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AERONAUTICAL DESIGN STANDARD

HANDBOOK

FOR

CONDITION BASED MAINTENANCE SYSTEMS

FOR US ARMY AIRCRAFT SYSTEMS

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CONDITION BASED MAINTENANCE SYSTEMS

FOR US ARMY AIRCRAFT SYSTEMS

UNITED STATES ARMY AVIATION AND MISSILE COMMAND

AVIATION ENGINEERING DIRECTORATE

REDSTONE ARSENAL, ALABAMA

FUNCTIONAL DIVISION: _____
Christopher Smith
AMCOM Director,
Condition Based Maintenance

SUBMITTED BY: _____
Dr. William D. Lewis
Director, Aviation
Engineering Directorate

APPROVED BY: _____
Randy Harkins
AMCOM and PEO Aviation
Standards Executive

FOREWORD

1. This document is approved for use by the US Army Research, Development, and Engineering Command, Aviation Engineering Directorate and is available for use by all agencies and elements of the Department of Defense.

2. This Handbook describes the Army's Condition Based Maintenance (CBM) system and defines the overall guidance necessary to achieve CBM goals for Army aircraft systems and Unmanned Aerial Systems (UAS). The Handbook contains some proven methods to achieve CBM functional objectives, but these suggested methods should not be considered to be the sole means to achieve these objectives. The Handbook is intended for use by:

    a. Aircraft life cycle management personnel defining guidance for CBM implementation in existing or new acquisition programs. This Handbook should be used as a foundation for program specific guidance for CBM to ensure that the resulting program meets Army requirements for sustained airworthiness through maintenance methods and logistics systems.

    b. Contractors incorporating CBM into existing or new acquisition programs for Army aircraft system equipment. In most cases, a CBM Management Plan should be submitted to the Government as part of the Statement of Work (SOW) for the acquisition, as required by the Request for Proposal (RFP) or Contract. The management plan should apply to aircraft systems, subsystems and the basic aircraft. The management plan will outline the contractor's proposed methods for achieving CBM goals listed in the RFP and the management control actions which will guide implementation.

3. This document provides guidance and standards to be used in development of the data, software and equipment to support CBM for systems, subsystems and components of US Army aircraft systems and, in the future, UAS. The purpose of CBM is to take maintenance action on equipment where there is evidence of a need. Maintenance guidance are based on the condition or status of the equipment instead of specified calendar or time based limits such as Component Retirement Time while not increasing the system baseline risk. This Design Handbook accomplishes that goal by describing elements that enable the issue of CBM Credits, or modified inspection and removal criteria of components based on measured condition and actual usage. This adjustment applies to either legacy systems with retro-fitted and validated CBM systems as well as new systems developed with CBM as initial design requirements. These adjustments can either decrease or increase the components installed life, depending on the severity of operational use and the detection of faults.

4. Comments, suggestions, or questions on this document should be addressed to Commander, US Army Research, Development and Engineering Command, Aviation and Missile Research, Development and Engineering Center, RDMR-AE, Huntsville, AL 35898. Since contact information can change, verification of the currency of this address information using the ASSIST online database at http://assist.daps.dla.mil/online/start/ is important.

5. Specific technical questions should be addressed to the following office:

   US Army Aviation and Missile Research, Development and Engineering Center
   Redstone Arsenal RDMR-AE
   Building 4488, Room B218
   Redstone Arsenal, AL 35898-5000
   Telephone: Commercial (256) 313-8996

Table of Contents

1. SCOPE 1
1.1 Scope 1
2. APPLICABLE DOCUMENTS 2
2.1 General. 2
2.2 Government Documents. 2
2.3 Other Government documents, drawings, and publications. 2
2.4 Non-Government publications 2
2.5 Other Government and Non-Government guidance documents. 3
3. DEFINITIONS 4
4. GENERAL GUIDANCE 7
4.1 Background. 7
4.2 General Guidance 8
4.2.1 Embedded diagnostics. 9
4.2.2 Fatigue damage monitoring. 9
4.2.3 Regime recognition. 10
4.2.4 Fatigue damage remediation. 11
4.2.5 Ground based equipment and information technology. 11
5. SPECIFIC GUIDANCE 12
5.1 External systems.. 13
5.2 Technical displays and technical and information presentation.. 13
5.3 Data acquisition (da). 13
5.3.1 Digital source collector (DSC) data collection (Data Size). 13
5.4 Data manipulation (DM). 14
5.5 State detection (SD). 15
5.6 Health assessment (HA). 16
5.7 Prognostics assessment (PA). 17
5.8 Advisory generation (AG). T 17
5.9 Guidelines for modifying maintenance intervals. 17
5.9.1Modifying overhaul intervals 18
5.9.2Transitioning to on-condition. 18
5.9.2.1Seeded fault testing.. 19
5.9.2.2 Field fault analysis. 19
5.9.3Statistical considerations 19
5.10 CBM Management Plan 20
6. HOW TO USE THIS ADS 20
6.1 Legacy aircraft. 21
6.2 New developmental systems. 23

APPENDIX A: Fatigue Life Management 26
A.1 SCOPE 27
A.1.1 Purpose 27
A.2 REFERENCES 27
A.3 DEFINITIONS 27
A.4 INTRODUCTION 27
A.5 POTENTIAL APPLICATIONS 29
A.5.1 Updating design usage spectrums 29
A.5.2 Managing service life of CSI components. 29
A.5.3 Remediation 30
A.5.4 Managing service life of damage tolerant structure 31
A.5.5 Maximizing FLM benefits. 32
A.6 RELIABILITY GUIDANCE 32
A.6.1 Reliability Analysis 33
A.6.2 Evaluation of reliability when usages are monitored and fatigue strength and flight loads are statistically evaluated 33
A.6.3 Evaluation of reliability when loads are monitored and fatigue statistically modeled. 33
A.6.4 Evaluation of reliability when usages are monitored usage and design damages applied. 33
A.6.5 Specific reliability guidance for unmanned aircraft systems 334
APPENDIX B: Regime Recognition/Flight State Classification with Validation of Regime Recognition Algorithms 35
B.1 SCOPE 36
B.1.1 Scope. 36
B.2 REFERENCES AND APPLICABLE DOCUMENTS 36
B.2.1 References. 36
B.2.2 Applicable Documents. 36
B.2.2.1 Government documents 36
B.2.2.2 Other Government documents, drawings, and publications. 37
B.3 DEFINITIONS 37
B.4 GENERAL GUIDANCE 37
B.5 SPECIFIC GUIDANCE 38
B.5.1 Flight regime definition. 38
B.5.1.1 Aircraft configuration. 38
B.5.1.2 Flight environment. 39
B.5.2 CBM instrumentation design 40
B.5.2.1Onboardflightstatesensing. 40
B.5.2.2 Flight state sampling rate 40
B.5.2.3 Classification of flight regimes. 42
B.5.2.4 Component lifecycle tracking 43
B.5.2.5 Data compromise recovery 43
B.5.3 CBM Instrumentation validation 43

B.5.3.1 Algorithm validation methodology. 44
B.5.3.2 Accuracy 44
B.5.4 Validation of structural usage monitoring system (SUMS). 44
B.5.4.1 Introduction. 45
B.5.4.2 Development of the structural monitoring system. 46
B.5.4.3 Design of the structural monitoring system. 46
B.5.4.4 Parameter identification and algorithms development. 46
B.5.4.5 Scripted flights.. 47
B.5.4.6 Unscripted flights.. 47
B.5.4.7 Flight testing. 48
B.5.4.8 Comparison of loads 48
B.5.4.9 Comparison of damage fraction... 48
APPENDIX C: Minimum Guidance for Determining CIs/HIs 49
C.1 SCOPE 50
C.2 REFERENCES AND APPLICABLE DOCUMENTS 50
C.2.1 References. 50
C.2.2. Applicable documents 50
C.2.2.1 Government documents 50
C.2.3 Definitions 50
C.2.4 Process description. 50
C.3 PROCESS GUIDANCE 51
C.4 GENERAL GUIDANCE 56
C.4.1 Condition indicator (CI) and health indicator (HI) behavior. 56
C.4.2 Health Indicator (HI) Usage. 58
C.5 APPROACH: CI/HI DEVELOPMENT FOR LEGACY AIRCRAFT 58
C.5.1 Initial situation 58
C.5.2 CI development process 59
C.5.2.1 Understand the failure mode.. 59
C.5.2.2 Determine the best means of measurement.. 62
C.5.2.3 Determine the existing system capabilities.. 62
C.5.2.4 Identify candidate feature extraction/CI algorithms 62
C.5.2.5 Obtain data to train & evaluate the CI 63
C.5.2.6 Code the algorithms & test performance 64
C.6 APPROACH: CI/HI DEVELOPMENT FOR NEW DEVELOPMENTAL AIRCRAFT 66
C.6.1 Initial situation 66
C.6.2 CI development process 66
C.6.2.1 Understand the failure mode. 66
C.6.2.2 Determine the best means of measurement.. 67
C.6.2.3 Determine the design system capabilities. 67
C.6.2.4 Identify candidate feature extraction/CI algorithms 67
C.6.2.5 Obtain Data to Train & Evaluate the CI 68

C.6.2.6 Code the algorithms & test performance 68
APPENDIX D: Vibration Based Diagnostics 77
D.1 SCOPE 78
D.2 REFERENCES AND APPLICABLE DOCUMENTS 78
D.3 TECHNICAL GUIDANCE 79
D.3.1 Sensor guidance 79
D.3.1.1 Sensitivity 79
D.3.1.2 Dynamic Range 80
D.3.1.3 Linearity 80
D.3.1.4 Drift 80
D.3.1.5 Bandwidth 80
D.3.1.6 Installation 81
D.3.1.7 Built-in test capability 81
D.3.2 Data acquisition and signal processing guidance 81
D.3.2.1 Data acquisition conditions 81
D.3.2.2 Data acquisition frequency 82
D.3.2.3 Analog to digital conversion 82
D.3.2.4 Sampling rate 82
D.3.2.5 Data windowing 83
D.3.3 Diagnostic algorithm guidance 83
D.3.3.1 Computational efficiency 83
D.3.3.2 Physical description 83
D.3.3.3 Confidence level 84
D.3.3.4 Algorithm validation 84
D.3.3.5 False positive rate 85
D.3.3.6 False Negative Rate 85
D.3.3.7 Fault Isolation Rate 85
D.3.3.8 Software Development 85
D.3.3.9 Recommended Maintenance Actions 85
D.3.4 Prognostic Algorithm Guidance 85
D.3.4.1 Predictability 85
D.3.4.2 Time Horizon Guidance 86
D.4 EXISTING VIBRATION BASED DIAGNOSTICS 86
D.4.1 Shaft Condition Indicators 86
D.4.2 Shaft Balancing and Rotor Smoothing 86
D.4.2.1 Shaft Balance Techniques 86
D.4.2.2 Rotor Smoothing Techniques 87
D.4.3 Bearing Condition Indicators 87
D.4.4 Gear Condition Indicators 88
APPENDIX E: Data Integrity 89
E.1 SCOPE 90

E.2 APPLICABLE DOCUMENTS.. 90
E.3 DEFINITIONS 90
E.3.1 Data Availability. 90
E.3.2 End-to-end.. 90
E.3.3 Data security. 90
E.3.4 Data reliability.. 90
E.3.5 Data integrity. 91
E.3.6 Data verification.. 91
E.3.7 Data reduction.. 91
E.3.8 Data mining.. 91
E.4 GENERAL GUIDANCE 91
E.5 SPECIFIC GUIDANCE 92
E.5.1 Criticality. 92
E.5.2 Data acquisition.. 93
E.5.3 Data computation. 94
E.5.4 Data transmission.. 94
E.5.5 Data storage.. 94
E.5.6 Security. 95
E.5.7 Data retrieval.. 95
E.5.8 Data mining.. 95
E.5.9 Data error correction and notification.. 96
APPENDIX F: Seeded Fault Testing 97
F.1 SCOPE 98
F.2 APPLICABLE DOCUMENTS. 98
F.3 DEFINITIONS 98
F.4 GENERAL GUIDANCE 99
F.5 SPECIFIC GUIDANCE 103
APPENDIX G: Acronyms 134
G.1 ACRONYMS 135

1. SCOPE

This document, an Aeronautical Design Standard (ADS) Handbook (HDBK), provides guidance and defines standard practices for the design assessment and testing of all elements of a Condition Based Maintenance (CBM) system, including analytical methods, sensors, data acquisition (DA) hardware, signal processing software, and data management standards necessary to support the use of CBM as the maintenance approach to sustain and maintain systems, subsystems, and components of Army aircraft systems. This includes the process of defining CBM Credits (modified inspection and removal criteria of components based on measured condition and actual usage) resulting from CBM implementation as well as Airworthiness Credits. The document is organized with its main body associated with general overarching guidance, and appendices governing more specific guidance arising from application of technical processes.

There are four goals in the implementation of CBM:

a. Reducing burdensome maintenance tasks currently required to assure continued airworthiness
b. Increasing aircraft availability
c. Improving flight safety
d. Reducing sustainment costs

Any changes to maintenance practices identified to meet these goals should be technically reviewed to ensure there has been no adverse impact to baseline risk. This document provides specific technical guidance for the CBM to ensure the resulting CBM system is effective and poses no greater risk than the original baseline design.

The functional guidance for a CBM system is intended to include:

a. Engine monitoring
b. Dynamic system component monitoring
c. Structural monitoring
d. Exceedance recording
e. Usage monitoring
f. Electronic logbook interface
g. Electronics

These functional capabilities are intended to implement CBM on all Army aircraft systems. Future revisions will include Unmanned Aerial Systems (UAS).

2. APPLICABLE DOCUMENTS

2.1 General. The documents listed below are not necessarily all of the documents referenced herein, but are those helpful in understanding the information provided by this handbook.

2.2 Government Documents. The following specifications, standards, and handbooks form a part of this document to the extent specified herein.

a. MIL-STD-1553B. Digital Time Division Command/Response Multiplex Data Bus

b. MIL-STD-1760E. Aircraft/Store Electrical Interconnection System

(Copies of these documents are available online at http://assist.daps.dla.mil/quicksearch/_or from the Standardization Document Order Desk, 700 Robbins Avenue, Building 4D, Philadelphia, PA 19111-5094.)

2.3 Other Government documents, drawings, and publications. The following other Government documents, drawings, and publications form a part of this document to the extent specified herein.

a. Army Regulation 70-62. "Airworthiness Qualification of Aircraft Systems." 21 May 2007.

b. Army Regulation 750-43 - Army Test, Measurement, and Diagnostic Equipment

Copies of these documents are available online at
http://www.army.mil/usapa/epubs/pdf/r70_62.pdf;
http://www.dtic.mil/whs/directives/corres/pdf/415122p.pdf
http://www.apd.army.mil/USAPA_PUB_pubrange_P.asp?valueAD=Pam+-+DA+Pamphlet

2.4 Non-Government publications. The following other Government documents, drawings, and publications form a part of this document to the extent specified herein."

a. ARINC-429. Avionics Bus Interface

b. IEEE 802.3 Standard for Information Technology Wireless Local Area Network

c. IEEE 802.11. Wireless Local Area Network

d. IEEE 802.15 Wireless Personal Area Networks (WPAN)

e. ISO 11898-1:2003. Controller Area Network (CAN)

f. ISO 13374:2003. Condition Monitoring and Diagnostics of Machines.

g. ISO 9001:2000. Certified Organization

h. MIMOSA Open Systems Architecture for Condition Based Maintenance, v3.2.

i. Felker, Douglas. "PM/FM Matrix & CBM Gap Analysis in Reliability Centered Maintenance." Presented to the 2006 DoD Maintenance Symposium.

j. Canaday, Henry. "Hunting for Productivity Gains." Aviation Week and Space Technology. September 10, 2004.

k. RTCA DO-178B. Software Considerations in Airborne Systems and Equipment Certification.

l. RTCA DO-200A. Standards for Processing Aeronautical Data

Copies of these documents are available at http://www.iso.org/iso/iso_catalogue.htm; http://www.mimosa.org/; http://www.arinc.com/; http://standards.ieee.org/

2.5 Other Government and Non-Government guidance documents. The following documents should be used to compliment the guidance of this handbook.

a. Army Regulation 25-2. "Information Management: Information Assurance." 24 October 2007.

b. Army Regulation 750–1. "Army Materiel Maintenance Policy." 20 September 2007.

c. Army Regulation 750–43. "Army Test, Measurement, and Diagnostic Equipment." 3 November 2006.

d. Army Pamphlet 738–751. "Functional Users Manual for the Army Maintenance Management System—Aviation, (TAMMS-A)." 15 March 1999.

e. DoDI 4151.22. "Condition Based Maintenance Plus (CBM+) for Materiel Maintenance." Department of Defense Instruction Number 4151.22. 2 December 2007.

f. US Army CBM+ Roadmap. Revised Draft 20 July 2007.

g. US Army AMCOM Condition Base Maintenance (CBM) Systems Engineering Plan (SEP), Rev: Feb 2008. (Includes Sections 2.2 and 2.3 only.)

h. SAE Standard AS 5391A. Health and Usage Monitoring System Accelerometer Interface Specification.

i. SAE Standard AS 5392A. Health and Usage Monitoring System, Rotational System Indexing Sensor Specification.

j. SAE Standard AS5393. Health and Usage Monitoring System, Blade Tracker Interface Specification.

k. SAE Standard AS5394. Health and Usage Monitoring System, Advanced Multipoint Interface Specification.

l. SAE Standard AS5395. Health and Usage Monitoring System, Data Interchange Specification.

m. STN 91-019 Apache Fatigue Substantiation

3. DEFINITIONS

Airworthiness: A demonstrated capability of an aircraft or aircraft subsystem or component to function satisfactorily when used and maintained within prescribed limits (Ref AR 70-62).

Airworthiness Credit: The sustainment or reduction of baseline risk in allowance for a CBM Credit, based on the use of a validated and approved CBM system. The change can be specific to a specific item (component or part), tail number of an aircraft, or any group of items or aircraft as defined in the respective Airworthiness Release (AWR).

Baseline Risk: The acceptable risk in production, operations, and maintenance procedures reflected in frozen planning, the Operator's Manuals, and the Maintenance Manuals for that aircraft. Maintenance procedures include all required condition inspections with intervals, retirement times, and Time Between Overhauls (TBOs).

CBM Credit: The approval of any change to the maintenance specified for a specific end item or component, such as an extension or reduction in inspection intervals or Component Retirement Time established for the baseline system prior to incorporation of CBM as the approved maintenance approach. (For example, a legacy aircraft with a 2,000 Component Retirement Time) CRT for a drive system component can establish a change to the CRT for an installed component for which CBM CI values remain below specified limits and the unit remains installed on a monitored aircraft.) Often, CBM Credits may be authorized through an Airworthiness Release (AWR).

Condition Based Maintenance: The application and integration of appropriate processes, technologies, and knowledge based capabilities to improve the reliability and maintenance effectiveness of Army Aircraft Systems and components. Uses a systems engineering approach to collect data, enable analysis, and support the decision-making processes for system acquisition, sustainment, and operations.

Confidence Bound: An endpoint of a confidence interval.

Confidence Interval: An interval constructed from random sampling that, with known probability, contains the true value of a population parameter of interest.

Confidence Level: The probability that a confidence interval contains the true value of a population parameter of interest. When not otherwise specified in this ADS, the confidence level shall be assumed to equal 0.9 (or 90%).

Credible Failure Mode: The believable manner in which a system or component may go beyond a limit state and cause a loss of function and secondary damage, or loss of function or secondary damage as supported by: engineering tests, probabilistic risk analysis, and actual occurrences of failures, or actual occurrences of failure.

Critical Failure Mode: The mechanism that leads a system or component to go beyond a limit state and causes a loss of function and secondary damage, or loss of function or secondary damage. System or component criticality is determined by criticality analysis in relation to its impact on system/component operation and environment, or system/component or environment.

Digital Source Collector: An onboard aircraft data recording system used to collect CBM data

False Positive: Failure mode is detected but not found by inspection; condition does not match recorded CI level (yellow or red CI = healthy component)

False Negative: Failure mode is not detected but is found to exist by inspection; condition does not match recorded CI level (green CI = faulty component)

Ground Air Ground Cycles: Relatively low-frequency large-amplitude load cycles occurring during a given flight, but not present in any single flight condition. Examples include rotor start and stop cycles and load fluctuations between the various flight conditions encountered during performance of a mission.

Health Indicator (HI): An indicator for needed maintenance action resulting from the combination of one or more CI values.

Health Monitoring: Equipment, techniques or procedures by which selected incipient failure or degradation can be determined.

Legacy Aircraft: An aircraft in an operational unit that has passed its scheduled IOC (initial operational capability).

Loads Monitoring: Equipment, techniques and procedures, or equipment, techniques or procedures to measure and calculate or procedures to measure or calculate procedures to measure the loads (forces or moments) experienced by an aircraft component during operational flight.

Regimes: Combinations of weight, altitude, C.G. and maneuvers that describe typical aircraft usage.

Reliability: The probability that a functional unit will perform its required function for a specified interval under stated conditions.

Remaining Useful Life (RUL): An estimate of failure free operation of the described component or system.

Top of Scatter: Flight load records and summary data, or flight load records or summary data which produce the highest fatigue damage for a given regime or load cycle when used in accordance with a given fatigue methodology.

Standard Deviation: A measure of the amount by which measurements deviate from their mean.

Structural Usage Monitoring: Managing fatigue lives via Usage Monitoring

True Positive: Failure mode is detected with condition verified by inspection and matching recorded CI level (yellow or red CI = faulty component).

