques that are discussed in the fourth report in this series (MJAD/R/75/91).

The report acknowledges the limitations in the MJAD rotorcraft dynamics model and goes on to describe a method of 'correcting' any such model by applying a transfer function based technique that takes the predictions of the rotorcraft dynamics model as input and produces an 'accurised' reference signal for the aircraft, at the flight condition of interest, as output.

A note of caution needs to be applied to this effort. If it is assumed that a fairly conservative approach is in order, then little faith can be placed in the correction technique if applied outside the range of flight conditions for which the transfer functions were originally calibrated. The generation of the transfer functions, given the fact that they will themselves have some, most likely non-trivial, dependence on flight condition would require sufficient measurement of real flight data to span the range of flight conditions for which the correction technique would be relied upon. Given the essentially algebraic structure of the transfer functions, the approach would then require some form of interpolation procedure to generate the transfer functions for those intermediate flight conditions not covered by the original flight test programme. In this case there would be a strong argument for replacing the complex, but limited accuracy, mathematical helicopter model plus 'correcting' transfer functions with a suitably complex, but accurate, set of interpolating functions which captures the dependence of the measured parameters on the flight parameters over the full range of applicability of the technique.

The authors hint at a further complication to the practical implementation of such a correction scheme when they mention briefly the concept of the 'model order' having to be as close to physical reality as possible - although this terminology is never rigorously defined nor are the surrounding arguments fully developed. It seems reasonable to assume that the model used to drive the reference signal should have sufficient fidelity to reproduce the 'normal' components of the reference signal that the presence of an anomaly would perturb - so that, for instance, if a rigid-blade type model were used to drive the reference signal, the technique would be of little use in identifying blade stiffness-related anomalies no matter what correction technique was used to bridge the gap between model prediction and accurate reproduction of normal flight conditions.

Implicit then, but unanswered, is the question of the degree of fidelity required in the model used to drive the reference signal. Little is said in the report regarding the additional, but highly practical, issue of the processing demands that would be made on a HUMS that is required to generate a reference signal (presumably in real time), using some form of computational rotorcraft model, for continuous comparison against measured airframe data. A trade-off would certainly be required between model fidelity and speed of execution, and it remains an open question, given the current state of the art in rotorcraft modelling whether an accurate enough reference model could be made to run in real time on any practical airborne computing system. On the other hand, if the rotorcraft reference model can be arranged to run off-line, with its input based on sufficient pre-recorded flight data so that the flight conditions experienced by the helicopter can be reconstructed to sufficient accuracy for the reference model to be able to produce valid results, then the issue might be shifted to the slightly less technologically challenging one of ensuring sufficient airborne data storage capacity.

1.4 The Implementation of a Pattern Recognition Technique to Classify Main Rotor System Faults' (Azzam, Callan and Andrew, September 1991. (MJAD/R/75/91)[5])

This report contrasts two basic philosophies for the implementation of machine learning techniques for rotor fault detection. The first philosophy uses a supervised learning method based on a neural network, while the second uses unsupervised learning based on clustering. The supervised method is most compatible with the first HUMS philosophy described in MJAD/R/30/90, i.e. the identification and classification of known faults with known signature. The obvious disadvantage of this technique though, in requiring sufficient training data to be effective at identifying faults in a variety of contexts, including in the presence of other faults, system noise and despite variations in the background signal, is well expressed by the authors of the report.

Although little detail of the implementation of the method is given, the authors describe an application of the alternative philosophy, i.e. unsupervised learning where a pattern recognition tool is used to classify input data into output 'clusters', to which physical significance can later be attached. It appears that the approach was used to classify the various components of the structural response of the fuselage, as measured at a small number of locations, into a number of tell-tale 'groups'. Elevated structural response throughout any one of the groups could then be used as a signature pattern that would be indicative of a particular rotor fault having occurred. The identified groups of signals could possibly be used as the basis for constructing a library of fault signatures required for training a neural network. The analysis described in the report, conducted using the MJAD math-dynamic model to generate the fuselage dynamics in response to simulated rotor faults, appears to yield a number of reasonably clearly differentiated such groups of signals that could be used to determine the presence of the three types of rotor faults that were simulated.

An important characteristic of any prototype fault signature, though, is that the signature should remain unaffected (or, at least, should remain clearly differentiable from other signatures) even if the fault severity were to vary (in other words, the fault should be associated with a particular equivalence class of signatures for its unequivocal identification). Some of the results presented in the report show evidence of cross-contamination, or non-uniqueness, between the groups of signal inputs that respond to any given fault within the system. The degree of this contamination appears to depend on the severity of the simulated fault, resulting in a violation of the previously mentioned requirement that the group of signals identified with a particular fault constitute an equivalence class. Hence the grouping process cannot be considered to have properly defined a sequence of unequivocal fault signatures, and, in this light, cannot thus be judged as having been an unqualified success - as was indeed realised by the authors of the report. It is interesting to speculate on how the grouping procedure would have fared in the presence of multiple faults in the system, but no such study appears to have been undertaken.

The lack of unqualified success of this simple approach is most likely a measure of the extremely complex, coupled and non-linear nature of the rotorcraft system. The authors introduce the terminology of 'linear' and 'non-linear' fault types (the distinction depending on whether or not the fuselage response to a number of rotor faults is a simple linear superimposition of the responses to the individual faults) but do not extend their analysis to its logical conclusion. Firstly they do not determine the fundamental orthogonality of their set of identified fault-indicating groups - and hence their suitability as a basis (the word used here in its mathematical sense) for identification of even the simplest, linear faults. Secondly they do not categorise the faults under consideration according to their proposed linear/non-linear dichotomy (although this issue is addressed, to some extent, for some simple examples in the fifth report in the series, MJAD/R/86/91. For instance, track and balance faults are broadly classified as 'weakly non-linear') - even given that the suspicion must be entertained, especially given the mathematical structure of the equations of motion that govern the rotorcraft system, that most faults will be inherently non-linear and hence not readily amenable to identification by a process based on underlying orthogonality of fault signatures.

The success of the approach that was followed was reliant on the eventual validity of some untested assumptions, in particular those regarding the manner in which rotor faults manifest themselves in the fuselage vibration signals being analysed. The process that was followed appears to have been essentially to explore various numerical possibilities using an experimental approach. A slightly more rigorous approach to the underlying theory, although perhaps rather more difficult to conduct in practice than to advocate, might have helped to constrain the approach. In particular the absence of some of the rigorous underlying theory regarding the mapping between rotor faults and fuselage vibration modes (do all rotor faults induce fuselage vibration? Does each fault induce a unique vibration signature?) adds technological risk to the endeavour. A very useful summary of numerically-generated information that might be useful for fault-signature identification and group construction is contained in the supplementary MJAD report MJAD/R/30/90 'The Identification of Potentially Catastrophic Helicopter Rotor Faults (Theoretical Study)'. In this report, correlations between fault and vibration signature are presented not in terms of fuselage vibration response, but rather in terms of vibration frequency and phase information measured directly at the rotor hub. A careful study of this material reveals clearly the problems that would be associated, as described above, with unique fault differentiation (for example certain potentially catastrophic faults manifest very similarly to benign faults such as blade tracking errors), but the vibration signatures do seem to be significantly better correlated with the various faults than in the case of the fuselage vibration measurements, and, additionally, appear to obey a logic that concurs very directly with a simple mechanical description of the dynamics of the rotor. If this observation could be substantiated, it might offer the possibility that suitable placement of sensors where they could directly measure rotor vibration, rather than inferring it, for example, from measurements of fuselage vibration, might obviate, or at least be traded off against, the need for high-fidelity modelling of the helicopter system (in the sense that it is usually implied, i.e. simultaneous, accurate resolution of the aerodynamics and structural dynamics of the helicopter) and allow a much simpler, purely mechanical model for the correlation between fault and signature to be used as the basis of a fault-detection and identification system. This argument would be weakened considerably though in the presence of strong non-linearity in the system that has nothing to do with sensor placement, for instance in the presence of strong coupling between the aerodynamics and structural dynamics of the rotor. Evidence that the behaviour of certain faults (in particular those resulting in a loss of torsional stiffness of a blade) may not be adequately characterised by a simple mechanical model, no matter what the sensor configuration, is presented in the Maryland group's 1996 AIAA conference paper.