True Negative: Failure mode is not detected with condition verified by inspection and matching recorded CI level (green CI = healthy component)

Usage Monitoring: Equipment, techniques and procedures or equipment, techniques or procedures which selected aspects of service [flight] history can be determined.

Validation: The process of evaluating a system or software component during, or at the end of, the development process to determine whether it satisfies specified requirements

Verification: Confirms that a system element meets design-to or build-to specifications. Throughout the systems life cycle, design solutions at all levels of the physical architecture are verified through a cost-effective combination of analysis, examination, demonstration, and testing, all of which can be aided by modeling and simulation.

4. GENERAL GUIDANCE

4.1 Background. Department of Defense (DoD) policy on maintenance of aviation equipment has employed Reliability Centered Maintenance (RCM) analysis and methods to avoid the consequences of material failure. The structured processes of RCM have been part of army aviation for decades. RCM analysis provides a basis for developing requirements for CBM through a process known as "Gap Analysis."[1]

Condition Based Maintenance (CBM) is a set of maintenance processes and capabilities derived primarily from real-time assessment of system condition obtained from embedded sensors and external test and measurements using portable equipment or embedded sensors or external test and measurements using portable equipment. CBM is dependent on the collection of data from sensors and the processing, analysis, and correlation of that data to material conditions that require maintenance actions. Maintenance actions are essential to the sustainment of material to standards that insure continued airworthiness.

Data provide the essential core of CBM, so standards and decisions regarding data and their collection, transmission, storage, and processing dominate the requirements for CBM system development. CBM has global reach and multi-systems breadth, applying to everything from fixed industrial equipment to air and ground vehicles of all types. This breadth and scope has motivated the development of an international overarching standard for CBM. The standard, known as ISO 13374:2003, "Condition Monitoring and Diagnostics of Machines," provides the framework for CBM.

This handbook is supported by the Machinery Information Management Open Standards Alliance (MIMOSA), a United States organization of industry and Government, and published as the MIMOSA Open Systems Architecture for Condition Based Maintenance (OSA CBM) v3.2. The standard is embodied in the requirements for CBM found in the Common Logistics Operating Environment (CLOE) component of the Army's information architecture for the Future Logistics Enterprise. The ISO standard, the OSA CBM standard, and CLOE all adopt the framework shown in FIGURE 1 for the information flow supporting CBM with data flowing from bottom to top. This document, however, considers the application of CBM only to Army aircraft systems and Unmanned Aerial Vehicles).

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[1] Felker, Douglas, "PM/FM Matrix & CBM Gap Analysis in Reliability Centered Maintenance," presented to the 2006 DoD Maintenance Symposium.

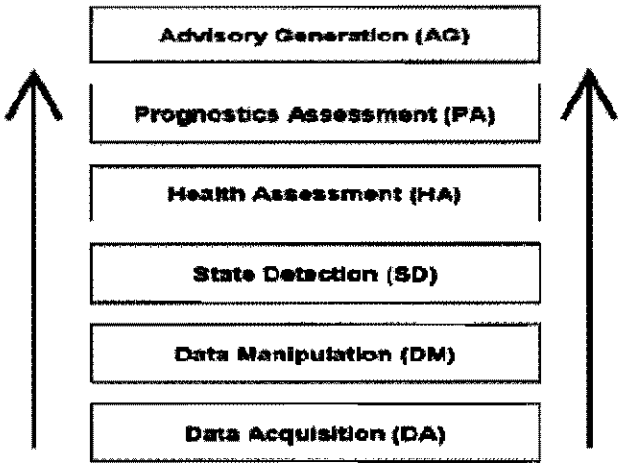

FIGURE 1: ISO-13374 Defined data processing and information flow

4.2 General Guidance. CBM practice is enabled through three basic methodologies:

a. Embedded diagnostics for components that have specific detectable faults (example, drive systems components with fault indicators derived from vibratory signature changes and sensors acceptable for tracking corrosion damage).

b. Usage monitoring, which may derive the need for maintenance based on parameters such as the number of power-on cycles, the time accumulated above a specific parameter value or the number of discrete events accumulate. Within this context, specific guidance is provided where benefits can be derived.

c. Fatigue life management, through estimating the effect of specific usage in flight states that incur fatigue damage as determined through fatigue testing, modeling, and simulation.

In the context of data management on the platform, every effort should be made to conform to existing vehicle architectures and common military standards for data acquisition and collection. Military vehicles typically use MIL-STD-1553B, Digital Time Division Command/Response Multiplex Data Bus[2], for sending multiple data streams to vehicle processors. As the use of commercial off-the-shelf (COTS) hardware and software has become more prevalent, the use of commercial standards for data transfer may be acceptable as design standards for CBM in aviation systems.

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[2] MIL-STD-1553B. Digital Time Division Command/Response Multiplex Data Bus. 15 January 1996. See also: <http://assist.daps.dla.mil/quicksearch/basic_profile.cfm?ident_number=36973>.

4.2.1 Embedded diagnostics. Health and Usage Monitoring Systems (HUMS) have evolved over the past several decades in parallel with the concepts of CBM. They have expanded from measuring the usage of the systems (time, flight parameters, and sampling of performance indicators such as temperature and pressure) to forms of fault detection through signal processing. The signal processing typically recorded instances of operation beyond prescribed limits (known as "exceedances"), which then could be used as inputs to troubleshooting or inspection actions to restore system operation. This combination of sensors and signal processing (known as "embedded diagnostics") represents a capability to provide the item's condition and need for maintenance action. When this capability is extended to CBM functionality (state detection and prognosis assessment), it should have the following general characteristics:

a. Sensor Technology: Sensors should have high reliability and high accuracy and precision. There is no intent for recurring calibration of these sensors.

b. Data Acquisition: Onboard data acquisition hardware should have high reliability and accurate data transfer (See Appendix E).

c. Algorithms: Fault detection algorithms are applied to the basic acquired data to provide condition and health indicators, or condition or health indicators. Validation and verification of the Condition Indicators (CIs) and Health Indicators (HIs) included in the CBM system are required in order to establish maintenance and airworthiness credits. Basic properties of the algorithms are: (1) sensitivity to faulted condition, and (2) insensitivity to conditions other than faults. The algorithms and methodology should demonstrate the ability to account for exceedances, missing or invalid data.

HUMS operation during flight is essential to gathering data for CBM system use, but is not flight critical or mission critical when it is an independent system which obtains data from primary aircraft systems and subsystems. When this independence exists, the system should be maintained and repaired as soon as practical to avoid significant data loss and degradation of CBM benefits. As technology advances, system design may lead to more comprehensive integration of HUMS with mission systems. The extent of that future integration may lead to HUMS being part of mission or flight critical equipment or software. In this case, the HUMS bear the same priority as mission or flight critical equipment relative to the requirement to restore its proper operation.

4.2.2 Fatigue damage monitoring. Fatigue damage is estimated through calculations which use loads on airframe components experienced during flight. These loads are dependent on environmental conditions (example, temperature and altitude) aircraft configuration parameters (examples: gross weight (GW), center of gravity (CG)), and aircraft state parameters related to maneuvering (i.e.: air speed, aircraft attitudes, power applied, and accelerations). To establish these loads, algorithms known as regime recognition algorithms, are used to take these parameters and map them to known aircraft maneuvers for which representative flight loads are available from loads surveys. In order to establish regime recognition algorithms as the basis for

loads and fatigue life adjustment, the algorithms should be validated through flight testing. Specific guidance for validation of regime recognition algorithms is contained in Appendix B.

Legacy aircraft operating without CBM capabilities typically use assumed usage and Safe Life calculation techniques to ensure airworthiness. Structural loading of the aircraft in flight, including instances which are beyond prescribed limits (i.e.: exceedances) for the aircraft or its components on legacy platforms typically use a rudimentary sensor or data from a cockpit display with required post-flight inspection as the means to assess damage. The advent of data collection from sensors onboard the aircraft, typically performed onboard an aircraft by a Digital Source Collector (DSC) enable methods that improve accuracy of the previous detection and assessment methods. The improvement is due to the use of actual usage or measured loads rather than calculations based on assumptions made during the developmental design phase of the acquisition.

4.2.3 Regime recognition (actual usage detection and measurement). Accurate detection and measurement of flight regimes experienced by the aircraft over time enable two levels of refinement for fatigue damage management: (1) the baseline "worst case" usage spectrum can be refined over time as the actual mission profiles and mission usage can be compared to the original design assumptions, and (2) running damage assessment estimates can be based on specific aircraft flight history instead of the baseline "worst case" for the total aircraft population. Perform running damage assessment estimates for specific aircraft components will require data management infrastructure that can relate aircraft regime recognition and flight history data to individual components and items which are tracked by serial number. Knowledge of the actual aircraft usage can be used to refine the baseline 'worst case' usage spectrum used to determine the aircraft service schedules and component retirement times. The refinement of the "worst case" usage spectrum, depending on actual usage, could result in improved safety and reduced cost, or improved safety or reduced cost. The criteria for acceptance of airworthiness credits from a fatigue life management perspective are provided in Appendix A.

The refined usage spectrum enables refining fleet component service lives to account for global changes in usage of the aircraft. The usage spectrum may be refined for specific periods of operation. An example is refining the usage spectrum to account for the operation of a segment of the fleet in countries where the mean altitude, temperature, or exposure to hazards can be characterized. The use of DSC data to establish an updated baseline usage spectrum is the preferred method (compared with pilot survey method).

The running damage assessment is dependent on specific systems to track usage by part serial numbers. In this case, the logistics system should be capable of tracking the specific part (by serial number) and the specific aircraft (by tail number). The actual usage of the part, and its Remaining Useful Life, can be determined from the usage data of the aircraft (tail numbers) for the part (serial numbers). Because usage monitoring and component part tracking are not flight critical systems, if either of these systems fail, the alternative is to apply the most current design usage spectrum and the associated fatigue methodology for any period of flight time in which the usage monitor data or the part tracking data is not available. As such, use of the running damage assessment method does not eliminate the need to periodically refine the fleet usage spectrum based on use of DSC data. Specifics for the implementation of the running damage assessment

are given in Appendix B: Regime Recognition/Flight State Classification with Validation of Regime Recognition Algorithms, and Appendix A: Fatigue Life Management.

4.2.4 Fatigue damage remediation. Remediation may be used to address components that are found to be routinely removed from service without reaching the fatigue safe life (a.k.a. component retirement time, CRT). The process of remediation involves the identification of removal causes that most frequently occur. Often the cause of early removal is damage such as nicks, dings, scratches or wear. Details for implementation of remediation are found in Appendix A. When remediation action is taken to increase repair limits, it should be documented in maintenance manuals, including Technical Manuals (TMs) and Depot Maintenance Work Requirements (DMWRs).

4.2.5 Ground based equipment and information technology. The use of data to modify maintenance practice is the heart of CBM. As such, the ground based equipment that is used to complete the data processing, analysis of sensor data, infer components integrity, forecast remaining useful life, and decide appropriate maintenance actions, is a vital part of the CBM system. The CBM data architecture and ground based equipment used to interface with the data should be capable of supporting several types of management actions that support optimal maintenance scheduling and execution:

a. Granting CBM credits (changes to scheduled maintenance) based on usage/loads monitoring, damage accrual or CI/HI values, and requires accurate configuration management of components and parts installed on the aircraft.

b. Ordering parts, based on exceeded CI/HI thresholds that indicate the presence of a fault, requires an interface of the data from the ground based equipment through STandard Army Management Information System (STAMIS), Standard Army Retail Supply System (SARSS), and Unit Level Logistics System-A (ULLS-A). This interface should be accomplished to eliminate the need for duplicative data entry. The ground based equipment should enable monitoring of CI/HIs and using the predetermined "thresholds" or CI/HI values to allow for anticipatory supply actions, optimizing maintenance planning, and enhancing safety by avoiding a precautionary landing/recovery/launch.

c. Modifying the CRT based on running damage assessment for a specific serialized component will require automated changes to be recorded in STAMIS record system.

d. Configuration Management of the Monitoring System should enable the following items to be displayed on any data output:

    i. The date, drawing number revision, and software version of the monitoring hardware/software

    ii. Any controlled changes to hardware/software configuration items of the monitoring system

iii. Compliance with any applicable safety of flight messages and aviation safety action messages

iv. A list of software versions, part numbers, and respective serial numbers being monitored.

For Army aircraft systems, tracking of individual serialized items begins at the time of manufacture through its life cycle and is accomplished by either manual records and an electronic log book, or either manual records or electronic log book which is an integral part of the STAMIS architecture. CBM credits can be given to groups of aircraft or parts, as long as they can be tracked. CBM credits cannot be applied to individual items based on running damage assessment estimates without accurate tracking of an individual part's installation and maintenance history as reflected in the electronic log book and other records.

While one of the objectives of CBM is to provide complete visibility of the operational history of a serialized component, the Army's current maintenance information systems do not have the capability to meet this objective. Shortfalls include:

a. Lack of quality control tools in the current system allow for duplicate entries, typographical errors, and erroneous entries.

b. Data requirements (scope, data size, and analysis requirements) for this effort have yet to be defined, which creates uncertainty and risk in defining the Data Storage, Analysis and Transmission capabilities required.

c. Software inoperability to calculate and manage varying usage rates (flight hours) based on operational history.

5. SPECIFIC GUIDANCE

Specific guidance for the CBM system is grouped by the functionality shown in FIGURE 1, to link the guidance to the overarching International Standards Organization (ISO) and DA architecture for CBM. Sections below briefly describe the elements of the CBM system architecture and link those elements to specific technical considerations for Army Aviation. To enable these technical considerations to be easily refined as CBM implementation matures, the technical considerations are grouped into six separate Appendices.

These appendices set forth acceptable means, but not the only means, of compliance with CBM detailed technical elements. They are offered in the concept of a Federal Aviation Administration (FAA) Advisory Circular. They include:

a. Appendix A: Fatigue Life Management

b. Appendix B: Regime Recognition/Flight State Classification with Validation of Regime Recognition Algorithms

c. Appendix C: Minimum Guidance for Determining CIs/HIs

d. Appendix D: Vibration Based Diagnostics

e. Appendix E: Data Integrity

f. Appendix F: Seeded Fault Testing

5.1 External systems. External system data guidance is defined by various STandard Army Management Information Systems (STAMIS). Any system designed to enable CBM on an Army platform should follow the guidance set for these systems.

5.2 Technical displays and technical and information presentation. Technical displays and information presentation to support CBM should be accredited and certified for compatibility with software operating systems. These systems are defined by Logistics Information Systems (LIS) for desktop systems that include other current standards for portable maintenance aids or Interactive Electronic Technical Manuals (IETMs).

5.3 Data acquisition (da). Data acquisition standards for collecting and converting sensor input to a digital parameter are common for specific classes of sensors (examples: vibration, temperature, and pressure sensors). The same standards extant for this purpose remain valid for CBM application, but with a few exceptions. In many cases, data from existing sensors on the aircraft are sufficient for CBM failure modes. Some failure modes, such as corrosion, may require new sensors or sensing strategies to benefit CBM. In all cases, certain guidance should be emphasized:

a. Flight State Parameters: Accuracy and sampling rates should be commensurate to effectively determine flight condition (regime) continuously during flight. The intent of these parameters is to unambiguously recreate that aircraft state post-flight for multiple purposes (example: duration of exposure to fatigue damaging states) (See Appendix A and B for additional guidance).

b. Vibration: Sampling rates for sensors on operational platforms should be commensurate for effective signal processing and "de-noising." Vibration transducer placement and mounting effects should be validated during development testing to ensure optimum location. (See Appendix D for additional description of other guidance).

c. System-Specific: Unique guidance to sense the presence of faults in avionics and propulsion system components are in development and will be addressed in subsequent versions of this ADS. Similarly, the promise of technology to sense corrosion-related damage in the airframe may mature to the point where detection with high confidence is included in the scope of this ADS at a later date.

5.3.1 Digital source collector (DSC) data collection (Data Size). Data storage and transmittal are significant design issues. On-board data storage and the capability to transfer flight data to the ground station are determined by the capabilities of the DSC and the ground station. Recognizing that these capabilities will change over time, it is desirable for the DSC software to have the flexibility to change the parameters and collection rates as the transmission

and storage capabilities improve, or change the parameters or collection rates as the transmission and storage capabilities improve. The potential exist for large amounts of aircraft usage data to be stored long term on board the aircraft and then downloaded, analyzed and stored periodically, (i.e. at phased maintenance). As a result, after each flight, it may be necessary to analyze and reduce the usage data on board the aircraft or at the ground station prior to data transmittal. Exceptions to these limitations are possible during the initial implementation/check-out phase of the DSC system.

Because usage monitoring is not a flight-critical function, the recording unit may not be serviced frequently enough to prevent the loss of data. The recorder should be sized to enable data storage to prevent data loss between downloads. The data recording and storage device, along with other HUMS components, should be repaired as soon as practical (even though they are not mission or flight critical), in order to prevent CBM system data degradation. The storage rate may be different from the sampling rate and still meet the needs for CBM.

However, consideration should be given to the practical limitations of data capture and storage. A balance should be found between the requirements for accurate condition sensing and the limitations of data transfers to and storage at the Top Tier level which is necessary in realizing a practical implementation. In general, these requirements can be specified separately according to: (1) on-aircraft; (2) ground station; (3) Top Tier data link; and (4) Web site. On-aircraft data storage is typically limited by the size and weight constraints of the platform operation concept as well as the bus bandwidth that services the data storage system. Ground station data handling is limited by the available storage hardware space and the need for reasonable operational transfer times from the aircraft to the offboard storage. Data transfer over the Top Tier level is limited by both satellite communication bandwidth and reasonable search technology constraints which limit file transmittal to approximately one megabyte of data per flight hour. Therefore, Top Tier data transfer should be limited to transmission of only processed CBM metrics and not raw, high-speed sampled sensor measurements. However, Web site archival storage should be sized to capture all collected data including unprocessed, sampled sensor measurements for later use in refining and developing new condition indicators. For specific guidance on the practical limits of data acquisition and handling with regard to Regime Recognition and Vibration refer to the discussion and tables found in Appendix B and D.

5.4 Data manipulation (DM). Data manipulation (also referred to as signal processing) should be governed by best practice throughout the data processing steps. Standardizing a specific set of practices is ineffective, as each application requires techniques best fitted to its particular needs. Each set of resultant files from raw data to de-noised data, data compression such as Synchronous Time Average (STA) and Fast Fourier Transform (FFT), feature or CI calculation, and state estimation should be linked to each other to demonstrate a "chain of custody" and also to indicate which set of algorithms were used. As CBM is a dynamic and evolutionary system, the outcome of fault detection and estimates of RUL is dependent upon the software modules used. Traceability of this software is essential for configuration management

and confidence in the result. Specific guidance for data integrity and data management as described in DO-178[3] and DO-200[4] are referenced in Data Integrity Appendix E.

5.5 State detection (SD). State Detection uses sensor data to determine a specific condition. The state can be "normal" or expected, an "anomaly" or undefined condition, or an "abnormal" condition. States can refer to the operation of a component or system, or the aircraft (examples, flight attitudes and regimes). An instance of observed parameters representing baseline or "normal" behavior should be maintained for comparison and detection of anomalies and abnormalities. Sections of the observed parameter data that contain abnormal readings which relate to the presence of faults should be retained for archive use in the knowledge base as well as for use in calculation of CIs in near real time.

The calculation of a CI should result in a unique measure of state. The processes governing CI and HI developments are:

a. Physics of Failure Analysis: This analysis determines the actual mechanism which creates the fault, which if left undetected can cause failure of the part or subsystem. In most cases, this analysis is to determine whether material failure is in the form of crack propagation or physical change (example: melting and embrittlement). This analysis determines the means to sense the presence of the fault and evolves the design decisions which place the right sensor and data collection to detect the fault.

b. Detection Algorithm Development (DAD): The process of detection algorithm development uses the Physics of Failure Analysis to initially select the time, frequency or other domain for processing the data received from the sensor. The development process uses physical and functional models to identify possible frequency ranges for data filtering and previously successful algorithms as a basis to begin development. Detection algorithms are completed when there is sufficient test or operational data to validate and verify their performance. At a minimum, systems underlying algorithms should provide a 95% confidence level in detection of incipient faults and also have no more than a 5% false alarm rate (indications of faults that are not present). Further details in are found in Appendix C.

c. Fault Validation/Seeded Fault Analysis: Detection Algorithms are tested to ensure that they are capable of detecting faults prior to operational deployment. A common method of fault validation is to create or to "seed" a fault in a new or overhauled unit and collect data on the fault's progression to failure in controlled testing (or "bench test") which simulates operational use. Data collected from this test are used as source data for the detection algorithm, and the algorithm's results are compared to actual item condition through direct measurement (see Appendix F).

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[3] RTCA DO-178B. Software Considerations in Airborne Systems and Equipment Certification.

[4] RTCA DO-200A. Standards for Processing Aeronautical Data

Anomaly detection should be able to identify instances where data are not within expected values and flag those instances for further review and root cause analysis. Such detection may not be able to isolate to a single fault condition (or failure mode) to eliminate ambiguity between components in the system, and may form the basis for subsequent additional data capture and testing to fully understand the source of the abnormality (also referred to as an "anomaly."). In some cases, the anomaly may be a CI reading that is created by maintenance error rather than the presence of material failure. For example, misalignment of a shaft by installation error could be sensed by an accelerometer, with a value close to a bearing or shaft fault.

Specific guidance for general CIs and HIs are found in Appendix C. Because many faults are discovered through vibration analysis, guidance for vibration-based diagnostics is found in Appendix D.