An additional point is that a (suitably exercised) unsupervised neural network could, quite feasibly, have fulfilled the role of the 'theory to test transfer function' postulated in the third report in this series (MJAD/R/69/91), or, rather more so, the set of interpolating functions that, as described earlier, might have been used to supplant the reference model and associated transfer functions. The advantage of the neural network approach in this context is that the effort expended in initial, direct derivation of the interpolating functions (or transfer functions) might be traded off against the time taken for the number of spurious false alarms generated by the system to abate and thus for the system to be trusted to discriminate reliably between anomalous events and events associated with the rotorcraft's correct operation.

1.5 A Nonlinear Supervised Diagnostic Technique for the Recognition of Main Rotor System Primary and Compound Faults' (Azzam and Andrew, November 1991. (MJAD/R/86/91)[6])

This report discusses some of the technicalities involved in implementing an integrated HUMS system that is capable of early detection of rotor faults, and contains some interesting extensions of previous work.

The idea of 'pre-processing' system data to provide the diagnostic system with a better chance of capturing emergent faults is introduced. This approach is well justified where this pre-processing is done to prevent damaged sensors, data corruption or other external effects from provoking an incorrect or spurious diagnosis of a rotor fault ('data integrity checking,' in the authors' terminology). Where it is done to provide a neural network-type system with data that has been conditioned according to some pre-derived notion of the (fault-free) dynamics of the system to allow it to better discriminate damage-related features within the data, the logic is less easy to follow though since there would then be the requirement to supplement the machine-learning capability of the system with an additional layer of knowledge regarding the behaviour of the system. This knowledge would need to be derived from some external source. The authors support their arguments for the use of pre-processing, by recalling their description in the first report in the series (MJAD/R/60/91) of how the use of fuselage modal amplitudes allowed clearer definition of rotor faults than vibration measurements at isolated locations on the helicopter fuselage. If it is taken into account, though, that the modal amplitudes are derived essentially as combinations of vibration amplitudes at isolated sensor locations, and that, in all likelihood, a neural network would be capable of discovering the intervening analysis for itself, then the argument might have been more simply expressed in terms of an analysis of the minimum set of information that would be required to be extracted from the system in order to expose a fault of a particular type.

The authors allude to a process whereby the 'machine dependent' component of vibration (implying a requirement for a 'reference' signal as described in MJAD/R/69/91) is stripped from the signal as a pre-processing step before passage through the neural network. This process limits the function of the neural network to a diagnostic one (fault identification or classification) rather than one of fault detection, in the sense that any deviation from the reference signal might potentially be construed as indicating the presence of a fault in the system.

The report goes on to present an extended example of the decomposition of faults into linear, weakly non-linear and non-linear type, supporting some of the discussion presented in the fourth report in the series (MJAD/R/75/91) and providing how multiple and non-linear faults might be differentiated and eliminated through a maintenance strategy based on the output of the HUMS.

1.6 The Use of Infield Data and Math Dynamic Models to Evaluate Diagnostic Strategies for Main Rotor Systems Faults' (Azzam and Andrew, December 1991. (MJAD/R/90/91)[7])

This report is essentially a summary and amplification of some of the issues presented in the previous five reports in the series.

A particular concern with any diagnostic strategy is that the process should not confuse a damage-related vibration signature with similar effects induced by normal operation of the aircraft. The authors present a somewhat clearer example than in the previous reports in the series of how vibration measurements might be corrected to remove the effects of operational conditions on the measured data. Essentially the process as described relies on an explicit representation of the effects of operational conditions on the vibrational characteristics of the system through a mathematical model - presumably this model need not necessarily be algebraic, as in the example presented, but might indeed be constituted more robustly by using a comprehensive rotorcraft modelling approach such as that advocated in the first two reports in the series. The analysis to produce corrected vibrational amplitudes, though, is potentially flawed as it is valid only for a linear dependence of the vibrational characteristics on the operating conditions of the aircraft. This assumption is certainly not consistent with any of the example dependencies presented earlier in the same report. Correction of the analysis presented in this report to yield a fully valid approach would be non-trivial. In any case, though, the approach appears to layer yet another level of complexity onto the process of extracting damage-related information from the 'normal' background signal - it is not clear what higher purpose normalisation to a standard reference operating condition serves if an accurate representation of the normal behaviour of the system at the actual flight condition is already available.

2 UOG REVIEW OF THE UOM PUBLIC DOMAIN PAPERS

2.1 UoM Apr-1995 [8] and Oct-1996 [9] Papers

Title: Formulation of a Rotor-System Damage Detection Methodology

Authors: Ganguli, Chopra and Haas,

In this set of papers, essentially a conference paper augmented and enhanced slightly for the purposes of journal publication, the Maryland authors introduce the basic elements of their approach to rotor fault identification and characterisation. They adopt, although, arguably, from a rather overly axiomatic point of view, the premise that damage detection requires as its basis an accurate mathematical model of the dynamics of the helicopter system, or, at least, of the dynamics of the rotor in question. To this end they employ their code UMARC (University of Maryland Advanced Rotor Code) incorporating a beam-type finite element model for the dynamics of the rotor blades and a free-wake model for the aerodynamic environment of the rotor. This model has been reasonably well validated in a wide range of flight conditions and, at the time of writing of the papers, this model did not fall too far short of the state of the art in helicopter rotor modelling. From a present-day perspective, however, the model could be criticised for its rather simplistic treatment of the dynamics of the rotor wake and some deficiencies in its lifting-line treatment of the blade aerodynamic behaviour, as exposed in an admirable sequence of recent papers by Chopra and Datta. It should be borne in mind throughout that some of the results presented in the sequence of papers under review are no doubt influenced by these deficiencies.

The effects of blade damage are introduced into the model via modification of the structural, inertial or aerodynamic characteristics of one of the rotor blades. Various types of damage (moisture absorption, loss of trim mass, misadjusted pitch-link, damaged pitch control system etc.) are characterised in terms of their effect on the structural stiffness of the blade about its various axes, changes in the aerodynamic coefficients that are applicable along the length of the blade, or simply by re-distribution of the mass of the blade. In this vein the authors are somewhat guilty of pandering to the eventual practical application of their work in the sense that a more rigorous and complete treatment of the problem could probably have been achieved by examining the influence of changes of mass, stiffness etc on the dynamics of the system before attributing these changes to particular rotor faults. Such an approach might have made very clearly apparent the non-uniqueness of the mapping between fault and vibration signature, and thus avoided entanglement in some of the logical issues involved with identification of fault signatures and groups that occupied such a significant proportion of their own as well as the MJAD study. In these early Maryland papers, no attention was paid to possible variations in, or growth of, fault severity - faults are simply treated as 'present' or 'absent', in an attempt to perform a basic characterisation of fault signatures within the rotor hub vibration signal (as measured in the non-rotating frame of reference). This deficiency was corrected in later papers in the series.

2.2 UoM Oct-1995 Paper [10]

Title: Helicopter Rotor-System Damage Detection

Authors: Ganguli, Chopra and Haas

In this paper the analysis contained in the previous papers is extended significantly by considering the effects of increasing damage severity on the hub vibration signature. Results obtained with the UMARC code led the authors to conclude that 'significant' changes in individual blade properties may be required to be induced by any damage to an individual rotor blade before the changes become detectable in the hub vibration signature, and that detection in any case might be critically dependent on the flight condition in which the rotor is operated (for instance, the symmetrical dynamic and aerodynamic environment of the rotor in hover may mask certain faults). This observation has bearing of course on the practicality of a HUMS that is capable of early identification of a developing fault, as well as to the efficacy of the proposed MJAD strategy of using ground-based tests to identify the presence of rotor faults. The argument presented in the paper is open to attack though through its rather arbitrary and subjective definition of what constitutes significant change in the vibrational characteristics of the system for the purposes of fault detection, and might have been better expressed by direct reference to the background sources of variability within the vibration signature of the system, for instance the variability in the signal introduced by simple tracking errors.

2.3 UoM Apr-1996 [11] and Apr-1997 [12] Papers

Title: Detection of Helicopter Rotor System Simulated Faults using Neural Networks

Authors: Ganguli, Chopra and Haas

This set of papers again consists essentially of a conference paper and a second archival publication that in this case is reduced somewhat in scope compared to the original, possibly in response to reviewer comments. In these works, the authors give an admirably comprehensive description of their feed-forward neural network model and the effects of variations in its architecture on its efficiency and performance. The presentation within this sequence of papers is more informative than that provided within the MJAD series of reports and should provide useful guidance to any similar future studies. The approach used is to employ two networks in series in a supervised-learning type approach, the aim of which is to detect and quantify a specific, pre-defined set of rotor faults. The first network is used to identify the type of damage to the rotor before a second network, trained specifically on the identified fault type, is used to identify magnitude of the damage. Obviously this structure relies heavily on the existence of an appropriate supervised learning regime, and the data for this is provided by output from the UMARC code with various types of simulated damage. No attempt is made to assess the performance of the resulting system outside the context of the UMARC-generated data set, but this approach is consistent with their approach through analogy with the real world as described in the introduction. In this set of papers, some elements of the real world are introduced by contaminating the model-generated vibration signal with white noise to simulate the effects of natural variability in the data. Improvements in the performance of the neural network in detecting faults from the contaminated model vibration signal is then demonstrated when the neural network is trained on similarly contaminated data. Nevertheless a significant degradation in the ability of the neural network to identify correctly the type of damage is observed as the noise level within the vibration signal is increased. Given the mechanical character of the system, real-world contamination of data is likely, of course, to have stronger content at certain specific frequencies than at others, and some of this frequency-specific content is likely to mask pertinent features or trends in the vibration signature in a much more obscure fashion than is produced by a simple white noise contamination. It is not entirely clear whether even the qualitative conclusions regarding the performance of the model system (in the presence of white noise) can thus be extrapolated with confidence to the real world.

This paper is perhaps most interesting for the methodology that it suggests for the identification of fault signature groups, using the terminology adopted in the MJAD study, in a rather more rigorous way than is described within the MJAD reports. This is despite the fact that the Maryland researchers appear entirely unaware of the relevance of their work to resolving some of the problems encountered during the earlier MJAD study. In the Maryland study, fault groups are identified through a systematic reduction in the size of the input vector supplied to the neural network; a reduction is regarded as valid if it leads to negligible change in the output of the network. Possible redundancies of the input vector that could be exploited by suitable sensor placement were also examined, and it was demonstrated that the N-per-rev and lower-frequency content of the blade-tip dynamic response, the hub forces and hub moments was sufficient to identify and characterise each of the specific fault types that were modelled during the study. The conference paper, in addition, contains the results of some simulations that support the MJAD contention that the vibration signature of some faults bears a relatively linear relationship to the fault severity whereas the signature of others, particularly where there is significant coupling between the aerodynamics and the structural response of the system, bear a highly non-linear relationship to the fault severity and thus may be more difficult to identify using any technique which relies on distinguishing any specific signature of the fault from the background signal, no matter what sensor configuration is employed.

2.4 UoM Sep 1996 [13] and Jun-1998 [14] Papers

Title: A Physics-Based Model for Rotor System Health Monitoring

Title: Helicopter Rotor System Fault Detection using Physics-Based Model and Neural Networks

Authors: Ganguli, Chopra and Haas

This set of papers is again a conference-journal pair. Together with some minor enhancements and embellishments to the material described in the previous papers in this series, these papers contain some simulated results that illustrate the performance of the neural network, described in the earlier papers, in identifying multiple coexisting faults in the rotor system. As in single-fault identification, the neural network is reliant on the severity of all faults present in the system exceeding a certain threshold value that is related to the amount of the noise in the data, before detection of faults becomes possible. Although not remarked upon by the authors, comparison of their data for single- and multiple-fault detection suggests that the threshold value at which any particular fault is identified is significantly elevated by the presence of other faults within the system. In addition, a somewhat uncharitable interpretation of some of the data that is presented leads to the conclusion that, because of the structure of their model, certain faults might be almost completely masked by the presence of certain other faults in the system (e.g. in the results presented, a very high severity of 'lag damper damage' is required for its correct identification in the presence of 'moisture absorption'). This interpretation undermines the validity of the input vector reduction employed in the previous papers in the series. As in the MJAD study, through adopting a heuristic, demonstrative, trial-and-error based approach, the Maryland researchers seem to have missed a prime opportunity to return to the fundamental issue of whether an orthogonal basis, consisting of individual fault signatures, exists within the input vector to the neural network and thus possibly to have resolved one of the fundamental issues that appears to bedevil the implementation of any approach to the identification and characterisation of rotor faults.

2.5 UoM Apr-1997 [15] Paper

Title: Helicopter Rotor System Health Monitoring using Numerical Simulation and Neural Networks

Authors: Ganguli, Chopra and Haas

This paper reports a rather ingenious attempt by its authors to construct a numerical model for the rotor-induced vibration of the fuselage. It is first assumed that the vibration levels at selected points on the fuselage are related to the various components of the rotor vibrational loading via a non-linear, algebraic relationship. This relationship is then constructed using a neural network that has been supplied with UMARC-generated rotor vibration data as input and trained to replicate flight-test measured fuselage vibration, under normal operating conditions, as its output. This model is then used to examine the influence of various rotor faults on the vibration spectrum experienced at the pilot's seat, by propagating the model-generated signal for the damaged rotor through the neural network. This information is then used, as in the first paper in this series, to perform a basic characterisation of fault signatures, but this time within the fuselage vibration spectrum rather than, as before, in terms of the vibrational characteristics of the rotor hub itself. The approach adopted in this paper is interesting but somewhat unrigorous, a serious concern has to be the extent to which training data based on a helicopter in operational if not perfect condition is sufficient to exercise the neural network during its training so that it is sufficiently well conditioned to be able to fully and accurately replicate the effects of rotor damage on the fuselage vibrational signal. The unique possibilities offered by an unsupervised, rather than a supervised neural network in this respect, as described earlier in the present report, seem not to have been considered at all by the Maryland researchers.