Operating state parameters (examples: gross weight, center of gravity, airspeed, ambient temperature, altitude, rotor speed, rate of climb, and normal acceleration) are used to determine the flight regime. The flight environment also greatly influences the RUL for many components. Regime recognition is essentially a form of State Detection, with the state being the vehicle's behavior and operating condition. Regime recognition is subject to similar criteria as CIs in that the regime should be mathematically definable and the flight regime should be a unique state for any instant, with an associated confidence boundary. The operating conditions (or regime) should be collected and correlated in time for the duration of flight for use in subsequent analysis. For specific guidance regarding regime recognition, refer to Appendix B.

For CIs that are sensitive to aircraft state or regime, maintenance threshold criteria should be applied in a specific flight regime to ensure consistent measurement and to minimize false alarms caused by transient behavior. Operating state parameters (examples: gross weight, center of gravity, airspeed, ambient temperature, altitude, rotor speed, rate of climb, and normal acceleration) are used to determine the flight regime. The flight environment also greatly influences the RUL for many components. Regime recognition is essentially a form of State Detection, with the state being the vehicle's behavior and operating condition. Regime recognition is subject to similar criteria as CIs in that the regime should be mathematically definable and the flight regime should be unique state for any instant, with an associated confidence boundary. The operating conditions (or regime) should be collected and correlated in time for the duration of flight for use in subsequent analysis. For specific guidance regarding regime recognition, refer to Appendix B.

5.6 Health assessment (HA). Using the existence of abnormalities defined in State Detection (SD) (Section 5.5), this function of the CBM system rates the current health of the equipment.

Health Indicator (HI): An indicator of the need for maintenance action resulting from the combination of one or more CI values.

Health assessment is accomplished by the development of HIs or indicators for maintenance action based on the results of one or more CIs. HIs should be indexed to a range of color-coded statuses such as: "normal operation", "prepare for maintenance" and "conduct when optimal for operations", and "required". Each fault should contribute to the determination of the overall

health of the aircraft. Status of the equipment should be collected and correlated with time for the condition during any operational cycle.

5.7 Prognostics assessment (PA). Using the description of the current health state and the associated failure modes, the PA module determines future health states and RUL. The estimate of RUL should use some representation of projected usage/loads as its basis. RUL estimates should be validated during system test and evaluation, and the estimates should show 90% or greater accuracy to the failures observed. For Army aviation CBM, the prognostics assessment is not required to be part of the onboard system.

The goal of the PA module is to provide data to the Advisory Generation (AG) module with sufficient time to enable effective response by the maintenance and logistics system. Because RUL for a given fault condition is based on the individual fault behavior as influenced by projected loads and operational use, there can be no single criteria for the lead time from fault detection to reaching the RUL. In all cases, the interval between fault detection and reaching the removal requirement threshold should be calculated in a way that provides the highest level of confidence in the RUL estimate without creating false positive rates higher than 5% at the time of component removal.

5.8 Advisory generation (AG). The goal of AG is to provide specific maintenance tasks or operational changes required to optimize the life of the equipment and allow continued operation. Using the information from the Health Assessment (Section 5.6) and Prognostics Assessment (Section 5.7) modules, the advisories generated for a CBM system should include:

a. provisions for denying operational use ("not safe for flight")

b. specific maintenance actions required to restore system operation

c. CBM credits for continued operation when the credits modify the interval to the next scheduled maintenance action.

The interval between download of data and health assessment is affected by operational use and tempo or conditions noted by the flight crew. Download is expected at the end of daily operations or at the end of the longest interval of continuous flight operations, whichever is greater.

Defining the basis for continued operation by limiting the qualified flight envelope or operating limitations is determined by the process of granting Airworthiness Credits. Since these limitations are situation dependent, analysis by Aviation Engineering Directorate (AED) staff engineers is normally required and considered outside the scope of the CBM system to provide through automated software.

5.9 Guidelines for modifying maintenance intervals. A robust and effective CBM system can provide a basis for modifying maintenance practices and intervals. As part of the continuous analysis of CBM data provided by the fielded systems and or seeded fault testing, disciplined review of scheduled maintenance intervals for servicing and inspection can be adjusted to increase availability and optimize maintenance cost. Similarly, the data can be used to modify the maximum Time Between Overhauls (TBO) for affected components. Finally,

CBM data can be used to transition away from current reactive maintenance practices to a proactive maintenance strategy in a manner that does not adversely impact the baseline risk associated with the aircraft's certification. For system reliability criteria refer to section A.6 in Appendix A.

5.9.1 Modifying overhaul intervals. In general, TBO interval extensions are limited by the calculated fatigue life of the component, unless the failure mode is detectable utilizing a reliable detection system and will not result in a component failure mode progressing or manifesting into a failed state within 2 data download intervals. A good example would be Hertzian Contact Fatigue Limit for bearings. Exceeding this limit would result in spalling, which is easily detected (through current methods or vibration monitoring) and also is associated with significant life remaining from the onset of spalling.

In the case of vibration monitoring, the capability of the monitoring system to accurately depict actual hardware condition should be verified prior to allowing incremental TBO increases. In addition, detailed analysis will be required to show fatigue life limits are not exceeded. Verification that CI's are representative of actual hardware condition will generally require a minimum of 5 detailed teardown inspections of the component to ensure commensurate confidence associated with the teardowns capturing the inherent variability that may occur with actual field usage. The results of these teardowns should confirm that the measured condition indicator value is representative of the actual hardware condition. Incremental TBO extensions should be limited to twice the current limit until such time the requirements of paragraph 5.9.2 are satisfied.

It is possible to obtain TBO extensions on unmonitored aircraft through hardware teardowns on components at or near their current TBO. To extend overhaul intervals on unmonitored aircraft, a compelling case must be developed with supporting detailed analysis, enhanced or special inspections, and field experience. Final approval of the airworthiness activity is required. The criticality of the component and all associated failure modes should also be taken into account. These factors will also impact the required number of satisfactory teardowns and associated TBO interval extensions. TBO increases may be used as a valuable tool for accumulating the data needed to show confidence level/reliability of a monitoring system in support of CBM programs.

5.9.2 Transitioning to on-condition. Prior to transition to On-Condition for legacy components/assemblies the requirements of 5.9.1 should be met. Guidelines for obtaining on-condition status for components on monitored aircraft having performed seeded fault testing versus data acquisition via field faults are outlined in paragraphs 5.9.2.1 and 5.9.2.2, respectively. Achieving on-condition status via field faults could take several years, therefore, incremental TBO extensions will be instrumental in increasing our chances of observing and detecting naturally occurring faults in the field. This also holds true for seeded fault selected components which have not completed all of the seeded fault test required to ensure each credible failure mode can be detected. Credible critical failure modes are determined through Failure Modes Effects Criticality Analysis (FMECA) and actual field data, or through FMECA or actual field data. Damage limits are to be defined for specific components in order to classify specific hardware condition to CI limit through the use of Reliability Improvement through Failure Identification and Reporting (RIMFIRE) or Structural Component Overhaul Repair Evaluation Category and Remediation Database (SCORECARD), Tear Down Analysis's (TDA),

2410 forms, and more. Implementation plans should be developed for each component clearly identifying goals, test requirements and schedule, initial CI limits, and all work that is planned to show how the confidence levels spelled out in paragraph 5.9.3 will be achieved.

5.9.2.1 Seeded fault testing. Seeded fault testing may dramatically reduce the timeline for achieving on-condition maintenance status because it requires less time to seed and test a faulted component than to wait for a naturally occurring fault in the field. However, if during the seeded fault test program a naturally occurring fault is observed and verified, it can be used as a data point to help reduce the required testing. Test plans will be developed, identifying each of the credible failure modes and corresponding seeded fault tests required to reliably show that each credible failure mode can be detected. The seeded fault test plan should include requirements for ensuring that the test is representative of the aircraft. Also, on aircraft ground testing may be required to confirm the detectability of seeded faults provided there is sufficient time between detection and component failure to maintain an acceptable level of risk to the aircraft and personnel. An initial TBO extension could be granted, assuming successful completion of the prescribed seeded fault tests for that particular component and verification that the fault is reliably detected on the aircraft. A minimum of three "true" positive detections for each credible failure mode are to be demonstrated by the condition monitoring equipment utilizing the reliability guidelines specified in paragraph 5.9.3 in order to be eligible for on-condition status. TDA's will be ongoing for components exceeding initially established CI limits. Once the capability of the monitoring system has been validated based on three "true" positive detections for each credible failure mode, incremental TBO interval increases are recommended prior to fully implementing the component to on-condition status. The number of incremental TBO extensions will be based on the criticality of the component. For more details, see Appendix F.

5.9.2.2 Field fault analysis. The guidance for achieving on condition status via the accumulation of field faults are essentially the same as those identified in paragraph 5.9.2.1. Incremental TBO extensions will play a bigger role utilizing this approach based on the assumption that the fault data will take much longer to obtain if no seeded fault testing is performed. A minimum of 3 "true" positive detections for each credible failure mode are to be demonstrated via field representative faults utilizing the detection guidelines specified in paragraph 5.9.3 in order to be eligible for on-condition status. TDA's will be ongoing for components exceeding initially established CI limits. Once the capability of the monitoring system has been validated based on three "true" positive detections for each credible failure mode, incremental TBO interval increases are recommended prior to fully implementing the component to on-condition status. The number of incremental TBO extensions will be based on the criticality of the component.

5.9.3 Statistical considerations. We are interested in the likelihood that the monitoring system will detect a significant difference in signal when such a difference exists. To validate our target detection and confidence levels (target detection = 90%, target confidence level = 90%) using a sample size of three possible positive detections, the minimum detectable feature difference is 3 standard deviations from the signal mean.

If at least one of the detections is a false positive, then evaluate to determine the root cause of the false positive. Corrective actions may involve anything from a slight upward adjustment of the

CI limit to a major change in the detection algorithm. Once corrective action is taken and prior to any further increase in TBO, additional inspections/TDAs of possible positive detections are necessary to continue validation of the CI.

A false negative occurrence for a critical component will impact safety, and should be assessed to determine the impact on future TBO extensions. Each false negative event will require a detailed investigation to determine the root cause. Once corrective action is taken and prior to any further increase in TBO, additional inspections/TDAs of possible positive detections are necessary to continue validation of the CI.

Components used for TDA and validation may be acquired through either seeded fault testing or through naturally occurring field faults.

5.10 CBM management plan. This handbook provides the overall standards and guidance in the design of a CBM system. It is beyond the scope of this document to provide specific guidance in the implementation of any particular CBM design. A written Management Plan or part of an existing Systems Engineering Plan should be developed for each implemented CBM system that describes the details of how the specific design meets the guidance of this ADS. At a minimum, this Management Plan is to provide the following:

Describe how the design meets or exceeds the guidance of this ADS by citing specific references to the appropriate sections of this document and its appendices.

a. Describe in detail how the CBM system functions and meets the requirements for end-to-end integrity.

b. Specifically describe what CBM credits are sought (examples are extended operating time between maintenance, overhaul, and inspection or extended operating time between overhaul or inspection).

c. Describe how the CBM system is tested and validated to achieve the desired CBM credits.

This Management Plan may be developed either by the US Army or by the CBM system vendor/system integrator subject to approval by the US Army. The Management Plan should be specified as a contract deliverable to the Government in the event that it is developed by the CBM system vendor or end-to-end system integrator. Also, the Management Plan for CBM design compliance should be a stand-alone document.

6. HOW TO USE THIS ADS

Department of the Army policy describes CBM as the preferred maintenance approach for Army aircraft systems and this ADS provides guidance and standard practices for its implementation. Establishing CBM is a complex undertaking with inter-related tasks that span elements of design engineering, systems engineering and integrated logistics support. The complexity and scope of the undertaking can cause uncertainty as to where or how to begin the process. The following guidance in FIGURE 2 is provided for two basic situations: (1) transition from the established

maintenance program to CBM for an aircraft already in service, known herein as "Legacy Aircraft" and (2) New Development aircraft or UAS.

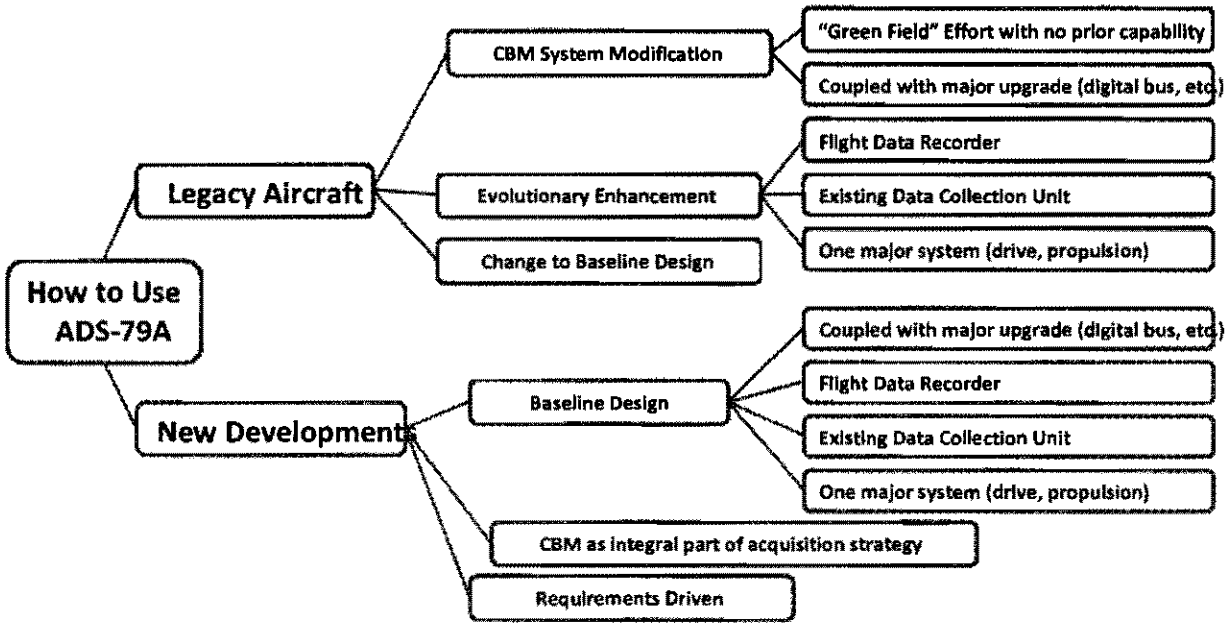

FIGURE 2. Mindmap of how to use ADS-79A-HDBK

6.1 Legacy aircraft. Legacy aircraft with established maintenance programs should consider incorporating CBM if the existing maintenance program is not providing sufficient aircraft or system reliability at affordable cost. CBM should be investigated and analyzed from a systems perspective to determine whether changing the maintenance program to incorporate CBM elements can increase readiness and decrease operating cost without penalizing aircraft performance, baseline risk or available funding, resources and time necessary to incorporate the CBM system design.

Using systems engineering and a total systems approach, legacy programs should establish a baseline of cost, reliability, performance and risk for the platform or system under study. The program should contain goals for improvements to these parameters to constrain the analysis and effort to design a CBM system for the aircraft which is under evaluation.

To establish the first description of the CBM system for the legacy platform, this ADS should be used in defining the requirements of the system design. The main body of the ADS provides guidance and descriptions of the overall system architecture and individual elements of the system needed to provide the data, analysis and basis for evaluating the maintenance needs of the aircraft or aircraft system based on the detection, identification and evaluation of faults through data collection and analysis. The Appendices provide more detailed guidance for elements of the CBM process. Developing and validating CIs and HIs are of the utmost importance.

FIGURE 3 shows a systematic approach to consider incorporation of CBM into an existing aircraft. Using existing data from reliability and maintenance, safety and operational

performance, life cycle sustainment analysis should be performed to evaluate the system performance. If the aircraft is sufficiently deficient to warrant further analysis, basic root cause analysis determines the cause of system's performance degradation. From this root cause analysis, Failure Mode Effects Criticality Analysis (FMECA) can identify a candidate list of components and associated faults that are candidates for CBM.

Further analysis of the faults and associated failure modes can determine the most effective means to sense the faults and develop the means to detect and identify the faults through sensor signal processing. The existing sensors and data collection system onboard the aircraft should be reviewed for suitability (using the guidance in the main body of the ADS as well as Appendix C, D and E for guidance on sensors, CIs and data management). If the existing system does not provide sufficient sensors and data for fault detection, Appendices C and D contain more detailed guidance.

As CI development progresses, data from laboratory testing or seeded fault testing may be required to validate the CI suitability and accuracy. For additional guidance, see Appendices C, D and F.

Flight testing of the system will be the final step toward CBM deployment. For guidance on flight data accuracy, flight regime recognition (including maneuver severity and duration), and other flight test requirements, see Appendix B.

Finally, the CBM system performance should be analyzed and estimated prior to the decision to go to full rate production and deployment. This analysis and recommendation should be accomplished using standard systems engineering methods and performance measures.

For aircraft with existing sensors and data collection systems, some portions of the analysis and design have already been completed. The decision to add additional components to the system follows the same flow as shown in FIGURE 3, with the emphasis focused on requirements for the additional aircraft system or component rather than the whole aircraft. It is important to review the existing system design and ensure that it meets the requirements for CBM as outlined in this ADS. Legacy sensors and data collection systems may lack elements which provide the means to modify the legacy maintenance program to CBM.

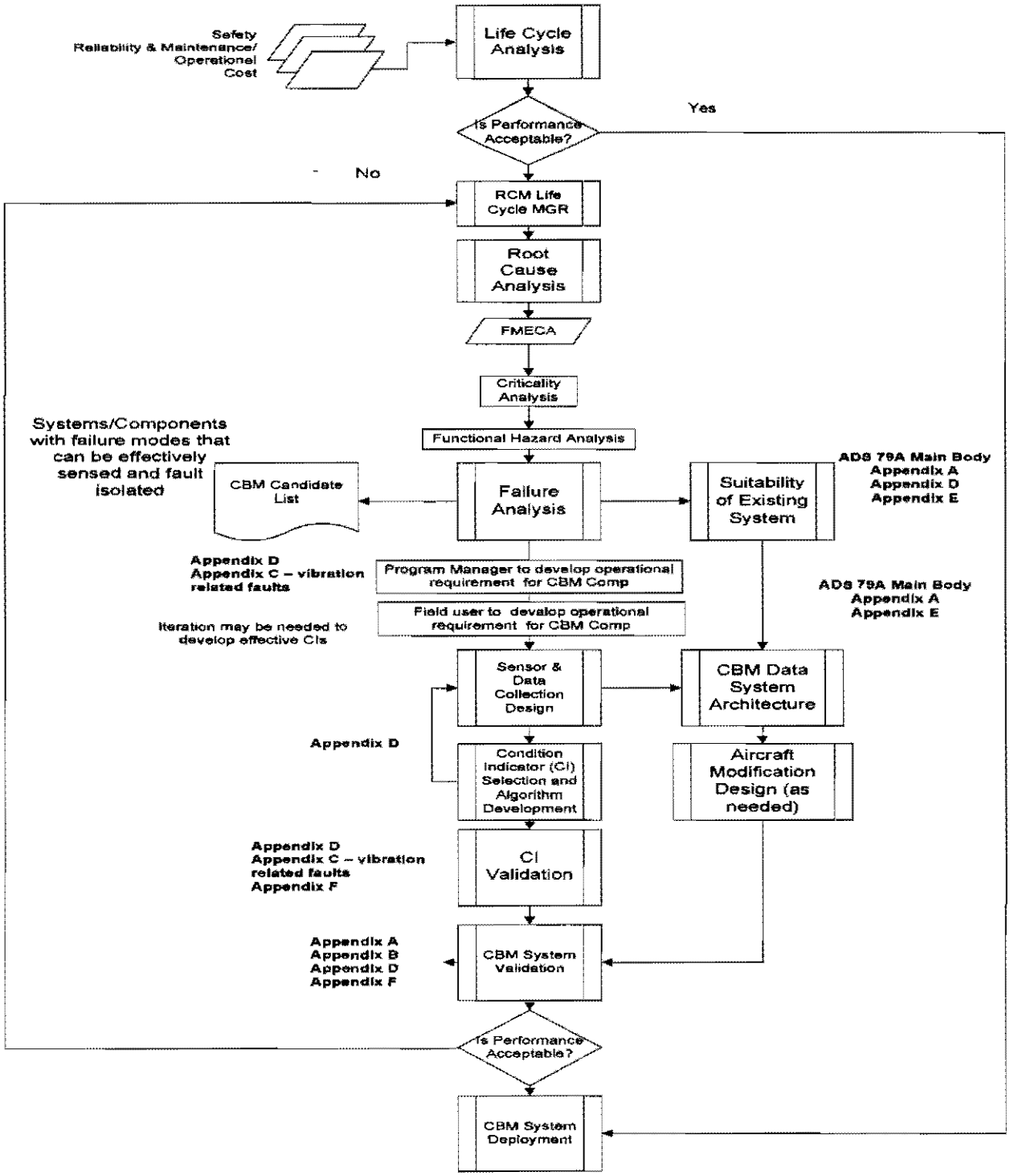

FIGURE 3: CBM development for legacy aircraft

6.2 New developmental systems. In the development of a new aircraft or UAS, CBM should be considered when evaluating the maintenance approach as part of the initial requirements determination. This decision enables the incorporation of CBM elements as part of

an integrated system of systems, potentially lowering the cost of incorporation of sensors, data collection hardware, aircraft systems and components.

The true value of CBM is found in the integrated logistics support elements, and design studies and trade-off analyses should be cognizant of potential improvements in spare parts inventory cost, repair labor costs and overall system reliability.