2.6 UoM Aug-1998 [16] Paper

Title: Simulation of Helicopter Rotor-System Structural Damage, Blade Mistracking, Friction and Freeplay

Authors: Ganguli, Chopra and Haas

This paper contains very little that is not already available within the previous papers in the series, other than a small discussion concerning the validation of the vibrational predictions of the UMARC code against limited experimental data for the UH-60 - doubtless in response to the conditions for publication in this particular journal.

2.7 UoM May-2000 [17] Paper

Title: Rotor System Health Monitoring Using Coupled Rotor-Fuselage Vibration Analysis

Authors: Yang, Chopra and Haas

The material contained in these and the following papers indicates somewhat of a shift from the path followed earlier in the Maryland programme, with more of an emphasis being placed on identification of the effects of rotor damage in the fuselage vibrational response and very much less emphasis being placed on development of the neural network-based techniques developed earlier in the programme. The data obtained during the 'Air Vehicle Diagnostic System Programme' (Haas, D.J., and Schaefer, C.G., "Air Vehicle Diagnostic System Technology Demonstration Program", 55th American Helicopter Society Annual National Forum, Montreal, May 1999.) during which a SH-60 with various relatively benign rotor faults was flown, was chosen as the basis for a comparison of numerical predictions of fuselage vibration against flight test. Comparisons were relatively disappointing throughout, with significant discrepancies in phasing and amplitude being shown between test and simulation for the once-per-rev vibration experienced at the location of the pilot and co-pilot's seats in the presence of the various rotor faults. Minimal attempt was made to correlate their model against the broader spectrum of vibration required to be captured for the purposes of input to a rotor HUMS.

2.8 UoM May-2001 [18] and Mar-2004 [19] Papers

Title: Rotor-Fuselage Vibration Analysis for a Dissimilar Rotor with Single and Multiple Faults

Title: Vibration Prediction for Rotor System with Faults using Coupled Rotor Fuselage model

Authors: Yang, Chopra and Haas

Possibly in an attempt to redress some of the deficiencies in the results presented in the previous paper, the effect of the hub-mounted vibration absorber of the SH-60 in modifying the vibration spectrum as measured at the pilot or co-pilot's seat is discussed in detail in this pair of papers. Of course, the presence of such devices within the helicopter rotor system poses significant additional problems to the accurate modelling of the dynamics of the rotor and may act to suppress or enhance the signature of certain forms of damage to the rotor. The major contribution of this set of papers is perhaps to expose the difficulty in accurately modelling the vibrational characteristics of the fuselage in response to forcing by rotor loads in the presence of faults. In most cases, correlation between flight test data and prediction of fuselage vibration is singularly unspectacular, even when a model for the vibration absorber is included in the simulation, whereas in the single case where flight test data for a direct rotor performance parameter (blade-tip displacement) in the presence of a fault (control-link freeplay) is compared against prediction the agreement is significantly better - although the case presented is perhaps not as challenging to model correctly as some other rotor faults. These observations are consistent with significant inaccuracies in the model of the dynamic characteristics of the fuselage, despite the sophistication of the finite element-based approach that was adopted in the Maryland work. These comparisons, although very limited, do not bode well for the effectiveness of a HUMS based on identification of fault signatures from vibrational data measured at any point on the helicopter fuselage. The Maryland study suggests that the complexity of the transfer function linking rotor loads to fuselage may pose significant practical problems to implementation of a rotor HUMS based on fuselage vibration, and, more specifically, that the quality of available fuselage modelling techniques would be the major impediment to the implementation of any HUMS based on a neural network that is reliant on numerically generated data and supervised learning for its effective operation.

2.9 UoM Jun-2002 [20] and Jul-2004 [21] Papers

Title: Sensitivity of Rotor-Fault-Induced Vibrations to Operational and Design Parameters

Authors: Yang, Chopra and Haas

This pair of rather obscure papers attempt to dissect the influence on fuselage vibration of various operational rotor faults, in particular mass imbalance, tab and pitch-link misadjustment. Some reference to the flight test data obtained in the presence of these faults on the SH-60 helicopter during the AVDSP tests mentioned in the earlier papers is again made, but it is not entirely clear from the papers how this data is actually employed in context. Most of the results presented in the paper appear to be obtained by forcing their finite-element model of the SH-60 fuselage using, presumably, model-generated rotor loads in the presence of these faults and observing the effects on the various modes of vibration of the fuselage. In some cases, comparisons between predictions and test data are made, but there are a number of missing pieces of information that make interpretation of presented data extremely difficult. Significant effects of operational condition (weight and centre of gravity location as well as the characteristics of the bifilar vibration absorber mounted on the rotor hub of this particular type of aircraft) on the vibrational spectrum are observed but the data is left in rather unprocessed form and hence it is difficult to determine whether trends are as expected or whether any anomalies or non-linearities might be encountered that might hinder effective operation of a rotor HUMS system, for instance. Some interesting data is presented showing significant differences in the vibrational spectrum for models of SH-60B and SH-60F fuselages that may have implications for cross-fleet uniformity of the effectiveness of a HUMS based on identification of faults from analysis of fuselage vibration. As such, some of the material in this paper follows firmly in the footsteps of the earlier components of the MJAD study.

INTENTIONALLY LEFT BLANK

Appendix C MORS Database Analysis – Aircraft Related Data

Appendix C presents the results of a further detailed analysis of the rotor fault data, including the breakdown of faults by aircraft type. The presentation of this data is intended for assessment of where it would be best to apply any further research effort, and where a monitoring system would be most useful, rather than analysis of aircraft reliability. In many cases, a high calendar fault rate on any aircraft type is due to the high flying rate of that aircraft, rather than any particular problem with the reliability of the rotor components. Where the aircraft type is the category axis the data is presented as faults per 100,000 flights, in all other cases the number of occurrences for each aircraft is shown. The data for benefit rating is not factored, since the likely benefit for an aircraft type is related to the amount of flying it does.

Figure 1 shows the main and tail rotor fault rates (per flight) for UK civil aircraft between 1976 and 2005. For clarity, the Chinook aircraft has been omitted from this graph; it had by far the highest fault per 100,000 flight rates, at 31.7 for the (main rotor) blades and 138.8 for the hub. The older aircraft types have higher fault rates than the newer types. Of the currently operating types, the S61, S76 and Super Puma have the highest rates, with the Super Puma data being dominated by hub faults.

Figures 2 and 3 show, for all data, the breakdown of the main rotor hub and blade faults by aircraft type and year. The Super Puma has had most hub faults, Figure 2, over a long time period (partly due to the high flying rate), with the Chinook problems confined to the 1980s, after which the aircraft was withdrawn from civil operations. S76 and S61 hub problems also feature in many years. In the equivalent data for the blade faults, Figure 3, the S61 and S76 aircraft types dominate, even in recent years, with the Dauphin and Super Puma less significant.

Figure 4 gives a further categorisation of the hub faults for the main rotor between 1976 and 2005; these were dominated by damper problems, then pitch control, pitch link and spindle issues. All these occurred on several different aircraft types. For the blade faults, Figure 5, most were caused by cracks or delamination on many different aircraft types. Pocket problems were only reported on Wessex or S61 aircraft, and the tip caps on the S76 have many reported occurrences.