Therefore, given the CBM system is critical to logistics and maintenance credit; it should be handled and maintained as a key component of the overall platform. The Government may also, at its discretion pending criticality of the maintenance item being monitored, use the CBM system to determine airworthiness of the aircraft. The Government will make the decision when an aircraft should be grounded by an inoperative CBM system. These operational considerations should be documented as part of the CBM Management Plan along with the steps to recovering normal logistics and maintenance following data loss or a time gap in CBM system operation.

Figure 4 shows a systematic approach to incorporation of CBM in a new acquisition. Establishing CBM as a system requirement by the Government is the first step, with this ADS serving as a source for guidance on the specific requirements. Both the Government and original equipment manufacturer (OEM) can use the ADS as the basis for the determination of requirements and the systems engineering processes related to design, validation and verification. Setting the requirement for CBM in the initial requirements document provides the greatest opportunity for integration of the sensors and data management hardware with other aircraft systems.

Once the preliminary design of the aircraft or UAS is underway, systems engineering methods are used to evaluate the reliability and maintainability of the emerging design. One of the outputs of this systems engineering process is the Failure Modes Effects and Criticality Analysis (FMECA). The FMECA documents the failure modes and effects of the system. Upon completion of the FMECA a Reliability Centered Maintenance (RCM) analysis is performed to identify the appropriate failure management strategy for each identified failure mode. While the FMECA identifies all areas where CBM could be utilized, the RCM analysis identifies where CBM is the most appropriate failure management strategy. Appendices C and D are useful in providing additional guidance on the selection and development of CIs for the components in the new design.

Once the candidate list has been chosen, analysis and planning to determine how to develop data to support CI development will most likely consider seeded fault testing as well as modeling and simulation. Appendices C, D and F contain additional guidance for this part of the analysis.

In parallel, the design of the overall CBM system architecture and data management elements can be assisted with guidance from the main body of this ADS as well as Appendix E. Design of the software and hardware/firmware elements can find additional guidance in the main body, and Appendices B thru E.

Validation of the CBM system through selected testing and flight testing can be assisted with guidance from Appendices B and E.

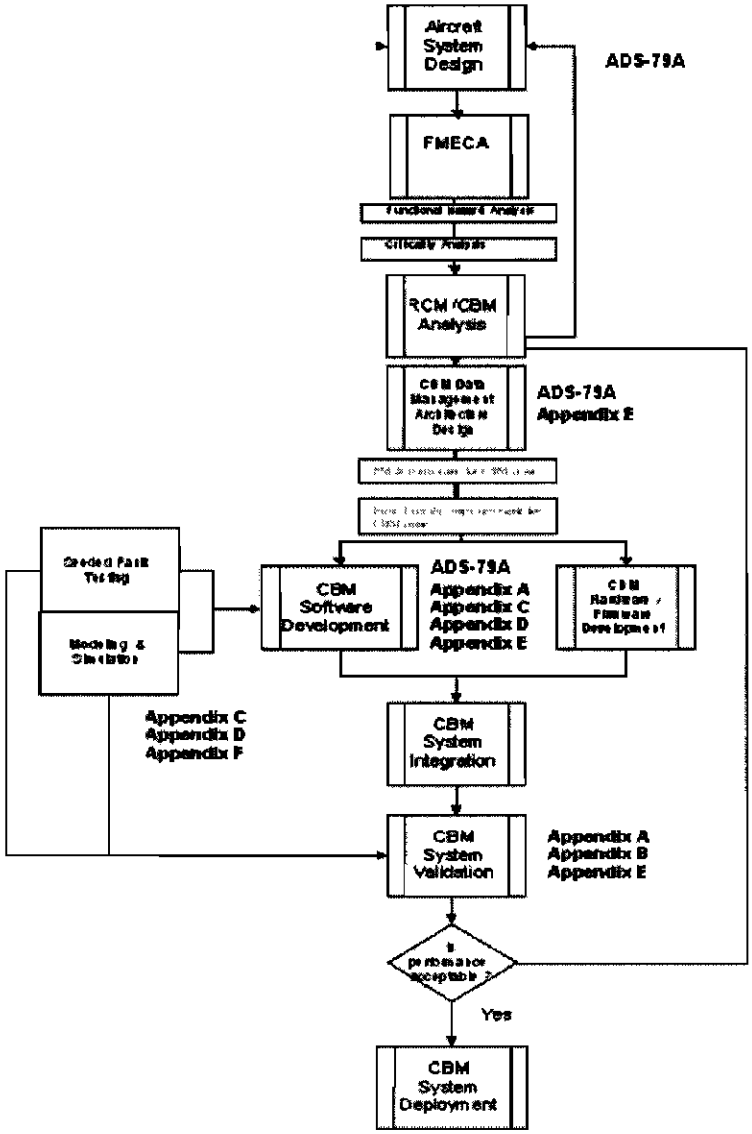

FIGURE 4. CBM development for new acquisition

APPENDIX A:

Fatigue Life Management

A.1 SCOPE

A.1.1 Purpose. The purpose of this appendix is to define the criteria for acceptance of airworthiness credit for incorporation of Condition Based Maintenance (CBM) into Army aircraft systems from a Fatigue Life Management (FLM) point of view. This appendix also documents potential applications of FLM.

A.2 REFERENCES

a. Memorandum, Program Executive Officer (PEO), Aviation Policy Memorandum Number 08-03, System Safety Risk Management Process, 20 Jun 2008.

b. Collins, J. A. *Failure of Materials in Mechanical Design: Analysis, Prediction, Prevention*. Wiley & Sons: New York, 1981.

c. JSSG-2001B, Department of Defense Joint Service Specification Guide, Air Vehicle, 30 April 2004

d. JSSG-2006, Department of Defense Joint Service Specification Guide, Aircraft Structure, 30 October 1998.

e. D. O. Adams and J. Zhao, "Searching for the Usage Monitor Reliability Factor Using an Advanced Fatigue Reliability Assessment Model", presented at the American Helicopter Society 65th Annual Forum, Grapevine, Texas, May 27-29, 2009.

A.3 DEFINITIONS

Ground Air Ground Cycles: Relatively low-frequency large-amplitude load cycles occurring during a given flight, but not present in any single flight condition. Examples include rotor start and stop cycles and load fluctuations between the various flight conditions encountered during performance of a mission.

Regime: Combinations of weight, altitude, C.G. and maneuvers that describe typical aircraft usage.

Top of Scatter: Flight load records and summary data, or flight load records and summary data which produce the highest fatigue damage for a given regime or load cycle when used in accordance with a given fatigue methodology.

A.4 INTRODUCTION

To qualify the structural integrity of an air vehicle, the US Army specifies a Structural Demonstration program and a Flight Load Survey (FLS) program. The structural demonstration tests are used to demonstrate the safe operation of the air vehicle to the structural design envelope. The objective of the FLS is to measure flight loads on components. Thus, the typical aircraft conditions flown represent the gross weight (GW), center of gravity (CG), airspeed, and

altitude combinations representative of the design load conditions. However, Army aircraft systems are subjected to almost continuous upgrades of capabilities and expansion of missions, creating new critical loading situations which were not flown during the initial FLS. It is essential that fleet management includes a task that will establish and track the relationship between the original design loads used by the original equipment manufacturers (OEMs) and the loads experienced during operational usage. FLM and usage/load monitoring, using flight recorder data, will provide the information needed to determine and track this relationship.

A FLM system should provide the capability to measure and record the actual environment (examples: usage, loads, configurations) experienced by Army aircraft systems. Through analysis these data can be correlated with established structural integrity methodologies, to establish appropriate maintenance actions.

As explained in the basic ADS (ADS-79A-HDBK), the goals of the FLM system are to minimize burdensome maintenance tasks, increase aircraft availability, improve flight safety and reduce maintenance cost. The primary objective of the FLM process is to enable updating of the usage spectrum required for maintaining airworthiness of Army aircraft systems.

The secondary objectives include providing:

a. Intervals at which specific component maintenance or replacement actions are required.

b. Usage statistics for each operational command base, unit or aircraft.

c. The rate at which the fatigue capability of a component is being used and an estimate of the remaining fatigue life.

d. Usage and loads data to support a balanced approach in establishing damage repair limits.

e. Data required for effective Risk Management of the Army's fleet of aircraft systems. (For example, the loads environment prior to and during a mishap incident provides data required to evaluate the incident and minimize the readiness impact on the fleet.)

It is not the intention of a FLM system to control the manner which Army pilots perform their missions. However, the CBM system will make possible the tracking of the loads environment that the aircraft experiences in terms of severity, duration, and frequency of occurrence. This will make it possible to adjust retirement times and inspection requirements based on the severity of the loads environment. Loads variability between pilots performing the same mission can be a dominant factor in establishing retirement times and inspection requirements. Feedback to the user concerning loads severity has a significant potential for reducing maintenance burden and enhancing safety.

The purpose of section A.5 is to provide insight of the Army's expectations of utilizing a FLM system to enhance Fatigue Life Management and Remediation. The Reliability Criteria for establishing maintenance actions based on a FLM system are provided in section A6.

A.5 POTENTIAL APPLICATIONS

A.5.1 Updating design usage spectrums. The FLM system provides the capability to update current design usage spectrums of Army aircraft systems. Refinement with respect to prorating velocity, load factor, angle of bank, sink speed, altitude and GW provides greater accuracy in representing actual usage. The number of aircraft required to participate in a usage survey should be statistically significant. Likewise, a survey should be conducted at sufficient locations to ensure inclusion of all missions, including training locations to ascertain appropriate usage severity. When possible, pilot interviews should be conducted in concert with FLM usage data in updating usage spectrums.

The updated usage spectrum provides greater accuracy of current usage. However, the updated spectrum should maintain its intended contribution to component reliability when used to compute retirement lives. Likewise, the impact on reliability for a segment of the fleet should not be compromised through creation of an overall fleet usage distribution. An example of this would be for a small population of the fleet operating at more severe usage (example, training aircraft with more GAG and autorotation cycles) which is allowed to interchange components with the majority of the fleet. Lives may be calculated based on an updated worst case usage spectrum for the entire fleet, including the effect of more severe usage for a portion of the fleet. Alternatively, the worst case life may be determined based on lives calculated in accordance with a basic usage spectrum for the majority of the fleet and a special case spectrum for a unique segment of the fleet.

An example of maintaining required 0.999999 (six nines) reliability using updated usage spectrum from HUMS is given in reference Adams and Zhao, AHS 2009[5] for the case where:

a. Design composite worst case usage spectrum was intended to reflect the 90-th percentile of total population of the anticipated usage.

b. Design Top of Scatter (TOS) load was intended to reflect the 99-th percentile of total population of the anticipated load.

c. Fatigue design working curve was selected to reflect the 99.9-th percentile of total population of components.

A.5.2 Managing service life of CSI components. The service life of Critical Safety Items (CSI) on Army aircraft systems is normally managed by a safe life process that is based on a calculation of a fatigue damage fraction. The inputs for establishing the safe lives include usage, flight loads and fatigue strength with damage fraction calculation based on Miner's linear

[5] D. O. Adams and J. Zhao, "Searching for the Usage Monitor Reliability Factor Using an Advanced Fatigue Reliability Assessment Model", presented at the American Helicopter Society 65th Annual Forum, Grapevine, Texas, May 27-29, 2009.

cumulative damage hypothesis.[6] Although there is no identified safety factor used to ensure the reliability of CSI reaching their retirement time without a structural failure, reliability goals are reached by a combination of conservative assumptions employed in developing the usage spectrum and flight loads in conjunction with statistical reductions included in the fatigue strength working curve. Incorporation of the FLM system allows greater certainty of aircraft usage and flight loads severity. Due to this increased certainty, the analysis of FLM data and correlation with component fatigue capability has great potential of achieving FLM goals of reducing burdensome maintenance tasks, increasing aircraft availability, improving flight safety and reducing sustainment costs. The following should be considered when implementing FLM in order to maximize benefits.

Usage: FLM regime recognition monitoring system will track the maneuvers and aircraft gross weight configuration (examples: CG, gross weight, external store.). To properly account for fatigue damage for a flight or mission, fatigue damage should be established for each damaging regime. In addition, maneuver to maneuver damage including GAG should be evaluated and included in total flight damage calculation. In the event the regime recognition monitoring system is not operational, the fatigue damage should be accounted for by applying the worst case assumed fatigue damage determined from the most current design usage spectrum at a minimum.

a. Loads: Maneuver damage assigned to each regime should be based on top of scatter loads (i.e. loads that produce the highest fatigue damage for the regime). Likewise, maximum/minimum loads for maneuver-to-maneuver including GAG cycles should be based on top of scatter loads. For systems that measure both usage and loads, the reliability of the strength curve and damage sum methodology or reliability of the strength curve or damage sum methodology should provide the reliability guidance of section A.5.

b. Fatigue Strength: Fatigue damage should be calculated using the mean minus 3 sigma (μ - 3σ) probability strength with a 95% confidence level or the working S-N curves in the approved fatigue substantiation reports.

c. Damage Sum: Component retirement when fatigue damages sum to less than 1 should be considered to ensure that the reliability threshold (i.e. 0.999999 (six nines) component reliability or 0.01 failure per 100,000 flight hour's system hazard) is met.

A.5.3 Remediation. There are myriad reasons why structural components are removed from service before reaching their respective component retirement time (i.e. fatigue life). In fact, the majority of Army components are removed due to damage (examples: nicks, corrosion, wear) prior to reaching a retirement life. Remediation is the concept of identifying and mitigating the root causes for part replacement in order to obtain more useful life from structural components (including airframe parts and dynamic components). The safe life process for

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[6] Collins, J. A. *Failure of Materials in Mechanical Design: Analysis, Prediction, Prevention.* Wiley & Sons: New York, 1981.

service life management bases fatigue strength on "as manufactured" components. Damage, repair and overhaul limits are established to maintain component strength as controlled by drawing tolerance limits.

The remediation process provides the means to trade repair tolerance for retirement time. Utilization of actual usage and loads provides the means to extend the retirement time at acceptable levels of risk. The steps in the remediation process follows:

a. Categorize and quantify the primary reasons for component removal and decision not to return the component to service.

b. Investigate regime recognition data for casual relations between usage and damage.

c. Perform engineering analysis on the component and evaluate the impact of expanded repair limits on static and fatigue capability. Regime recognition data provides information on load severity and usage for projecting revised fatigue life.

d. Perform elemental or full-scale testing to substantiate analysis.

e. Implement the results of the analysis and testing phase by adjusting repair limits and repair procedures where applicable, thereby increasing the useful life of the component and reducing part removals.

The result is an increase in damage repair limits in the TMs and DMWRs allowing the component to stay on the aircraft longer. Remediation enhances the four goals of the FLM process and can be considered a subset of the elements; analysis and correlation of data to component fatigue strength.

A.5.4 Managing service life of damage tolerant structure. The FLM process will provide necessary usage and loads data for continual airworthiness support of damage tolerant aircraft structure. The categories of damage tolerant structure include; slow crack growth structure, fail-safe multiple load path structure and fail-safe crack arresting structure.[7] A potential application is in the establishment of inspection requirements for airframe hot spots where fatigue cracking is discovered during the service life of the aircraft. The FLM derived actual usage, a direct load measurement or an updated usage spectrum will provide the loads data to establish the inspection procedure and frequency required to achieve the reliability requirement of section A.6 to prevent a catastrophic failure. The inspection would be performed until a repair or appropriate design change of the critical structure is incorporated in the fleet. The FLM collected data would also be used in the substantiation of the repair/redesign. The damage tolerance repair or new design should be substantiated to meet the goal of two design service lives without fatigue cracking.[8] The inspection requirements for the repair/redesign must be substantiated to the reliability requirements of section A.6 to prevent a catastrophic failure.

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[7]JSSG-2006, Department of Defense Joint Service Specification Guide, Aircraft Structure, 30 October 1998.

[8] JSSG-2001B, Department of Defense Joint Service Specification Guide, Air Vehicle, 30 April 2004

The FLM database will be utilized in the evaluation of existing structure, repairs, beef-ups and redesigns.

Also, the FLM system has the potential to provide input to the pilot that fatigue damage is occurring during sustained flight conditions (example level flight). The avoidance of or minimum duration in such a condition will significantly reduce aircraft fatigue damage and subsequent repair or catastrophic loss.

Application of the FLM process has the potential of significant improvements in readiness and reduction of sustainment costs for Army aircraft systems.

A.5.5 Maximizing FLM benefits. Regime recognition provides the tools necessary to continuously improve aircraft design, maintenance, and safety based on actual usage. Also, the potential exists for enhanced pilot training, improved understanding of regime damage variability and tailored risk management. The FLM Management Plan should include feedback of results to the user. Analysis of FLM data from a fatigue life management point of view will include the identification of significantly damaging usage and load environments. For systems capable of monitoring the damage severity of a regime (example loads or severity monitoring) the parameters correlating with the degree of damage will be identified. This will allow the preparation of guidance on how to perform maneuvers and missions that are less structurally damaging. Feedback to unit commanders will maximize mission reliability and allow them to better manage their logistic requirements associated with performing each type of mission. The potential exists to extend component lives and to minimize inspection requirements by reducing the severity of the usage environment of Army aircraft systems.

A.6 RELIABILITY GUIDANCE.

The incorporation of a FLM management plan in Army aircraft systems should not create a system hazard as defined by Program Executive Officer (PEO), Aviation policy memorandum number 08-03, System Safety Risk Management Process.[9] Acceptable methods of substantiating this guidance for manned aircraft systems are as follows:

a. Substantiate that the frequency of the system hazard is less than the threshold of the risk matrix (i.e., probability of occurrence is less than 0.01 per 100,000 flight hours). This is a cumulative frequency of all components managed by the FLM process. Incremental incorporation should require allocation of risk.

b. Substantiate that the incorporation of FLM has not increased the aircraft system level risk.

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[9] Memorandum, Program Executive Officer (PEO), Aviation Policy Memorandum Number 08-03, System Safety Risk Management Process, 20 Jun 2008.

c. Substantiate that a threshold component reliability of 0.999999 (six nines) is achieved. This means that the probability of failure for components managed by the FLM process is less than 1 out of 1,000,000 components.

A.6.1 Reliability Analysis. The FLM objective is to retire structural components based on actual usage to reduce operation and support costs, hence, improve readiness. The FLM process will provide necessary usage and loads data for continued airworthiness support. The FLM structural monitoring system provides potential service life benefit and meets the reliability requirement identified in this appendix. The following sections present examples on how reliability can be evaluated when implementing FLM for potential service benefits. The reliability analysis is a method for determining the probability of non-failure based on statistical evaluation of all critical parameters which include fatigue strength, flight loads, and usage spectrum. Fatigue reliability analysis can be predicted using analytical probabilistic models or Monte Carlo simulations.

A.6.2 Evaluation of reliability when usages are monitored and fatigue strength and flight loads are statistically evaluated. FLM usage monitoring track aircraft maneuvers and accumulates component fatigue damage. Component is removed when the tracked component reaches the minimum threshold of required reliability defined in this appendix. The reliability analysis is based on statistical evaluation of fatigue strength and flight load distributions when the usages of aircraft are monitored. The fatigue strength and flight load may be modeled as normal, log normal, Weibull or other distributions. Failure data from component qualification bench test should be the basis for development of the statistical distributions on fatigue strength. Flight load survey should be the basis for development of the statistical distributions on flight loads.

A.6.3 Evaluation of reliability when loads are monitored and fatigue statistically modeled. Loads monitor will be part of the FLM activities for understanding reliability of retired parts. The reliability analysis is based on statistical evaluation of fatigue strength when the component load spectrum is monitored. The fatigue strength may be modeled as normal or log normal distributions. Bench fatigue test data should be the basis for development of the statistical distributions. The fatigue damage calculated using the baseline mean-3 sigma fatigue strength curve for a normal distributed strength would result in 0.99865 reliability when actual load spectrum is applied. Component is removed from aircraft when it reaches the minimum threshold of required reliability defined in this appendix.

A.6.4 Evaluation of reliability when usages are monitored usage and design damages applied. For legacy aircraft baseline fatigue substantiation may not have sufficient data in the bench fatigue tests or load survey tests that allow development of statistical distributions of critical parameters. If a detail probabilistic analysis is not available for determination of component reliability, maximum accumulated damage should be tracked to no more than 0.5. Baseline retirement time are based on composite worst case design spectrum. The adjustment

of the accumulated damage is to ensure baseline reliability is maintained when component damages are accumulated using the actual flight maneuvers. Damage fractions greater than 0.5 can be used for retirement criteria if probabilistic based analyses demonstrate that baseline fleet risk levels are maintained.

A.6.5 Specific reliability guidance for unmanned aircraft systems._AE70-62-1-UAS provides reliability requirements for Unmanned Aircraft Systems (UAS).

APPENDIX B:

Regime Recognition/Flight State Classification

with Validation of Regime Recognition Algorithms

B.1 SCOPE

B.1.1 Scope. This Aeronautical Design Standard (ADS) Appendix provides guidance and standards for the development and validation of a method to measure flight regimes of aircraft as part of a Condition Based Maintenance (CBM) system for acquiring maintenance credits for onboard components.

B.2 REFERENCES AND APPLICABLE DOCUMENTS

B.2.1 References.

a. McCool, K. and Barndt, G., "Assessment of Helicopter Structural Usage Monitoring System Requirements," DOT/FAA/AR-04/3, April 2004.

b. Arthur E. Thompson and David O. Adams, "A Computation Method for the Determination of Structural Reliability of Helicopter Components", Presented at the AHS Annual Forum, May 1990.

c. Robert E. Vaughan, "Obtaining Usage Credits from Monitoring of Helicopter Dynamic Components without Impacting Safe Life Reliability", Presented at the AHS 63rd Annual Forum, May, 2007.