In the breakdown of tail rotor faults by aircraft type and year (Figures 6 and 7), apart from the early years most faults occurred on Super Pumas and Dauphins, and there were generally more hub problems than blade problems. Pitch link problems were the major cause of tail rotor faults (Figures 8 and 9), followed by blade sleeve defects which predominantly occurred on the Super Puma.

Figure 10 shows the expected benefit by aircraft type, giving a rough indication of the potential number of accidents which could be prevented by a rotor monitoring system. The totals are greatest for the S61, S76 and Super Puma, with the Super Puma having the greatest potential benefit of all the helicopter types for tail rotor monitoring.

Figure 11 details the causes of accidents and serious incidents, and the aircraft types on which they occurred. These were spread over most types, but the tail rotor hub on the Super Puma was the source of the highest number of accidents and serious incidents (probably due to the high flying rate of this aircraft type).

1 Discussion

Of the aircraft types currently operating, the Super Puma and S76 are the most prominent types in the main rotor hub faults and the S61 and S76 the blade faults. However, these aircraft had high flying rates, so the graphs showing numbers of faults per flight give a different picture. The Chinook aircraft had the highest by an order of magnitude (not shown in Figure 1), then the older types of the Wessex and Puma. The Bell 214 had a high rate, but with relatively few flights, then the Super Puma, S61 and S76 had the next highest rates, though with fewer than 10 faults per 100 000 flights.

In the sub-classification of fault causes, most main rotor blade faults were cracks (S61, S76), then pockets (S61). Most main rotor hub faults were caused by dampers (S61, S76) or pitch control/links. For the tail rotor, the faults were mainly pitch link related (S61, Dauphin, Super Puma).

The benefit ratings are greatest for the S61, S76 and Super Puma aircraft types, corresponding to the number of faults on these aircraft.