B.2.2 Applicable Documents. The documents listed below are not necessarily all of the documents referenced herein, but are those most useful in understanding the information provided by this handbook. In addition to the below documents review the main ADS-79A (of which this is Appendix B). ADS for Condition Based Maintenance for Army aircraft systems, for additional guidance in CBM system design should be considered.

B.2.2.1 Government documents. The following specifications, standards, and handbooks form a part of this appendix to the extent specified herein.

a. ADS-24. US Army Aeronautical Design Standard – Structural Demonstration for Rotary Wing Aircraft (inactive)

b. ADS-29. US Army Aeronautical Design Standard – Structural Design Criteria for Rotary Wing Aircraft (inactive)

c. ADS-51 HDBK. US Army Aeronautical Design Standard – Rotorcraft and Aircraft Qualification Handbook

B.2.2.2 Other Government documents, drawings, and publications. The following other Government documents, drawings, and publications form a part of this appendix to the extent specified herein.

a. DOT/FAA/AR-04/3. Assessment of Helicopter Structural Usage Monitoring System Requirements

b. DOT/FAA/AR-04/19. Hazard Assessment for Usage Credits on Helicopters Using Health and Usage Monitoring System

B.3 DEFINITIONS

Ground Air Ground Cycles: Relatively low-frequency large-amplitude load cycles occurring during a given flight, but not present in any single flight condition. Examples include rotor start and stop cycles and load fluctuations between the various flight conditions encountered during performance of a mission.

Structural Usage Monitoring: Managing fatigue lives via Usage Monitoring

Top of Scatter: Flight load records and summary data, or flight load records or summary data which produce the highest fatigue damage for a given regime or load cycle when used in accordance with a given fatigue methodology.

B.4 GENERAL GUIDANCE

In a standard, scheduled maintenance program, component retirement times (CRTs) are derived from the total expected exposure to regimes for which flight strain survey data is available. This expected exposure is based on an assumed mission spectrum determined by the class of aircraft. In a CBM system, however, component life calculations can be refined through knowledge of the actual amount of operational time spent in each flight regime. CRTs can be extended when an aircraft is actually exposed to less severe mission profiles and lower flight loads. Or, in the interest of safety, they can be reduced in the presence of higher flight loads than assumed in the original CRT calculations.

The process begins with identifying the set of flight regimes encountered in the mission spectrum for the class of aircraft. For each regime, the strain loads are determined during the flight load survey performed during the development phase of the airframe. Next, testing is performed to determine the rate of useful life reduction due to fatigue as a function of time or number of occurrences under the regime load for each component for which airworthiness credits are sought by the CBM system. Finally, one should develop an onboard instrumentation package that measures the flight state of the aircraft and accurately classifies the flight regime.

An accurate characterization of the operational flight regime is a key characteristic of the CBM system. A dynamic maintenance measurement system should not be implemented that might compromise flight safety in an attempt to extend operational life. Therefore, the flight regime classification system should be submitted to a rigorous validation procedure that guarantees

component Airworthiness Credits are not allocated through flight state measurement error, regime misclassification, or a compromise in data integrity.

Usage monitoring equipment is not flight or mission critical; if the system fails, an alternative is to apply the most current Design Usage Spectrum and the associated fatigue methodology for any period of flight time in which the usage monitor data is not available.

B.5 SPECIFIC GUIDANCE

B.5.1 Flight regime definition. Flight regimes are flight load events or states typically flown during a flight load survey to determine flight loads experienced by aircraft parameters based on combining the following types of parameters:

a. Aircraft configuration: On a mission by mission basis, items may be added or removed from the aircraft in a manner that might affect flight loads and aircraft center of gravity. For example, the presence of external stores, position of landing gear, weight of external or internal cargo, fuel quantity. These parameters are required to determine flight loads experienced by aircraft components.

b. Flight environment: Altitude, outside air temperature and other parameters that allow reasonable estimation of density altitude, which is required to determine flight loads experienced by aircraft components.

c. Flight Conditions or Maneuvers: General type of maneuver, its severity (examples: speed, load factor, angle of bank, rate of climb/descent), and duration.

Prior to conducting flight load surveys and fatigue life substantiation, flight regimes in the usage spectrum are typically specified for each aircraft model based on aircraft classification, current tactics, mission profiles, and anticipated threat environment (see ADS-51-HDBK for details). These regimes form the basis of fatigue calculations and should also form the basic requirement for regime recognition algorithms. However, changes in service use are common for aircraft since military tactics, operational tempos, and missions may change drastically from development to operation of the systems. Identification of new regimes using CBM data is possible based on inspection of raw parametric data for time spent in unrecognized regimes. Additional flight load surveys may be required to determine flight loads corresponding to previously unrecognized regimes.

B.5.1.1 Aircraft configuration. TABLE B-I is an example of items that define the aircraft configuration. This data is typically collected and maintained in the aircraft electronic logbook with information on serial numbers of each installed end item normally linked to flight data by the HUMS "ground station" or off board data collection and storage software.

TABLE B-I. Typical military helicopter configuration items (EXAMPLE ONLY)

	General Configuration Items
1	Main Rotor Blades
2	Main Rotor Swashplate
3	Main Rotor Shaft
4	Main Transmission
5	Engines
6	Auxiliary Power Unit
7	Tail Rotor Drive Shafts
8	Intermediate Gear Boxes
9	Tail Rotor Gear Box
10	Tail Rotor Blades
11	Flight Control Actuators
12	Flight Control Rods
13	Electrical Generators
14	Hydraulic System(s) Pumps
15	Landing Gear (s/n for each)
16	Mission/Weapon System Computers
17	EO/IR Sensor Systems Components
18	EW/Defensive Systems Components
	Mission Configuration
19	Ordnance Racks installed
20	Ordnance load (recorded for each flight)
21	External Fuel Tanks installed

The sample list of components above contain subassemblies and individual parts that are also often tracked by serial number to determine operational history, so databases containing configuration information should follow the work unit code (WUC) structure and serial number tracking requirements set by the initial design specifications.

B.5.1.2 Flight environment. TABLE B-II shows typical Flight Environment parameters, some of which are important to Regime Recognition as well.

TABLE B-II. Typical military helicopter flight environment parameters (EXAMPLE ONLY)

	Local Base Environment – Off Board Data Collection
1	Geographic Location
2	Afloat/Ashore (for landing severity and salt water effects)
3	Ambient Temperature - exposure (duration) at extremes
	Operational Environment – Collected On-Board
1	Outside Air Temperature
2	Altitude

B.5.2 CBM instrumentation design

B.5.2.1 Onboard flight state sensing. A set of measurable flight state parameters should be used as inputs to the regime classification algorithms. A typical set of flight state inputs are provided in *TABLE B-III.*

TABLE B-III. Flight states (EXAMPLE ONLY)

	PARAMETER		PARAMETER
1	Pilot's Indicated Airspeed	18	Pitch Rate (INS)
2	Co-Pilot Indicated Airspeed	19	Roll Rate (INS)
3	Outside Air Temperature	20	Yaw Rate (INS)
4	Barometric Pressure Altitude	21	Left Main LG WoW
5	Barometric Rate of Descent	22	Right Main LG WoW
6	Radar Altitude	23	Refueling Probe Ext
7	Normal Load Factor at CG	24	Heading (INS)
8	Main Rotor Speed	25	Roll Attitude (INS)
9	No. 1 Engine Torque	26	Pitch Attitude (INS)
10	No. 2 Engine Torque	27	Trim Ball
11	Average Engine Torque	28	Gross Weight
12	Longitudinal Cyclic Position	29	Increasing Fuel Quantity
13	Lateral Cyclic Position	30	Percent Vh
14	Collective Position	31	Equiv Retreat Ind Tip Speed
15	Directional Pedal Position	32	Elapsed Time
16	Roll Attitude (SGU)		
17	Pitch Attitude (SGU)		

The above list is provided as an example. The implemented list of parameters will be a function of available parameter sources onboard the aircraft and the input needs of the classifier algorithms. However, where possible, one should select natively available flight sensor sources and data buses (such as a 1553 data bus) that are available on the aircraft in lieu of adding custom instrumentation. This design decision serves to reduce the cost and complexity of implementation as well as insuring that flight state sensors are guaranteed to be operational and calibrated as part of normal aircraft maintenance procedures.

B.5.2.2 Flight state sampling rate. The CBM designer should select the appropriate sampling rate for acquiring flight state parameters. The selected rate should strike a balance between under-sampling with the potential of missing a desired effect and over-sampling which might produce more input than a data collection system can handle. A study for the FAA[10] points out the problem of having a sample rate that is too low. FIGURE B-1 from the referenced

[10] McCool, K. and Barndt, G., "Assessment of Helicopter Structural Usage Monitoring System Requirements," DOT/FAA/AR-04/3, April 2004.

report shows the maximum load factor that would be recorded for a pull-up maneuver at 2 different sample rates.[10] Figure B-1 clearly illustrates that too low a sample rate will miss the peak of the vertical acceleration and, thus, under-report the severity of the maneuver or, perhaps, not recognize the maneuver at all.

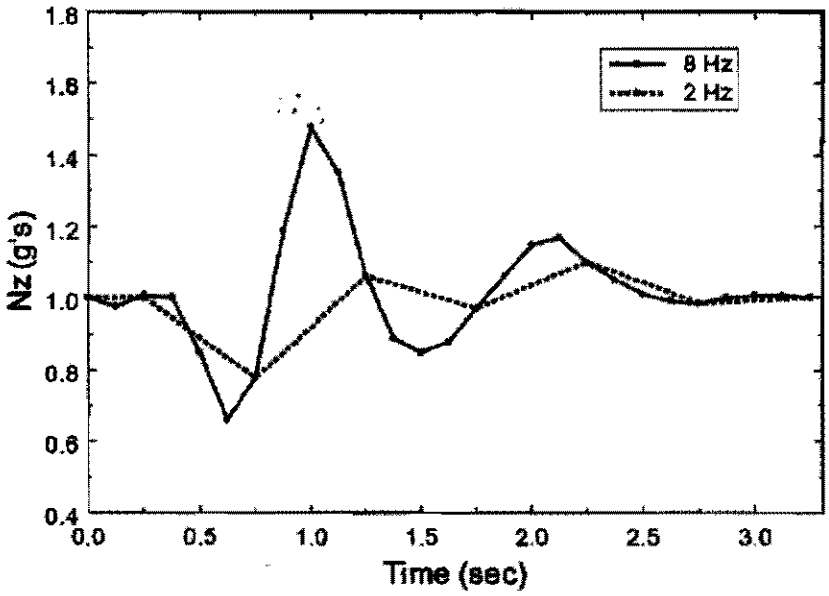

FIGURE B-1. Effect of data rate on vertical acceleration[10] (EXAMPLE ONLY)

The primary difficulty in supporting a high sample rate is data storage. One approach to reducing the amount of data acquired is to sample each parameter at its lowest acceptable rate. This requires knowing how quickly parameter values change during a given maneuver, particularly high fatigue damage maneuvers. Such considerations should also consider validation guidance provided in paragraph B.5.4. TABLE B-IV shows the example data rates for military helicopters for each parameter. Using the example rates in Table B-IV should not be considered a substitute to performing the validation described in paragraph B.5.4.

TABLE B-IV. Example typical military aircraft data rates[10] (EXAMPLE ONLY)

Parameter	Data Rate (Hz)	Max Error
Rotor Speed	6	0.83%
Vertical Acceleration	8	0.13 g's
Pitch Attitude	2	1.8 degs
Roll Attitude	4	2.0 degs
Pitch Rate	4	3.0 degs/sec
Roll Rate	8	2.8 degs/sec
Yaw Rate	4	2.5 degs/sec
Airspeed	2	4.3 kts
Engine Torque	6	3% error
Longitudinal stick position	6	3.1%
Lateral stick position	6	3.9%
Collective stick position	5	3.4%
Pedal position	6	3%
Long. acceleration	6	0.03 g's
Lateral acceleration	7	0.05 g's
Radar altitude	2	13 ft
Vertical velocity	8	242 fpm
Long. Flapping	8	0.61 degs
Lateral Flapping	8	1.0 degs
Lateral swashplate tilt	8	1.1 degs
Long. swashplate tilt	8	1.5 degs

Another approach to reducing data storage is to define bands within the expected range of values for each sensor and record only changes in the sensor bands. Hysteresis is typically used at the boundaries between bands to eliminate frequent toggling between bands at their boundaries.

B.5.2.3 Classification of flight regimes. A set of algorithms that use flight state measurements to classify regime and allocate occurrences/operational flight time and/or events to each regime should be developed. The regime classification and allocated flight recording should typically be performed in real-time onboard the aircraft in order to minimize the necessary amount of onboard data storage. However, pending selected sample rates and available onboard data storage capacity, one may elect to store raw, unprocessed fight state measurements for later processing on the ground during maintenance.

B.5.2.4 Component lifecycle tracking. To enable running damage assessments (by serial number), a maintenance database system should be developed that accurately allocates regime flight load time and occurrences to the specific component serial numbers flying on the aircraft. This requires that a database containing indentured parts lists with component serial numbers for each aircraft tail number be maintained as part of the maintenance logistics process. Also, relational integrity checks should be performed as the regime measurement data package is used from the aircraft to update the component ground maintenance records in order to insure that flight time is correctly assigned to the correct component serial number.

B.5.2.5 Data compromise recovery. A recovery procedure should be specified for regaining integrity of component ground maintenance records in the event of data corruption or loss. For example, a mismatch occurs in relating the regime measurement data package with a component in the maintenance database or the occurrence of a catastrophic loss of either the measurements or the ground database. The recovery procedure insures that a component serial number is not orphaned without any means of determining its retirement time.

The recovery process may be as simple as maintaining a hardcopy log that records when a component serial number was put in service. The CBM Management Plan should address the process when an event of CBM system data loss or corruption occurs. An acceptable approach is to account for the time lost using the damage rate produced by the design usage spectrum, as updated throughout the life cycle of the aircraft. For example, if a part has a 2000–hr CRT under a scheduled maintenance program for a given aircraft and an error occurs in component tracking resulting in a complete loss of data for the component's first 2000 flight hours, then the part reverts to the 2000-hr retirement schedule because no maintenance credit may be awarded by the CBM system based on running damage assessments.

One should consider the criticality of the failure associated with a component when specifying a data compromise recovery strategy. A more conservative procedure should be specified when failure consequences are more severe. As a result, the CBM system designer may specify a different recovery procedure for every component in the maintenance tracking database. In the worst case, it may be specified that a component be replaced immediately when data loss occurs.

B.5.3 CBM Instrumentation validation. Prior to deploying the flight regime measurement package as part of operational usage monitoring, a test aircraft should be instrumented for demonstration that the algorithms can accurately classify flight regimes. For developmental programs this can be performed as part of the Flight Loads Survey Testing (FLST) where the aircraft will be exposed to the range of flight regimes specified in the design usage spectrum. The bin range of regimes should be set for an aircraft equipped with usage monitoring in order to maximize maintenance credits. The current large bin ranges and associated loads data will not permit maximum benefits for a monitored aircraft. For legacy aircraft, flight testing should be performed to verify the capability of the usage monitoring system in identifying the regimes of the design usage spectrum. Also, additional FLST may be beneficial to maximize maintenance credits for usage monitoring. These additional flights allow smaller bin ranges that will improve the accuracy of fatigue damage calculations. For example, if the current regimes bins turns into 45 and 60 degree angle of bank (AOB), any turn recognized by the usage monitoring system with an AOB less than 45 degrees would be assigned to the damage accumulated for a 45 degree turn. Gathering load data for AOB less than 45 degrees

and restructuring the bin range for turns will allow more accurate tracking of usage and realistic damage fraction calculations.

B.5.3.1 Algorithm validation methodology. A series of flights should be performed with a test aircraft that is fully equipped with the regime measurement package and additional recording systems for capturing data needed to evaluate and tune the algorithms.

Engineering should prepare a series of flight cards identifying the maneuvers for which algorithms have been developed. The monitoring flight test engineer should know the sequence in which the pilots are flying the maneuvers and their target severity and duration. After the flight, the data records will be surveyed to determine which maneuvers were sufficiently detected and which maneuvers require improved algorithms. Algorithm optimization will be performed and a subsequent flight made in a totally different sequence using the improved algorithms. The post flight process will be the same. Usually two optimization flights are sufficient but additional flights may be necessary to achieve the desired regime classification accuracy. For aircraft with a very large range in gross weight (GW) it may be desirable to check the accuracy of the algorithms at very heavy and very light GW. Additionally, an aircraft that has a very high altitude mission may require algorithm validation at both high altitude and near sea level conditions.

Finally, without any knowledge of the flight card content, a comprehensive flight card should be developed which incorporates all of the maneuvers for which algorithms have been developed. The regime recognition design should identify the maneuvers flown, their severity and duration, such that 97% of the entire flight time is properly identified.

B.5.3.2 Accuracy. CBM RRAs should demonstrate that they can define 97% or greater of the actual flight regimes. Also, for misidentified or unrecognized flight regimes, the system should demonstrate that it errs on the side of selecting a more severe regime. This insures that a component is not allowed to receive maintenance credit where it is not due and therefore allows a component to fly beyond its margin of safety.

B.5.4 Validation of structural usage monitoring system (SUMS). The primary objective of the SUMS is to enable updating of the usage spectrum required for maintaining airworthiness of the aircraft. Composite design usage spectrums typically are very conservative, and contribute to overall system reliability. Quantifying how much reliability is attributed to conservative usage spectrums is very difficult. However, a study on UH-60 components[11] indicates that for the assumed 0.999999 (six nines) reliability, one nine was attributed to the design usage spectrum. The composite design usage spectrum must account for usage variations between aircrafts, units, and missions as flight hours are accumulated. When performing running damage assessments, usage is known and reliability is based solely on variations in loads and fatigue strength (see Fatigue Life Management guidance in Appendix A for details).

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[11] Arthur E. Thompson and David O. Adams, "A computation Method for the Determination of Structural Reliability of Helicopter Components", Presented at the AHS Annual Forum, May 1990.

Accordingly, when replacing the composite design usage spectrum with an actual usage spectrum the component reliability is reduced through loss of conservatism in the usage spectrum. Loss of conservatism is primarily due to the differences in fleet reliability that result when converting from the flight hour retirement approach, where retired components have variable reliability due to different usage, and the usage monitoring approach, where retired components have a comparable reliability. This reliability (~one-nine) can be restored by applying a factor (Life Factor) to the damage fraction from Miners Rule.[12]

A SUMS in which the primary objective of the system is to develop an actual usage spectrum and in which a Life Factor is used to restore reliability can be validated using the scripted flights as described in paragraph B.5.4.5 (Scripted Flights).

Implementing SUMS with this Life Factor approach may significantly reduce the benefits of the system. As mentioned above, inadequacies in the current flight loads data prevent the calculations necessary to rigorously quantify aircraft reliability (i.e. Loads Variability). It is necessary then to establish the parameters of aircraft reliability concurrently with the qualification process of the SUMS. The reliability of the aircraft can then be established and maintained by the SUMS throughout its lifecycle. As a result, the reliability knock-downs or adjustments (Life Factor) will no longer be necessary. The objective of the following is to provide guidelines for the qualification a Structural Usage Monitoring System that will form the basis for establishing the current reliability and establish the basis for maintaining that reliability for the entire lifecycle of the aircraft. Fully validated SUMS should be considered an intimate part the airworthiness process throughout the aircraft's lifecycle. Accordingly, the SUMs process should be included in the airworthiness qualification process for the aircraft.

B.5.4.1 Introduction. The design usage spectrum defines the number of occurrences or amount of time spent in different flight regimes during a block of operational flight hours. This defines the amount of time for each different configuration and the amount of time at different altitudes. Also, defined in the usage spectrum are assumed fixed number of occurrences for certain events (e.g., number of ground-air-ground (GAG) cycles per flight hour). SUMS have the ability to measure and provide the actual usage of aircraft for utilization in fatigue damage calculations.

The plan for validating SUMS should consider the components of the aircraft that are to receive maintenance credits. The regimes that are fatigue damaging to these components are documented in the fatigue substantiation and qualification databases of the aircraft. This includes all spectrum maneuvers flown at the various GW and CG loadings. Also defined is the magnitude of the fatigue damage fraction for the different regimes for usage per the design spectrum. Fatigue damage is also identified as being from within maneuver damage, maneuver to maneuver damage or GAG damage. To appreciate the data requirements for the usage monitoring system it is important to understand the characteristics of the loads producing the fatigue damage. For instance, damage within the maneuver can be caused by loads generated

[12] Robert E. Vaughan, "Obtaining Usage Credits from Monitoring of Helicopter Dynamic Components without Impacting Safe Life Reliability", Presented at the AHS 63rd Annual Forum, May, 2007.

during the entry or exit portions of a maneuver. Here, the duration time of the maneuver does not correlate with the amount of fatigue damage. In contrast, when blade performance (example, stall) produces cyclic loads that are damaging, the duration of the maneuver correlates with the amount of damage. Maneuver to maneuver damage depends on the pairing of maximum and minimum loads. The pairing can be between within maneuver loads but most often the pairing involves loads from different regimes. The sequence should include a pre or post flight static event ("unloaded") to assure proper representation of the GAG which pairs the highest and lowest load magnitude over the entire flight. Here, an optimum usage monitoring system will aid in a realistic pairing of loads to generate appropriate cyclic and mean loads. Usage monitoring will provide data to increase certainty on the magnitude of the loads as well as the number of occurrences. The usage monitoring system should have the ability to identify and store the sequence of regimes for maneuver to maneuver damage.