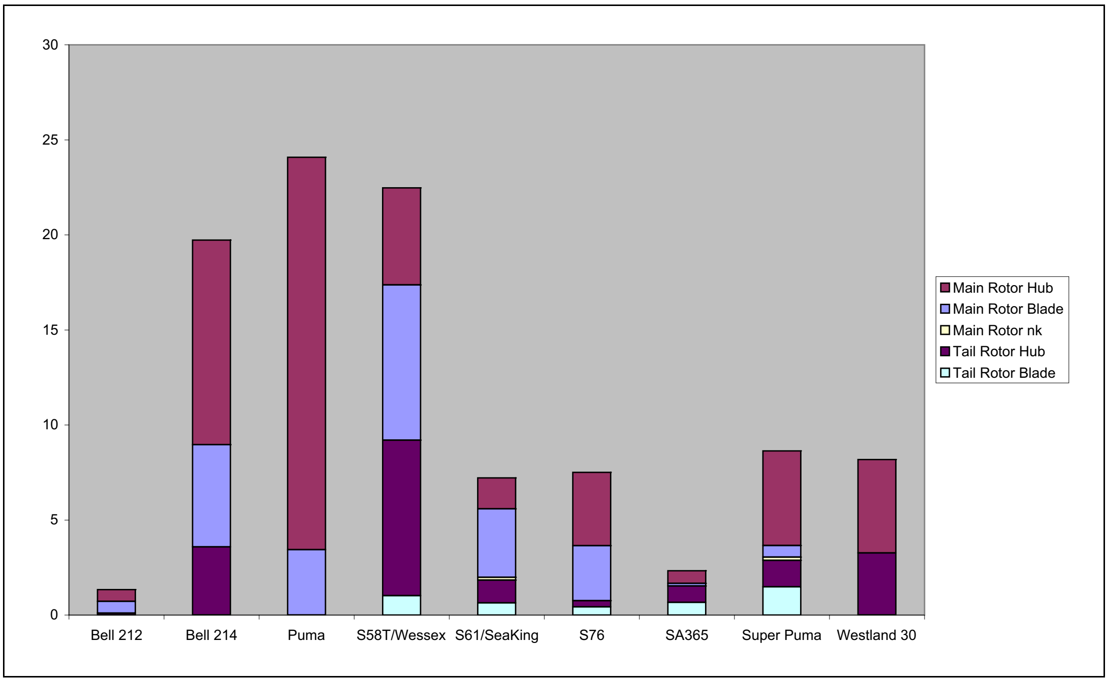

Figure 1 UK Civil Occurrences Per 100,000 Flights by Aircraft Type

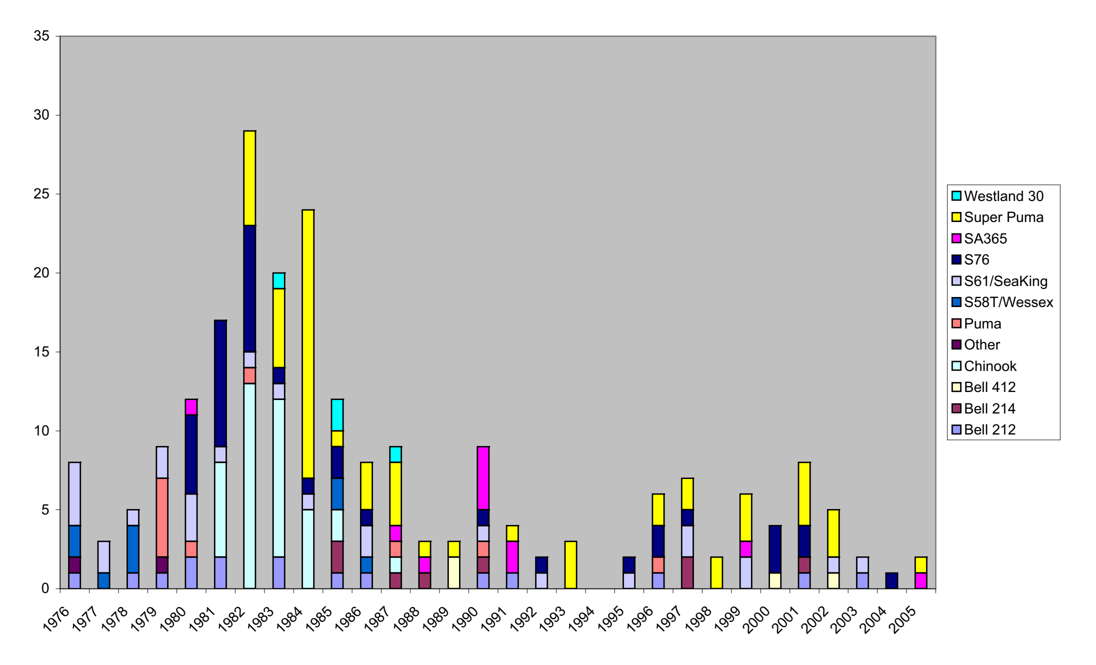

Figure 2 Aircraft Types Contributing to Main Rotor Hub Faults by Year

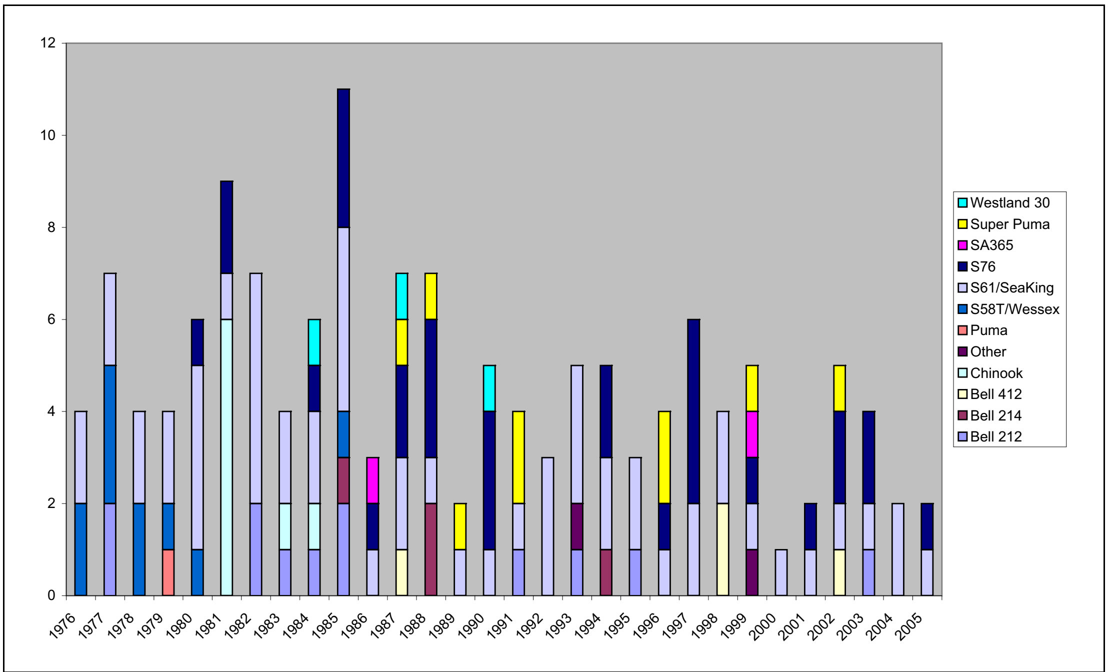

Figure 3 Aircraft Types Contributing to Main Rotor Blade Faults by Year

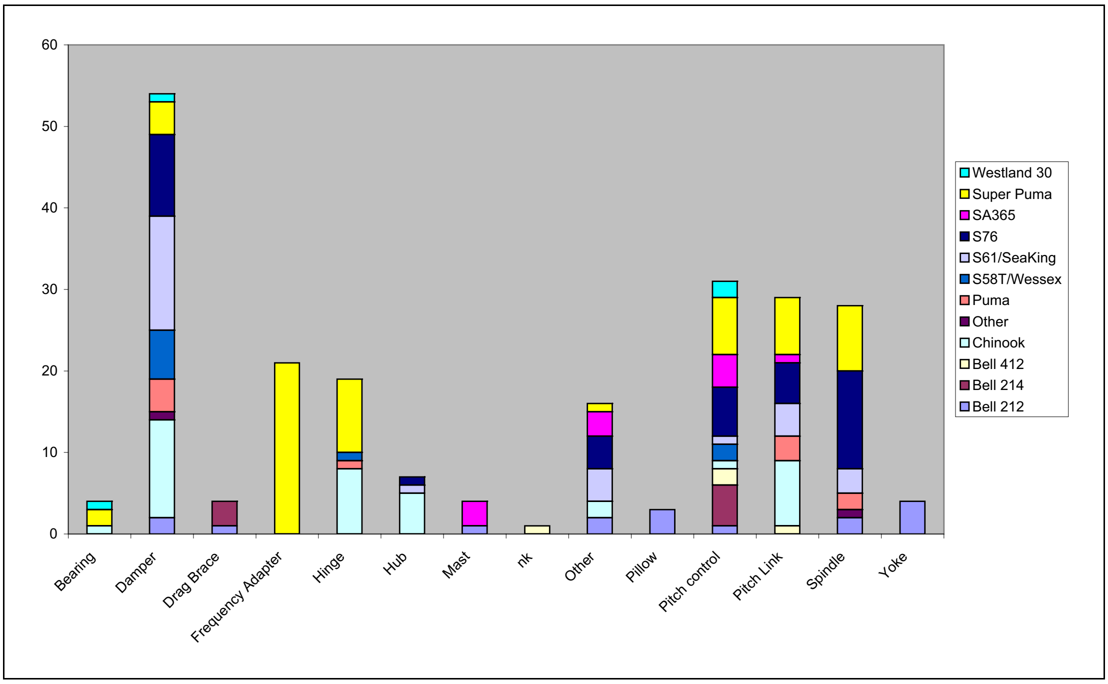

Figure 4 Main Rotor Hub Fault Classification

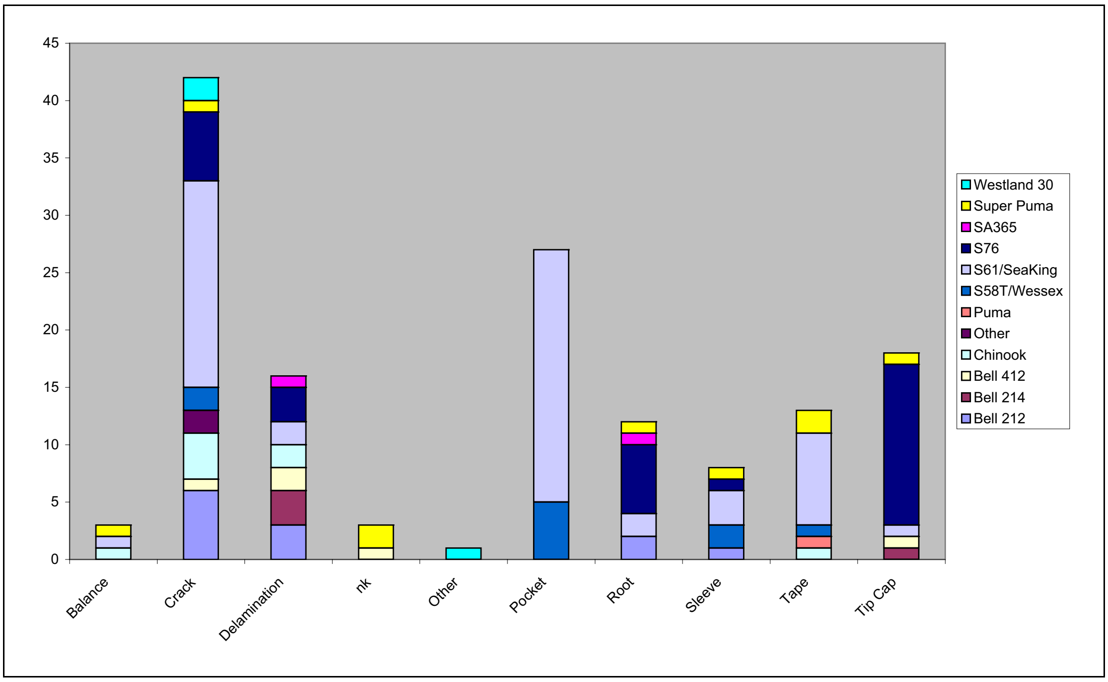

Figure 5 Main Rotor Blade Fault Classification

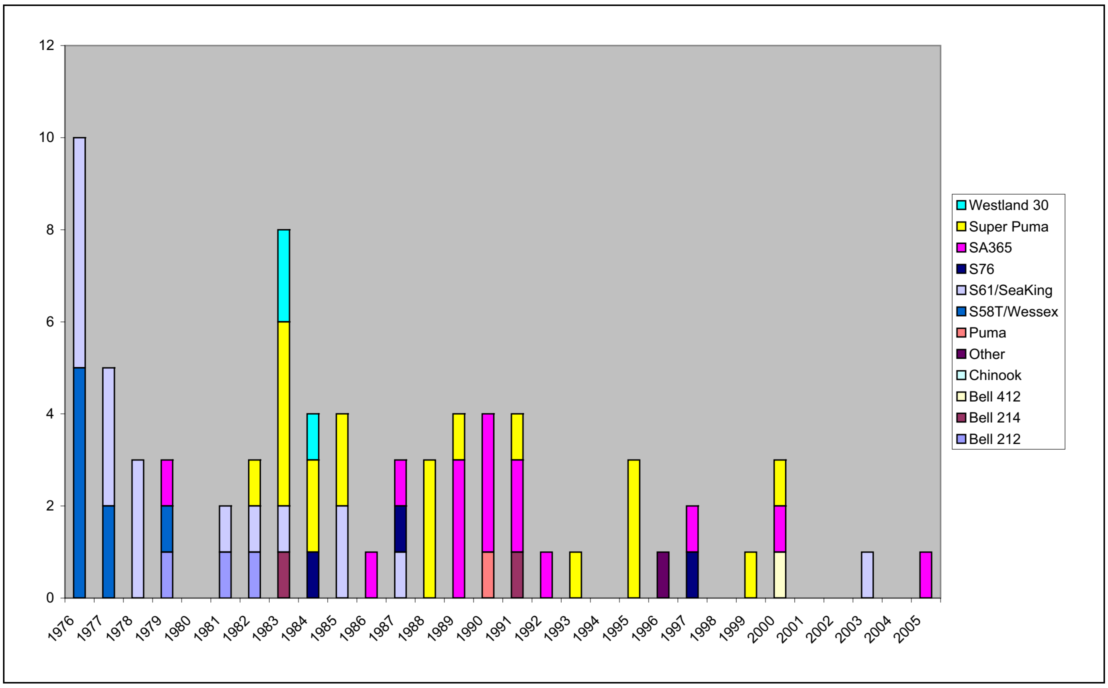

Figure 6 Aircraft Types Contributing to Tail Rotor Hub Faults by Year

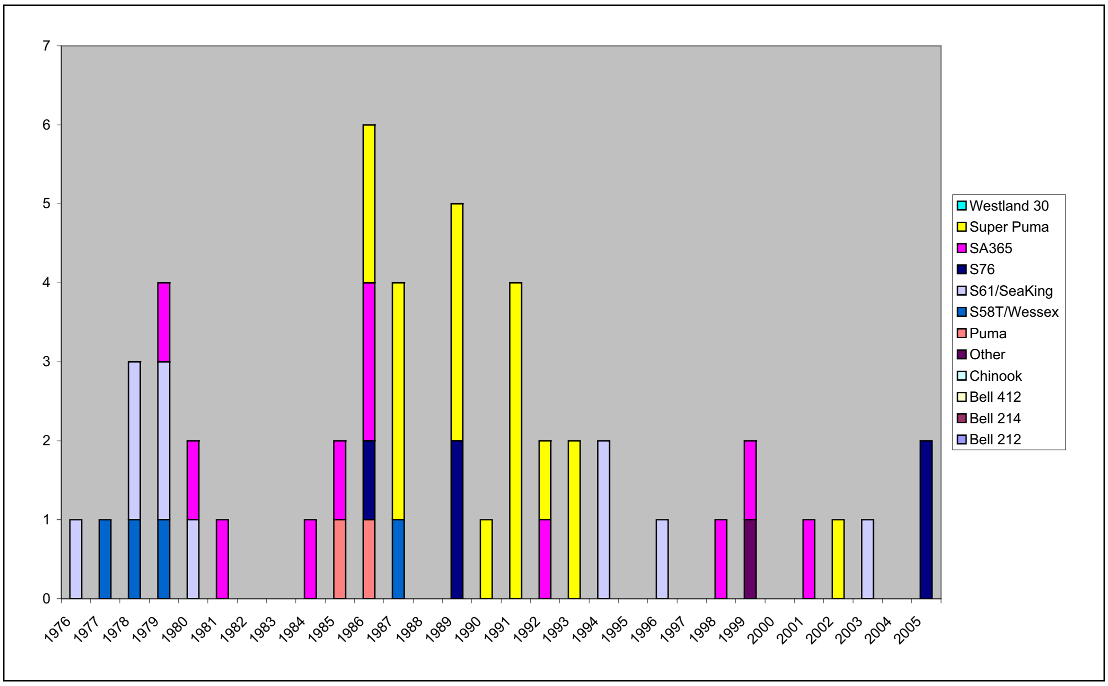

Figure 7 Aircraft Types Contributing to Tail Rotor Blade Faults by Year