Qualification of the structural usage monitoring system to obtain maintenance credits requires validation. Aspects of the validation include; definition of the structural usage monitoring system design, identification of parameters and development of algorithms, verification of the ability to identify regimes, and verification that managing the continued airworthiness with the structural monitoring system will result in the reliability requirements identified in Appendix A of this ADS. This discussion will be limited to the approach where the monitoring system is utilized to identify the actual usage (regimes) and where the associated regime loads exist from prior flight load surveys. Also, it is assumed that the analysis of data that substantiates the maintenance credit may include a ground based computer system.

B.5.4.2 Development of the structural monitoring System. This effort consists of the design of the monitoring system and parameter identification and algorithm development for usage recognition. The design includes the onboard and ground software and hardware systems for collecting and storing usage data. A formal report that documents this effort will be provided to the certifying official as part of system validation. The topics to be addressed in the report submittal are provided in the paragraphs B.5.4.3 and B.5.4.4.

B.5.4.3 Design of the structural monitoring system. The report will define the structural monitoring system, including software and hardware including location (on-board or ground-based). A data integrity verification check process will be designed into the system and documented in the report. Dataflow and data management are an integral part of a usage monitoring system and will be considered in the validation process. The approach to ensure data integrity considering dataflow, data storage, access and retrieval will be provided. Also, a system for identification and tracking the monitored components will be identified as will a procedure to address a condition of an inoperative monitoring system.

B.5.4.4 Parameter identification and algorithms development. SUMS monitor aircraft state parameters in order to identify the maneuver that the aircraft is performing. Parameters will be selected and data collection rates established such that critical regimes will be decisively identified. Sufficient parameters will be monitored to differentiate between regimes that cause different levels of component fatigue damage. Aircraft GW, CG location (longitudinal and lateral), and store configurations are key characteristics of damaging regimes. An effective structural monitoring system will be capable of identifying the configuration of the

aircraft in order to identify the correct regime and associated damage. The following capabilities of the monitoring system will be substantiated:

a. Ability to identify the regimes that cause fatigue damage to the identified components. The parameters sampling rate should be sufficient to identify the severity of the maneuver. However in order to minimize the quantity of data, the sampling rate should not be higher than required for that purpose.

b. Ability to identify the duration of regimes when damage depends on maneuver duration.

c. Ability to identify and store the sequence of regimes for maneuver to maneuver damage.

The formal report will document the algorithm development and verification. The report will provide the basis of algorithm development, the flight test database utilized in the development of the algorithms, and a listing of all parameters utilized in regime recognition algorithms. The report will document the sensitivity of regime algorithms to specific parameters. The selection of data rates will be substantiated such that peak maneuver information is properly captured while excessive rates are not selected such that a large quantity of unnecessary data is collected. The process used for optimizing the regime recognition reliability will be provided, including the process utilized in selecting between similar regimes. The process for identifying aircraft configuration (GW, CG and stores) will be defined. Also, the configuration/regime association will be stated (example, the configuration associated with a regime will be the configuration at the start of the regime).

B.5.4.5 Scripted flights. Scripted flights should be flown based on a series of flight cards that identify the maneuvers that correspond to the regimes that are damaging to components that have been identified to receive maintenance credits based on structural usage monitoring. The characteristics of the regime that are significant to component fatigue damage will be matched during the scripted flights. The ability to identify aircraft configuration (GW, CG, and stores) will be demonstrated. The regimes identified by the structural monitoring system will be compared to the regimes defined by flight cards and by a review of the recorded state parameter time history data. The purpose of these flight tests is to verify that the usage monitoring system can identify the significant regimes of the usage spectrum. The maneuvers will be flown 3 times with 3 different pilots for a total of nine repeated flights of all critical regimes. The repeats are planned to address the variability introduced by pilot technique in order to assess this influence on regime identification and classification. Data collection and processing will utilize the onboard and ground software and hardware proposed for structural monitoring of fleet aircraft. The data integrity checking process will be demonstrated.

B.5.4.6 Unscripted flights. The unscripted flights should be performed to verify that execution of continued airworthiness utilizing the structural monitoring system will meet or exceed the safety requirements defined in Appendix A of this ADS. Actual fleet usage of the aircraft may involve maneuvering that does not fit neatly into precisely defined regime bins. Therefore, this effort will include flight testing of a load/strain instrumented aircraft, comparison of loads and comparison of fatigue damage for simulated missions. The missions and associated

usage will be representative of the regime environment in which the monitoring system will be used. Likewise, usage data will be collected and processed utilizing the onboard aircraft and ground software and hardware proposed for fleet airworthiness management.

B.5.4.7 Flight testing. A goal of the mission flight testing is to provide multiple repeats of both commonly flown missions or mission segments and also missions segments that are less frequently performed, but could result in high fatigue loads. Identified missions should be flown a minimum of 3 times. A minimum of 3 operational pilots should be utilized such that each trial of the same mission is flown by a different pilot. Extensive steady level flight elements of missions such as transit legs can be eliminated from the test mission flights; however transit time which includes contour flight should be included for a representative length of time.

B.5.4.8 Comparison of loads. Measured loads should be separated into the regimes identified by the structural monitoring system. These loads will be compared to the Top of Scatter (TOS) loads measured in Flight Loads Surveys and utilized in establishing the current fatigue lives of the selected components. The goal is to identify the magnitude of the TOS load relative to the load distribution of the selected regime. For example a 95% load would have only 5% of the loads in the distribution larger than the TOS load. This is a significant input when evaluating the reliability of structurally monitored damage fraction calculations.

B.5.4.9 Comparison of damage fraction. The damage calculated from the measured loads for each mission should be compared to the damage predicted by using the usage identified by the monitoring system and the TOS loads for each of the identified regimes. Direct comparisons should be made of within maneuver, maneuver to maneuver and GAG damage and overall flight damage. The damage calculated for measured loads per maneuver will use rain flow cycle counting to pair maximum and minimum loads. This damage will be compared to the damage calculated utilizing TOS loads and the procedure for maneuver to maneuver and GAG as documented in the aircraft's fatigue methodology report. Overall flight damage will be calculated from rainbow cycle command loads from flight start to flight end for comparison to the usage based damage sum and the maneuver load based damage sum.

Appendix C:
Minimum Guidance for Determining CIs/HIs

C.1 SCOPE

This Appendix to the Aeronautical Design Standard (ADS) for CBM provides guidance for the development and testing of all Condition Indicators (CIs) and Health Indicators (HIs) used in the Condition Based Maintenance (CBM) system. It includes analytical methods, signal processing software, and data management standards necessary to support their use to implement CBM as the maintenance approach to sustain and maintain systems, subsystems, and components of US Army aircraft systems.

C.2 REFERENCES AND APPLICABLE DOCUMENTS

C.2.1 References.

a. Vachtsevanos, G., Lewis, F.L., Roemer, M., Hess, A., and Wu, B. *Intelligent Fault Diagnosis and Prognosis for Engineering Systems*. Wiley & Sons: New York, 2006.

C.2.2 Applicable Documents. The documents listed below are not all specifically referenced herein, but are those useful in understanding the information provided by this Appendix.

C.2.2.1 Government documents

a. ISO 13374:2003, Condition monitoring and diagnostics of machines.

b. MIMOSA Standard "Open Systems Architecture for Condition Based Maintenance" v3.2, December 2006.

c. MIMOSA Standard "OSA CBM for Enterprise Application Integration" v 3.2, December 2006.

d. US Army CBM+ Roadmap, Revised Draft 20 July 2007.

e. US Army AMCOM Condition Base Maintenance (CBM) Systems Engineering Plan (SEP) Revision – 30 Nov 2007.

C.2.3 Definitions

Condition Indicator (CI): A measure of detectable phenomena, derived from sensors that show a change in physical properties related to a specific failure mode or fault.

Health Indicator (HI): An indicator of need for maintenance action for a component resulting from either a single CI value or a combination of two or more CI values.

Physics of Failure: The physical phenomena that are analytically defined and describe the process by which a mechanical component fails during operation.

C.2.4 Process description. Condition Based Maintenance (CBM) is a maintenance approach that uses the status and condition of the asset to determine its maintenance needs.

CBM is dependent on the collection of data from sensors and the processing, analysis and correlation of that data to maintenance actions.

The processes governing CI and HI development are:

a. Physics of Failure Analysis.
b. Detection Algorithm Development
c. Fault Correlation Data Mining
d. Fault Validation/Seeded Fault Analysis
e. Inspection/Tear Down Analysis
f. Electronic and Embedded Diagnostics (BIT)/(BITE)

Related processes that develop estimates of remaining useful life and therefore establish the actions necessary to restore system operation (the objective of HIs) include:

a. Usage Monitoring / Regime Recognition
b. Failure Prognosis and Health Management Systems Analysis
c. Remediation / Remaining Useful Life
d. Airworthiness Release for Maintenance Benefits (a.k.a. Airworthiness Credits)
e. Technical Manual Changes

Each of these technical processes are described in detail in the AMCOM CBM system Engineering Plan (SEP) and are subject to review and analysis to ensure that the resulting algorithms and supporting software achieve accurate and repeatable results.

The technical processes described above are used to create a comprehensive and integrated knowledge base which develops effective maintenance tasks and supporting processes necessary to sustain normal operations. The knowledge base changes during the life cycle of the aircraft and serves as the foundation for changes to maintenance practice created by new failure modes, aging effects, and changes to the mission profiles of the aircraft. In addition, as new technology, such as corrosion sensors or improved diagnostics for avionics, becomes proven, new data and detection algorithms will be added to the knowledge base.

C.3 PROCESS GUIDANCE. Detailed Failure Modes Effects Criticality Analysis (FMECA), often completed as a part of Reliability Centered Maintenance (RCM) analysis, is a favorable starting point for understanding the system, subsystem or component for which the CIs are being developed. Part of this analysis should develop physical and functional models of the system, subsystem and components as a means to determine the likely faults that may arise and their effect on the functions of the various elements of the system.

Models of the fault modes, developed through either simulation and modeling or empirical measurement and analysis through testing should be used to develop first estimates of the fault behavior as it progresses from initiation to failure. This is often described as "Physics of Failure" modeling and analysis. This modeling and analysis is accomplished with the scale and

resolution acceptable to model the particular fault and item geometry. For example, crack size is important to understand the presence and progression of a fault mode, the modeling should be capable of representing crack geometries of the critical crack size as calculated by the analysis. Similarly, if pressure transients of 0.5 psi are important, the model is ineffective if it can only model transients of 2 psi.

If a CBM system design is being undertaken, selecting the most effective faults for inclusion in the effort is normally done in a selection process. From the total population of possible fault modes for all parts, components and subassemblies in the systems of the aircraft, the criticality analysis employed by RCM is used to determine which faults are important enough to equip sensors and data collection for monitoring. While fault modes which affect safety naturally rise toward the top priority for inclusion, fault modes which result in degraded availability and increased maintenance effort can also become high priority for development. The same basis for criticality in RCM analysis applies to CBM, i.e., if RCM analysis has indicated that a particular failure mode requires inspection or remediation, those same modes can be investigated for feasibility analysis for CBM. Fault modes that represent single point failures that have led to the loss of aircraft, death, or major injury are obvious candidates for investigation. Other faults that drive significant costs or readiness degradation are also strongly acceptable for CBM feasibility analysis. This feasibility analysis should include trade studies which optimize the cost (example: weight, system complexity, data collection, and processing infrastructure) for the benefit of being able to detect and diagnose the specific fault being considered. There are no fixed or rigid criteria that mandate a particular fault mode as requiring CBM application—the decision to sense and measure data to identify faults and base maintenance decisions on that information is like any other design decision that optimizes cost and risk with benefit.

The results of FMECA and fault models should be used to develop a candidate group of faults for which "features" or characteristics obtainable from signal processing of the data from sensors to detect the presence of the fault modes selected from the above FMECA are feasible. These "features" are referred to as Condition Indicators throughout this ADS. This selection process, which is application dependent, establishes the domain of the feature (example: time, frequency, wavelet) and the property of the feature (example: energy, rms value, sideband ratios) that will be employed to develop the feature (or CI) for use in fault diagnosis.

The FMECA results are also used to consider which faults require feature extraction and CI measurement in flight versus those that can be delayed until after flight. In general, the use of signal processing algorithms and software onboard the aircraft during flight should be limited to:

a. Algorithms to compute CIs for faults on components which are flight critical. Any faults for which the progression could lead to loss of the aircraft in the duration of a normal flight (different for each aircraft) are strong candidates for "onboard" processing. Further ranking of the CIs can be done through risk analysis of the fault likelihood. For example, if one fault has an occurrence of 1 per 100,000 flight hours and another 1 per 10 Million flight hours, inclusion of the former before the latter seems reasonable.

b. Algorithms to compute CIs for faults which are combat mission critical. Again, ranking within this category by occurrence factors is the most reasonable approach.

All existing data that provides sensor data responding to both normal operation and failure conditions should be consolidated in a data warehouse for use in algorithm development. Assessing the data to determine data "gaps" can provide insight into any additional testing or modeling and simulation required to support algorithm development.

Performance metrics for the Diagnostic and Prognostic modules should be established for use in the validation and verification of the diagnostic and prognostic algorithms and the maintenance actions and maintenance credits which result. Since the mathematical processes produce results which are estimates of the probability of the existence of faults and RUL, CIs and RUL confidence levels should be established. For CIs this is commonly expressed as a false alarm rate, such as 5% false alarms (detecting the existence of a fault that is not present).

The Diagnostic Module should deliver results that provide determination with high confidence of the following characteristics: Characteristics of high confidence level include:

> Detectability: The extent to which the diagnostic scheme can detect the presence of a particular fault. Detectability should relate the smallest failure signature that can be detected at the prescribed false alarm rate. Often, it is quantified by the probability of detecting the fault.
>
> Identifiability: A measure that tracks the ability of the CI to distinguish one fault from another which may have similar properties.
>
> Accuracy: A measure of how closely the CI value correlated to the severity of the fault.

Any development of CIs for use in diagnostics should include the metrics above and a validation of those metrics. Only those CIs capable of being detected with high confidence, identifiability and accuracy should be used in deployed CBM systems.

Algorithms used to preprocess the sensor data (de-noising, filtering, synchronous time averaging (STA)) compress and reduce the data necessary to extract or develop the feature or CI used to confirm the presence of a fault. The preprocessing routines, selected for the application, are intended to improve the signal to noise ratio to correspondingly improve the probability of fault detection. Best practice and experience for the specific application may develop guidelines regarding the best range of signal to noise ratio for feature extraction. If those guidelines exist, every effort should be made to develop algorithms consistent with best practice.

The sub-process labeled Detection Algorithm Development (DAD) is often an iterative process that optimizes the data compression filtering and de-noising steps to develop the most effective group of features/CIs to be used as inputs to the diagnostic process. That process can create a feature "vector" or group of individual features/CIs to be used in the diagnostic process to provide the most effective inputs to the diagnostic process. Data from actual failures or seeded fault testing, along with confirmation gained from Inspection/Tear Down Analysis (I/TDA) is used to evaluate the features and optimize their use for diagnosis. The algorithms that calculate each CI can also evaluate the value of the CI against values or "thresholds" that define the fault severity. An individual CI can be assigned values that are "normal", "marginal" (indicating potential for action such as ordering a part or scheduling a maintenance task) or "abnormal"

(indicating the need for maintenance action). Thresholds can be "hard" or single values (example: bearing energy is normal below 1.25 ips) or "variable" where a range of values is provided (example: marginal is between 3.2-3.3 ips).

Estimation of RUL should provide a confidence interval identification of the incipient fault and the fault severity which is creating the degradation. If HI values are to be used to assess fault severity, sufficient data from fault validation testing and I/TDA should exist to fully understand the relationship of HI value to fault severity and the progression of fault severity with time. HI values that are not well correlated to fault severity should not be used to estimate RUL.

Prognosis, or the estimation of RUL, forms the basis for projecting the time at which maintenance action should be taken.

Estimation of RUL through "trend analysis" of HI values is only legitimate when:

a. Data for the HIs is taken at frequent, regular intervals (application dependent based on the estimated time of failure growth).
b. HI behavior with fault progression is not cyclical or highly non-linear.

Prognosis through trend analysis should be biased to yield conservative estimates of RUL, with greater bias for cases where HI severity and failure progression data is incomplete or non-robust.

Estimation of RUL through model-based techniques is legitimate when:

a. Baseline data for normal, non-faulted operation exists
b. Baseline data for the specific serial number tracked item exists (taken within 10 hours of operation since installation).
c. Seeded Fault data exists to sufficiently describe the behavior of the fault under the normal range of operational loading.

The primary metric used to assess prognostic effectiveness is:

Accuracy[13]: A measure of how close a point estimate of failure time is to the actual failure time. Assuming that, for the ith experiment, the actual and predicted failure times are $t_{af}(i)$ and $t_{pf}(i)$, respectively, then the accuracy of the prognostic algorithm at a specific predicting time t_p is defined as:

$$ACCURACY(t_p) = \frac{1}{N}\sum_{i=1}^{N} e^{-\frac{D_i}{D_0}},$$

[13] Vachtsevanos, G., Lewis, F.L., Roemer, M., Hess, A., and Wu, B. *Intelligent Fault Diagnosis and Prognosis for Engineering Systems*. Wiley & Sons: New York, 2006.

where $D_i = |t_{pf}(i) - t_{af}(i)|$ is the distance between the actual and predicted failure times, and D_0 is a normalizing factor, a constant whose value is based on the magnitude of the actual value in an application. N is the number of experiments. Note that the actual failure times for each experiment are (slightly) different due to the inherent system uncertainty. The exponential function is used here to give a smooth monotonically decreasing curve. The value of $e^{-\frac{D_i}{D_0}}$ decreases as D_i increases, and it is 1 when $D_i = 0$, and approaches 0 when D_i approaches infinity. The accuracy is the highest when the predicted value is the same as the actual value, and decreases when the predicted value deviates from the actual value. The exponential function also has higher decreasing rate when D_i is closer to 0, which gives higher measurement sensitivity when $t_{pf}(i)$ is around $t_{af}(i)$ as in normal scenarios. The measurement sensitivity is very low when the predicted value deviates too much from the actual value. Figure C-1 illustrates the fault evolution and the prognosis, the actual and predicted failure times, and the prediction accuracy.

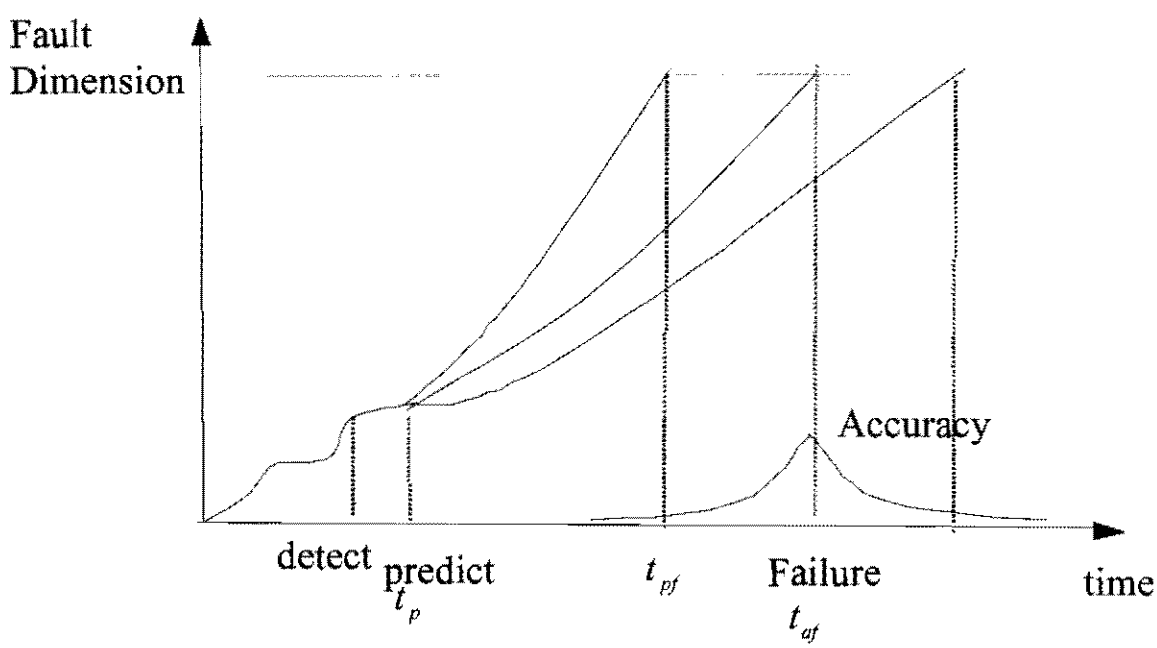


FIGURE C-1. Schematic of prognostic accuracy

Three evolution curves split from the predict time labeled t_p, which represents the time the RUL was calculated, and show 3 possible evolutions of the fault dimension. There is actually a wide range of possible failure evolutions, with a statistical distribution around the actual time to failure, labeled t_{af} as shown along the horizontal axis. The accuracy of the prognostics calculation is the highest (one) when the predicted failure time is equal to the actual failure time. Note that "failure" as defined for prognostics is not limited to the material failure of the item affected by the fault. Failure can be a limit imposed by engineering analysis that prevents

catastrophic damage or cascading failures that affect safety or repair cost. Failure can also be defined as failure to satisfy required functionality or performance.