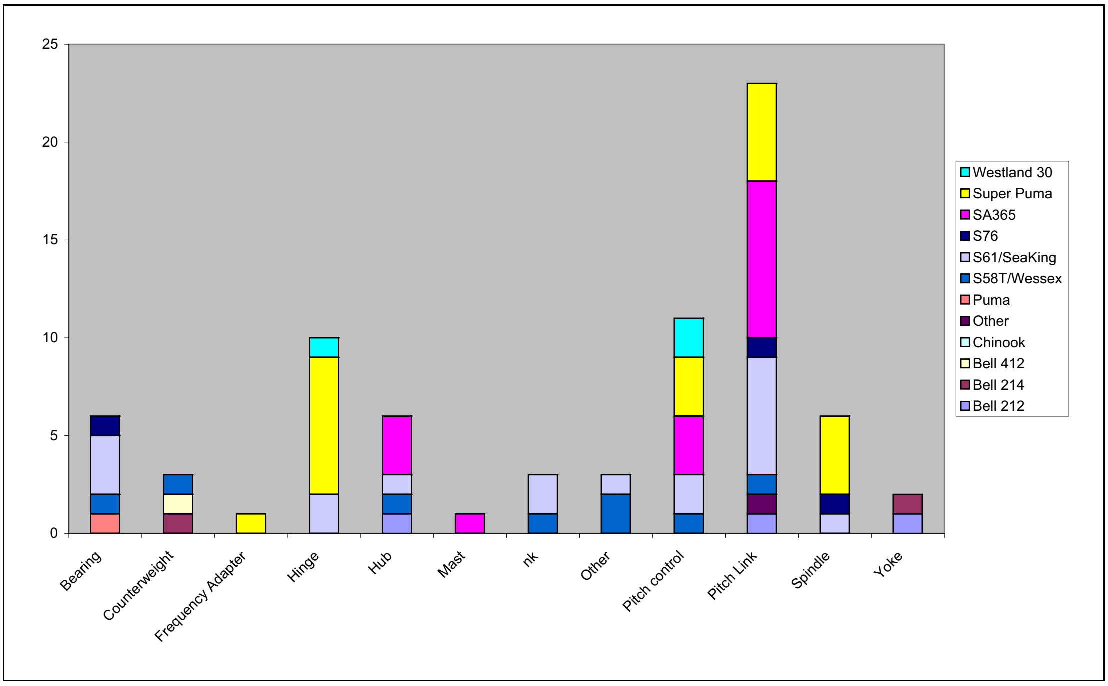

Figure 8 Tail Rotor Hub Fault Classification

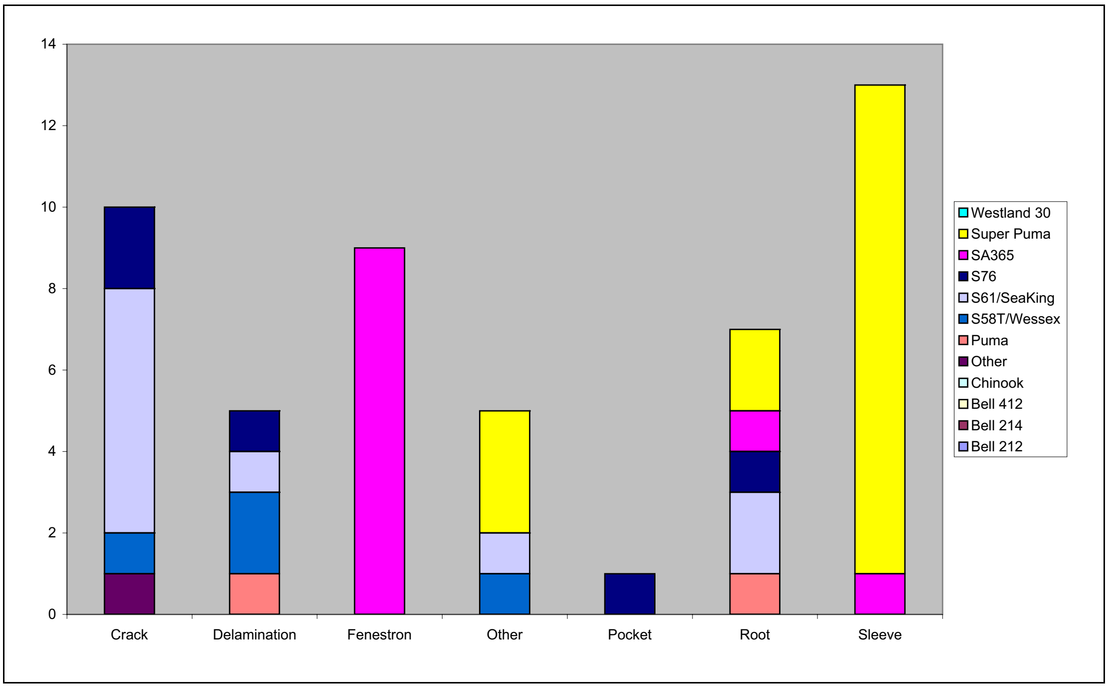

Figure 9 Tail Rotor Blade Fault Classification

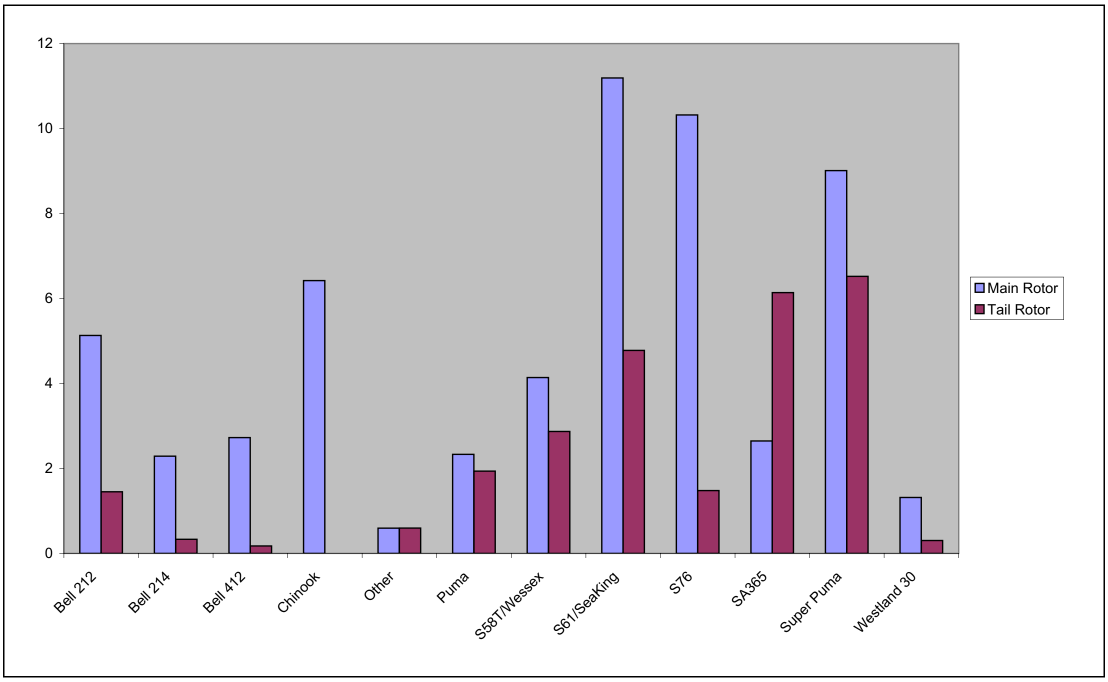

Figure 10 Benefit Rating by Aircraft

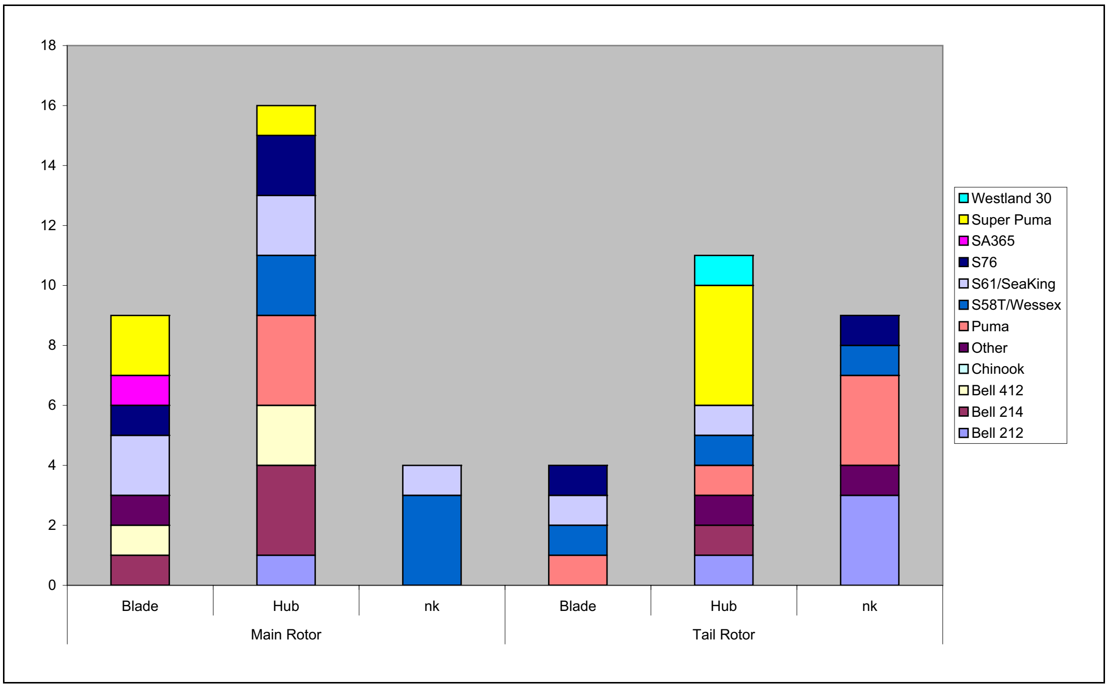

Figure 11 Cause of Accidents and Serious Incidents (Where Known)

INTENTIONALLY LEFT BLANK