For legacy aircraft, development of a CI can be the result of an emergent requirement, which has been identified by such actions as Accident Investigations or operational experience. In this case, the analysis and development of the CI may be pressed for time and resources. The process of defining the fault mode of interest, the sensor and sensing strategy, algorithm development, CI validation and verification, and Army wide implementation will be a dynamic and tailored process. In some cases, abbreviating the steps associated with CI development may be necessary to meet time constraints. However, even the most urgent development process should follow an organized implementation to ensure that the results are effective.

The processes related to identifying candidate CI and HI should be guided by performance of the results. Since the process of CI and HI development is data driven, there are a number of proven methods to assess the fault detection, isolation and RUL estimation performance as defined in following paragraph. Determining the CI and HI capability to discover the fault early and with high confidence, as well as providing an estimate of RUL with high confidence is essential to success for CBM.

The indicator will show significant separation between faulted conditions and healthy conditions as defined by Receiver Operating Characteristic (ROC) curve analysis or other comparable analysis. The indicator should be physically meaningful, designed to detect specific fault conditions that are named in the FMECA. The indicator should be designed to operate in an aircraft environment taking into account aircraft noise and components that would not be installed on laboratory test stands. The indicator's response should be unique for the fault mode(s) that apply to it. The indicator should not respond to external noise or other fault modes.

C.4 GENERAL GUIDANCE

C.4.1 Condition indicator (CI) and health indicator (HI) behavior. CIs and HIs included in the CBM system for a particular Army air item or Unmanned Aeronautical System (UAS) are based on the following criteria:

a. They are identified through Reliability Centered Maintenance (RCM) methods including Failure Modes Effects Criticality Analysis (FMECA) and may be categorized as:

   i. Category 1–Catastrophic: Faults that could result in death or loss of the aircraft. All Category 1 faults identified in RCM analysis should have CIs/HIs developed , unless the forecast rate of occurrence is less than 1 per 10 million flight hours and selected by the AED

   ii. Category 2–Severe: Faults that could lead to severe injury or damage to the aircraft. At least 75% of all Category 2 faults should have CI/HI coverage unless the forecast rate of occurrence is less than 1 per 1 million flight hours. The coverage should be allocated to the most frequent faults to the least frequent faults

iii. Category 3–Major: Faults that may result in damage or injury. Included only in cases where the degradation in readiness or cost exceeds thresholds determined by the PM for the aircraft. May also be included if the fault leads to cascading failures of Categories 1 and 2. Coverage for Category 3 faults should be determined from analysis of maintenance costs and readiness and selected by the PM.

b. The CIs/HIs should be explainable in physical terms, such as bearing failure, shaft misalignment or high temperature.

c. The CI/HI is identified by analysis that considers its functional role in the system as well as its physical properties. The functional analysis describes the impact of degradation or loss of the function on the rest of the component or system. This analysis may include Principle Component Analysis (PCA), a technique that reduces multi-sensor data or data from correlated variables into a smaller set of data which optimizes CI/HI sensitivity and accuracy.

d. The CI/HI is analyzed with respect to the feasibility of sensing the fault; the repeatability of gathering accurate fault data through the sensor; the relative cost or effort required to obtain the CI/HI versus its projected benefit. Any CI/HI that fails to meet these criteria should be eliminated from the development process.

e. The resulting CI/HI behavior should be mathematically definable.

f. The ideal case for a CI/HI is that it should exhibit monotonic behavior (increasing or decreasing with increasing fault size) if the value of the CI/HI is to be used to assess fault severity.

g. The CI/HI should be insensitive to extraneous factors (those unrelated to the fault origin or operational state of the aircraft) or to be corrected to account for those extraneous factors.

h. The CI/HI should be capable of detecting the fault as required by engineering analysis to ensure that the fault is detected at the minimum size specified.

i. The CI/HI should be capable of detecting the fault as required with the minimum acceptable level of false alarms and probability of detection. Typical values for false alarms are no more than 5%, depending on fault criticality.

j. The CI/HI should be uncorrelated to other CI/HI values (showing redundant behavior) unless redundancy is beneficial to system performance.

k. The CI/HI should be computationally efficient. The calculation of CIs/HIs should be able to meet requirements for timeliness and effective action by maintenance and engineering personnel. For example, computation of CI/Hl values should be able to be completed prior to the next flight of the aircraft, in order for maintenance personnel to be able to take the appropriate action to restore system operation to normal.

l. CIs/HIs which are derived from proprietary algorithms are authorized as long as:

i. Their functional description is provided to, understood by and accepted by the Government

ii. The results of the CI/HI are validated, verified and documented during the development process.

m. HIs should result in actions that restore system condition with a "first pass" success rate of at least 80%. In other words, the actions linked to the HI should restore the system to Mission Capable status 8 out of 10 times without subsequent repair for the same fault conditions.

n. HIs that combine multiple CI values can use any of the following methods (not intended to be an exclusive list), subject to validation and verification of effectiveness:

i. Weighted Averages: using weights that modify the straight CI values for criticality and severity

ii. Bayesian Reasoning

iii. Dempster-Schafer Theory: A formalized method for managing uncertainty

iv. Fuzzy Logic Inference

o. HIs that use CI values to assess system health should have a clear understanding of CI correlation to fault growth. The non linear behavior of many faults and corresponding CI values precludes the ability to base actions on simple "trend analysis" which tends to make the fault progression linear.

C.4.2 Health Indicator (HI) Usage. HIs are indicators of maintenance action based on the value of one or more CIs. The HI provides the link to the standard maintenance action contained in the appropriate Technical Manual (TM) that restores the operation of the system and aircraft to normal levels. HIs serve the function of Health Assessment (HA) in the MIMOSA Standard, as well as Advisory Generation (AG) in the International Standards Organization (ISO) Standard, as they describe the health of the system and the action to be taken to restore the system to normal. HIs should be compatible with troubleshooting and repair tasks as published in the appropriate TM.

HIs that result from ground station post flight processing should integrate with the existing maintenance and logistics information systems (See this ADS main body for additional details). This integration extends to IETMS where applicable.

C.5 APPROACH: CI/HI DEVELOPMENT FOR LEGACY AIRCRAFT

C.5.1 Initial situation

a. An existing Army aircraft system with existing vibration based data collection system.

b. The Intermediate Gearbox (IGB) on the tail boom experiences a rash of failures related to a crack on the input side of the gearbox (closest to transmission), specifically in the input bevel gear. Reference Figure C-2 for example.

c. Because of safety implications and insufficient utility of current vibration monitoring practice to detect the crack in time, the program office decides to explore developing a new or modified CI which can detect the crack more effectively, and begin to establish conservative estimates of remaining useful life.

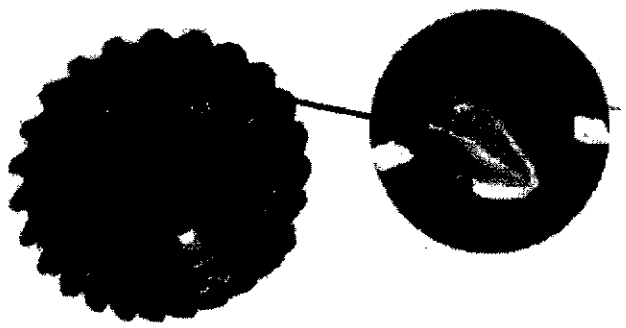


FIGURE C-2. Example of Typical failure of input bevel gear

C.5.2 CI development process. Figures C3, C4, and C5 show overview process and tools needed to develop CI/HI

C.5.2.1 Understand the failure mode. From recovery of several of the failed IGB, it appears that the failure is along the tooth of the spiral bevel gear, and that all other aspects of the input pinion assembly appear to be normal. The cracks appear to initiate near the machined edge at the root of the tooth, but review of the drawings shows that the physical dimension and method of manufacture are as specified. The cracks are initiating in the areas of greatest stress, but there are no specific manufacturing defects which require an Airworthiness Release limiting flight or recalling specific parts.

Because the failure is related to material fatigue resulting in crack propagation, there are two major ways to detect the crack: changes in the vibration sensed by the accelerometer or monitoring oil debris for pieces of gear tooth that fall away and collect in the lubrication fluid. Experience with the oil analysis program and maintenance history have shown there is relatively little operating time from the point where small bits of metal collect in the lubrication oil until the gears become so dysfunctional that loss of tail rotor thrust occurs. Clearly, detecting the crack prior to physical separation of portions of gear tooth would be more beneficial. This requires data from the accelerometer, which, while installed, may not be sampling data and recording the right data stream for use by signal processing algorithms.

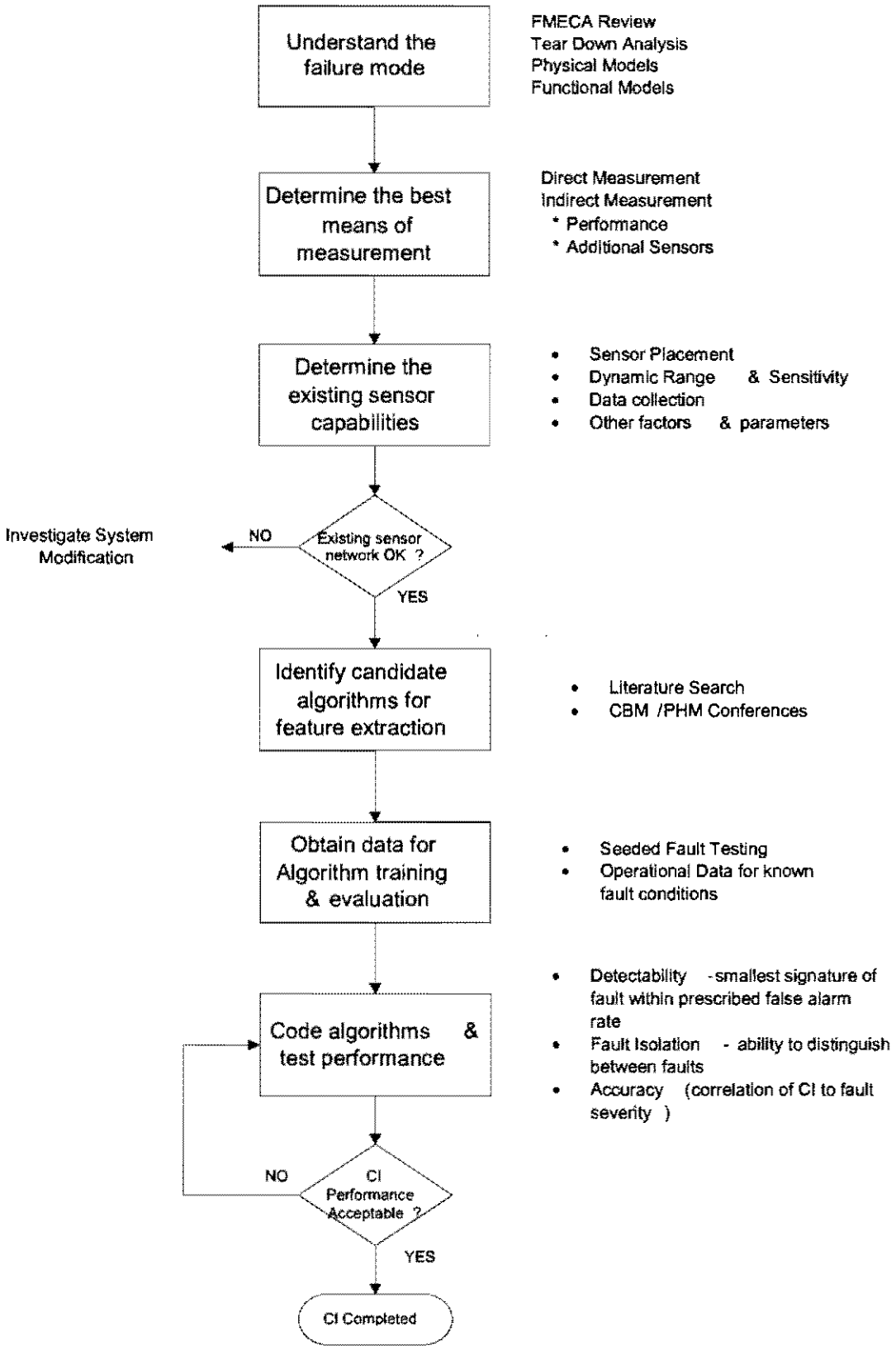


FIGURE C-3. CI development flow diagram

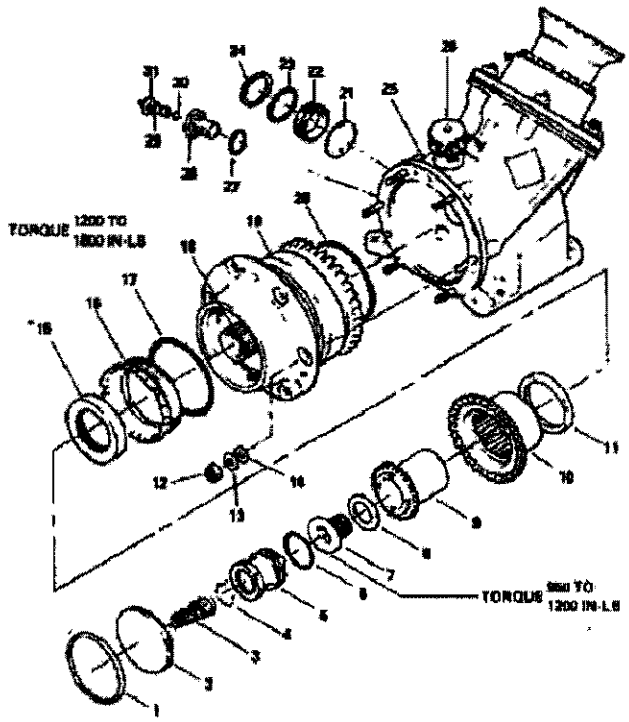


FIGURE C-4. An example of a typical schematic of intermediate gear box used to understand physical parameters

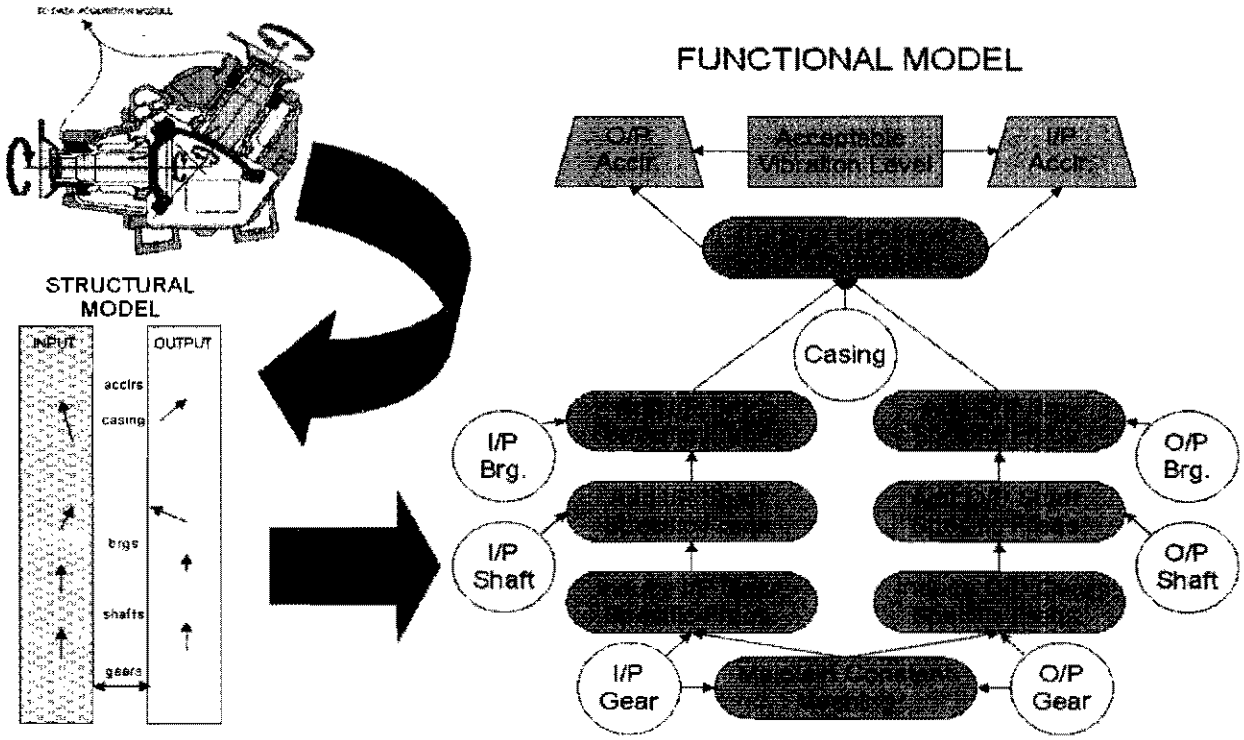


FIGURE C-5. An example of method of physical & functional modeling

C.5.2.2 Determine the best means of measurement. From a review of the physical and functional models of the IGB, engineers know that the input assembly rotates at a specific RPM or Hz, and that a crack in a single tooth would be detected on a once per revolution basis by an accelerometer with sufficient sensitivity and dynamic range.

Signal processing methods have several rules of merit with regard to data sampling. First, the data sampling frequency should typically be at least 5 times greater than the frequency range of interest. For vibration analysis of typical aircraft components, the maximum frequency for data sampling is typically between 35 and 40 kHz. It is possible to sample at greater levels during initial testing to ensure that no useful data is lost, but established systems in the field can typically be de-rated once the algorithms are developed and verified.

C.5.2.3 Determine the existing system capabilities. The helicopter has an existing vibration data collection system with the capability of sampling accelerometer data at 40 kHz. The processor and storage capacity of the Vibration Measurement Unit have the capability of storing an additional 4 mB of data, which should be sufficient for sampling data in at least 3 established flight regimes (flat pitch on the ground/flight idle, in hover and at 100 kts straight and level flight) per flight. The accelerometers are identical to those placed on the main gearbox casing, and these accelerometers have been proven capable of detecting cracks on the planetary gear assembly as well as the accessory drive shaft. Changing the software in the in flight data collection equipment is executable as a limited software release.

C.5.2.4 Identify candidate feature extraction/CI algorithms. With the large number of vibration sources on an aircraft, the data collected by any one sensor has a tremendous amount of noise. The first set of algorithms to be developed are those that can enhance the signal to noise ratio, giving the algorithms the best chance of extracting the characteristics, or features which describe the fault through sensor readings. There are a number of possible techniques for de-noising. Three popular methods are listed below (not inclusive or exclusive):

a. Soft Thresholding (Donoho, 1995)

b. Wavelet shrinking (H. Zeng, 2002)

c. Adaptive Thresholding (S. Menon, 2000)

The methods should be tested with the sample data to determine which technique works best.

The signal conditioning for feature extraction continues with some technique for signal compression that can save as much of the true "information" in the signal as possible. For vibration analysis, the most common compression technique is Synchronous Time Averaging (STA). Figure C-6 identifies an example of typical signal processing steps from data collection to CI comparison. STA is possible whenever there is a means to indicate the start of an individual revolution, by means of a pulse signal or other means. The STA takes the readings for a number of individual revolutions and averages them, resulting in an averaged data segment with a length corresponding to a single rotation. STA results enhance the vibration frequencies that are multiples of the shaft frequency.[13]

The feature or CI to be extracted from the signal is the basis for accurate diagnosis. The CI should be capable of detecting the fault prior to its causing significant damage or injury and it

should be reliable and consistent enough to merit the trust of maintenance personnel. Appendix D of ADS-79A lists a number of established CI algorithms. Engineering and scientific literature should also be searched for other promising feature extraction techniques.

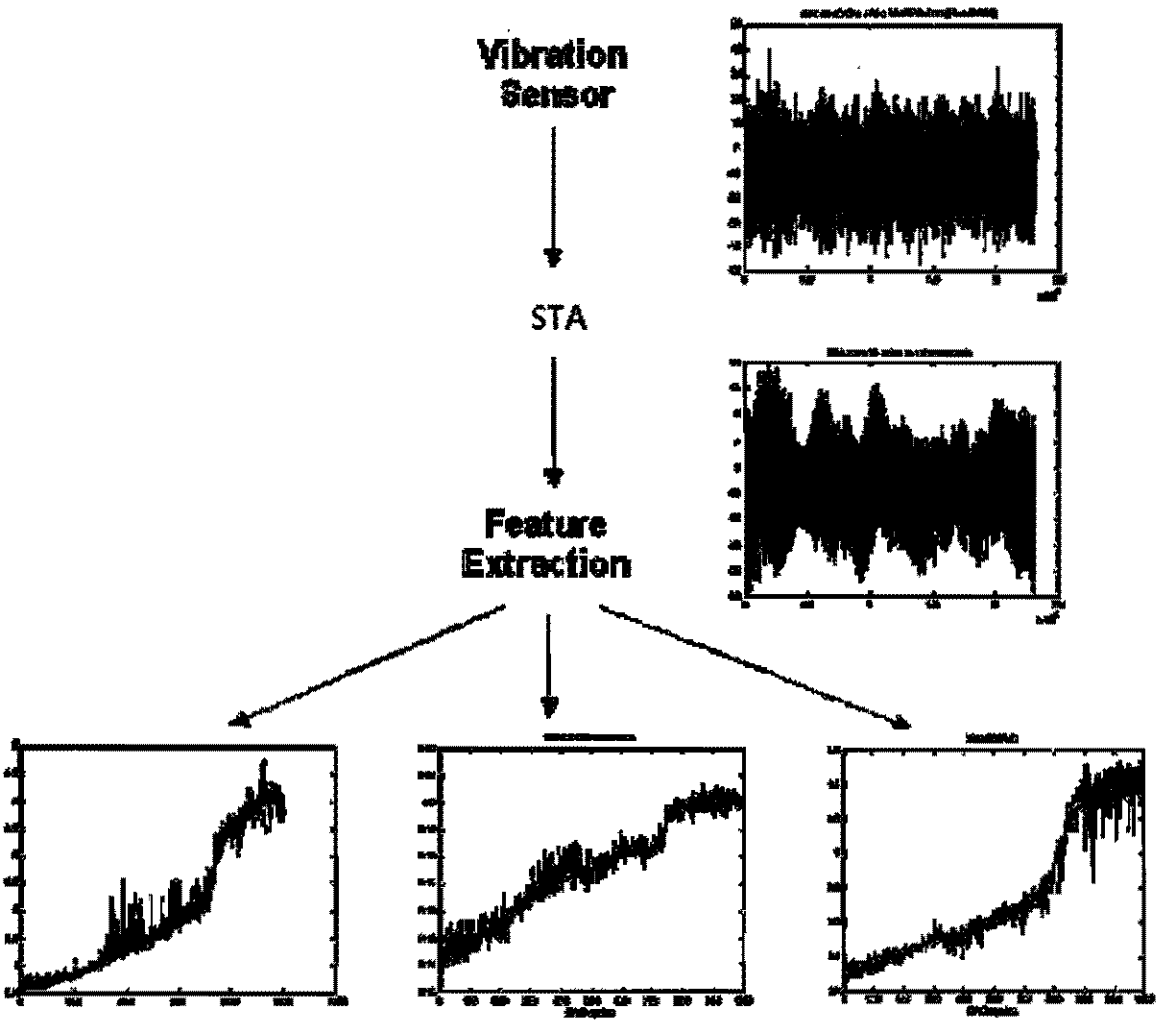


FIGURE C-6. Example of typical signal processing steps from data collection to CI comparison

C.5.2.5 Obtain data to train & evaluate the CI.

CI selection is application dependent, and the only way to ensure the CI is sufficient is to test the CI with data. In this example, we assume that technical obstacles to obtaining useful data are overcome and data sets are available for both known good IGBs and IGBs with known

faults. This data can be obtained in controlled laboratory tests, such as the test rig at the University of South Carolina, the Original Equipment Manufacturers or other service system commands and labs. Data from faulted components can be obtained from Seeded Fault Testing (See ADS-79 Appendix F) or in some rare instances, from data collected from installed systems for which a CI has not been developed (a new fault or one lower on the priority list, for example).

C.5.2.6 Code the algorithms & test performance. After selecting a number of candidate algorithms for the CI, the algorithms are converted to software, typically through the use of COTS packages such as MatLab™ or Mathematica™. These programs are easily configured to read the data files obtained in Step 5 and run through the algorithm calculations. The output of the calculations is then easily portrayed in graphs for use by the engineers and analysts in determining the performance of the algorithms. The first performance metric of interest is the accuracy of the CI, or its ability to correlate with both the existence of the fault and its increasing severity over time. In the process of obtaining data for the CIs, the testing or data collection should strive to collect the sensor data of the fault as well as the physical dimensions or other characteristics of the fault (examples: crack length, pressure drop) in order to correlate the CI value with the fault severity. Figure C-7 shows an example of such a detailed data collection. The values of the fault (crack size) are measured at specific intervals in the data collection (shown as the vertical lines in the graph to the left). It is obvious from the graphical depiction that the fault and CI exhibit closely correlated behavior. In this case, the correlation was done with a simple linear calculation.

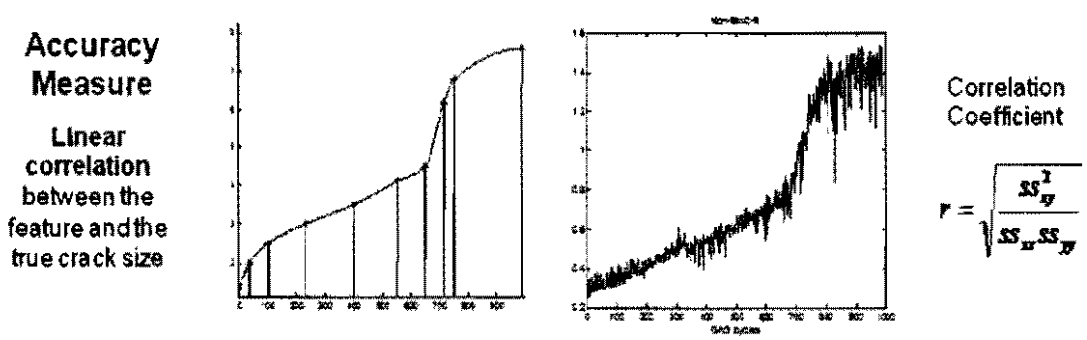
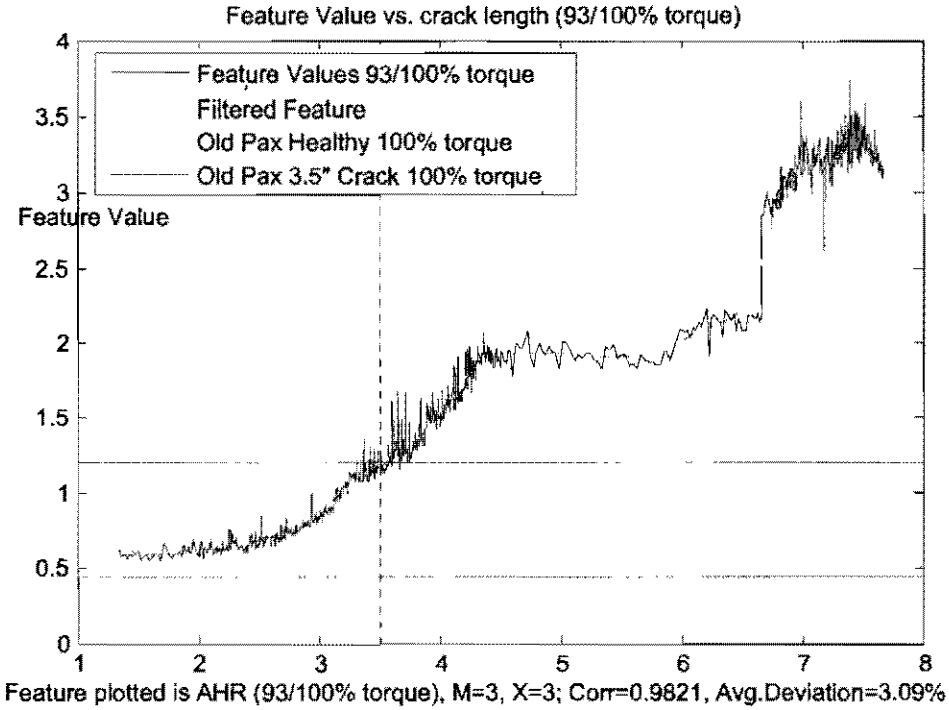


***FIGURE* C-7. An example correlation of fault dimension and CI value**

The CI should also be able to detect the fault within the limits specified by engineering analysis, and do so with a high degree of confidence. If a specific crack length is known to be the threshold beyond which catastrophic damage occurs, then the objective would be to develop a CI with the capability of detecting the crack prior to reaching that threshold value.

In the top portion of Figure C-8, the CI varies with the fault progression, but the general behavior of the CI alert would not provide a high confidence level of the fault's existence prior to reaching the threshold value (top horizontal line).



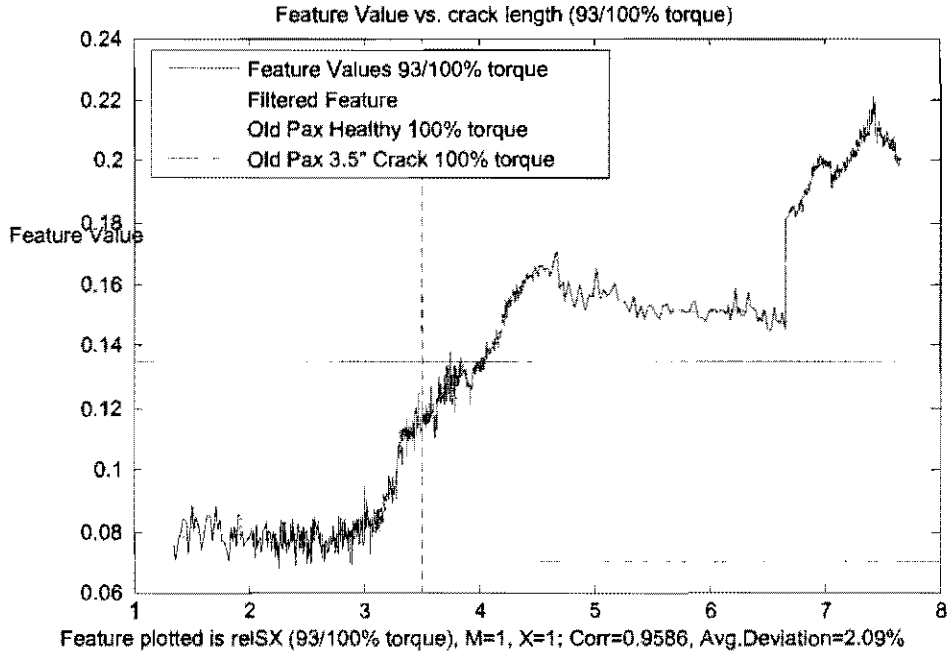


FIGURE C-8. Two examples of CI plots to compare detectability

In the bottom portion of Figure C-8, the steep increase in CI value between 3 and 4 on the horizontal axis could provide sufficient detection with high confidence. Both CIs demonstrate one reality: the fault progression may result in CI values remaining nearly constant even though the fault is growing; this is clearly not ideal, and an indication that more than one CI may be required for detection with high confidence.

For the purpose of this example, we assume that comparison of the CIs selected from Appendix D and the technical literature indicate that CI has the best available performance in detect ability, accuracy, and fault isolation (identifiability) for this particular fault.

When performance criteria are met with the sample data sets, the selection process shifts to validation of a flight qualified system. This entails the process of moving the preliminary software code from the laboratory environment to flight qualified hardware for the portions of the process to be accomplished on board, and moving the other portions of the algorithms to the 'ground station' or post flight processing portion of the system. Once the performance of the algorithms has been validated in this environment, they may proceed to implementation as directed by the aircraft program manager.

C.6 APPROACH: CI/HI DEVELOPMENT FOR NEW DEVELOPMENTAL AIRCRAFT

C.6.1 Initial situation

a. A new development aircraft which is an evolutionary design from a previous design.

b. The acquisition strategy and PM guidance mandate the use of CBM for critical systems. The requirements include a target availability of 85% and mean time to repair (MTTR) of under 3 hours.

C.6.2 CI development process

C.6.2.1 Understand the failure mode. Reliability and Maintainability studies typically allocate "not-mission–capable" fractions to various systems based on past practice, modified by new design data. Vendors supplying the new designs have some modeling and testing to substantiate R&M estimates as well as some preliminary engineering judgment regarding failure modes. From the allocation and preliminary data, some choices can be made to focus on particular components and failure modes for CBM feasibility. Again, using data from previous similar designs and experience, some estimates can be developed which model the CBM benefits and costs (weight, power, complexity). The initial design stage can then mature those estimates through Component Advanced Design (CAD) studies prior to the completed system preliminary design.

C.6.2.2 Determine the best means of measurement. From a review of the physical and functional models of the components, engineers can match the parameters to sensor requirements for sensitivity and range. These designs occur in parallel during CAD, using models and any other means to assess the effectiveness of sensor placement and to estimate the signal strength and fault feature characteristics.

C.6.2.3 Determine the design system capabilities. During CAD and subsequent design iterations, determining the system performance through modeling and potentially small scale testing can improve the CBM system design and mitigate risks of CI development in later testing phases.

C.6.2.4 Identify candidate feature extraction/CI algorithms. Candidate features can be identified through literature searches for new techniques as well as trials of previously developed work for analogous systems and fault modes (See Appendix D for examples of proven CIs for vibration based fault detection). Another approach is to use simulation and modeling. The figure below (Figure C-9) shows an approach to model based development of a CI, in this case involving a crack in a transmission subcomponent. Using finite element modeling and estimated load profiles, it is possible to develop a simulation of the fault behavior that can be used as a starting point for CI development. As in the case of data driven selection for a legacy system, it may take several iterations to develop CIs with the appropriate accuracy, detect ability and fault isolation measures.

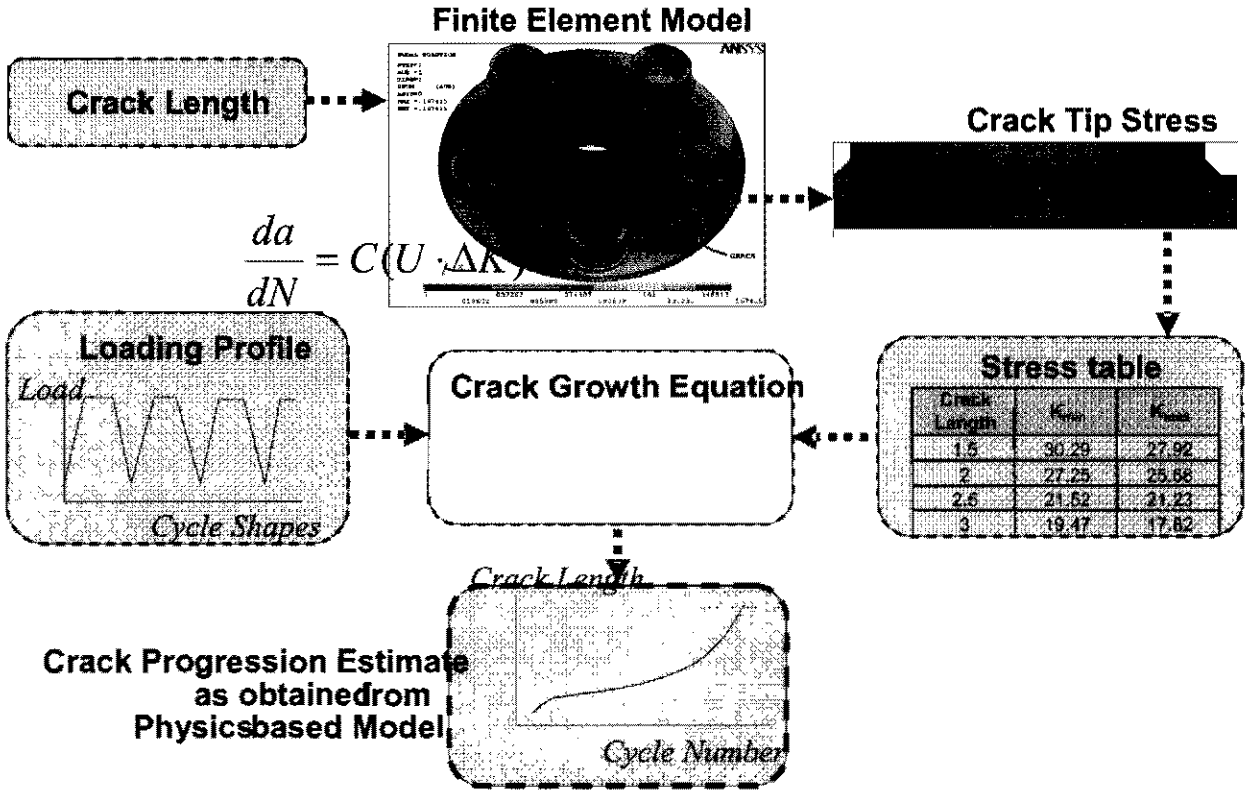


FIGURE C-9. An example of a framework for model based development of CIs

C.6.2.5 Obtain Data to Train & Evaluate the CI. The only way to ensure the CI is sufficient is to test the CI with data. In early stages of development, surrogate data from a similar component or simulated data from extensive simulation and modeling may be the only means to test the CI. As the development matures and actual devices from vendors are placed under test (or previous test data is made available), CI testing and iterative improvement is possible if sufficient time and resources are allocated to the effort.

C.6.2.6 Code the algorithms & test performance. After selecting a number of candidate algorithms for the CI, the algorithms are converted to software, typically through the use of COTS packages such as MatLab™ or Mathematica™ in the same manner as the legacy aircraft. These programs are easily configured to read the data files obtained in Step 5 and run through the algorithm calculations. The algorithms are subjected to the same analysis for accuracy, precision, detect ability and fault isolation (identifying the correct fault). This process is essentially the same for both cases.

Once performance has been validated and verified at the system level, on aircraft testing for the full system is accomplished as discussed above in the legacy case. The validation and verification process for the new development should be able to address the key metrics of availability and impact on MTTR, with some statistically reasonable approach to factor in the limited number of aircraft and flying hours accumulated during Developmental Test or Operational Test. These methods and techniques are no different for CBM systems than for any Test and Evaluation (T&E) results of other major systems on the aircraft.

A good reference article may be found as an example of CI creation process for the Apache aft hanger bearings in Figure C-10.

FIGURE C-10 EXAMPLE: CI creation process for apache aft hanger bearings

CI Creation Process for Apache Aft Hanger Bearings

A relatively simple process was followed to develop a condition indicator (CI) for the aft hanger bearings on AH-64 Apache helicopters. The resulting CI has proved to be effective in the detection of both naturally-occurring faults in the field and seeded faults on test stands. The CI development process is described here to serve as a guide for bearing CI development on other components and on other platforms.

Fault Frequency Calculation

Due to the design of a bearing, the various components (rolling elements [RE], races, and cage) of the bearing come in contact with each other at various frequencies. These frequencies are known as fault frequencies because a fault or defect in one of these components will produce a vibration at that frequency as it comes in contact with the other elements of the bearing. The four fault frequencies are the cage fault frequency (CFF), the ball spin frequency (BSF), the outer race ball pass frequency (BPFO), and the inner race ball pass frequency (BPFI). The actual frequency of a vibration produced by a fault may differ somewhat from the nominal value due to rolling elements slipping slightly rather than purely rolling.

The first step in developing an aft hanger bearing CI was to calculate the bearing fault frequencies for the bearing of interest. These frequencies can be calculated based on the geometry of the bearing and the rotational speed of the bearing. Unless the bearing separates two rotating components, the rotational speed of the bearing is simply the rotational speed of the shaft or gear to which it is attached.

AH-64 hanger bearings are single ball bearings with a fixed outer race and the following dimensions:

No. of rolling elements, N	RE diameter, d_{RE}	Pitch diameter, d_{pitch}	Contact angle,
9	0.5000 in	2.362 in	0°

Table 1: AH-64 Aft Hanger Bearing Properties

The rotational speed of the tail rotor drive shaft (ω_{shaft}) is 81.06 Hz on AH-64Ds (101%) and 80.25 Hz on AH-64A (100%). These calculations will only show the numbers for 101%, but converting to 100% is trivial.

CFF

CFF is the rotational speed of the cage. It will be less than the rotational speed of the bearing. It is designed to capture vibrations due to defects in the cage.

If the outer race is fixed,

$$CFF = \frac{1}{2}\omega_{shaft}\left[1 - \frac{d_{RE}}{d_{pitch}}\cos(\theta_{contact})\right] \quad [1]$$

If the inner race is fixed,

$$CFF = \frac{1}{2}\omega_{shaft}\left[1 + \frac{d_{RE}}{d_{pitch}}\cos(\theta_{contact})\right]$$

Using the properties of AH-64 hanger bearings,

$$CFF = \frac{1}{2} \times 81.06Hz\left[1 - \frac{0.5000in}{2.362in}\cos(0°)\right] = 31.95Hz$$

BSF

BSF is the frequency at which the rolling elements themselves rotate. It is designed to capture the frequency of vibrations produced by defects on the surface of the rolling elements. Twice this frequency is often used because if a defect strikes both races, an impact will occur twice during every rotation of the rolling element; however, the fundamental frequency is shown here,

$$BSF = \frac{d_{pitch}}{2d_{RE}}\omega_{shaft}\left[1 - \left(\frac{d_{RE}}{d_{pitch}}\right)^2 \cos(\theta_{contact})^2\right] \quad [1]$$

Using the properties of AH-64 hanger bearings,

$$BSF = \frac{2.362in}{2 \times 0.5000in} \times 81.06Hz\left[1 - \left(\frac{0.5000in}{2.362in}\right)^2 \cos(0°)^2\right] = 182.9Hz$$

BPFO

BPFO is the frequency at which rolling elements pass over a point on the outer race. It is designed to capture the frequency of vibrations produced by defects of the outer race.

$$BPFO = \frac{N}{2}\omega_{shaft}\left[1 - \frac{d_{RE}}{d_{pitch}}\cos(\theta_{contact})\right]_{[1]}$$

Using the properties of AH-64 hanger bearings,

$$BPFO = \frac{9}{2} \times 81.06Hz\left[1 - \frac{0.5000in}{2.362in}\cos(0°)\right] = 287.6Hz$$

BPFI

BPFI is the frequency at which rolling elements pass over a point on the inner race. It is designed to capture the frequency of vibrations produced by defects of the inner race.

$$BPFI = \frac{N}{2}\omega_{shaft}\left[1 + \frac{d_{RE}}{d_{pitch}}\cos(\theta_{contact})\right]_{[1]}$$

Using the properties of AH-64 hanger bearings,

$$BPFI = \frac{9}{2} \times 81.06Hz\left[1 + \frac{0.5000in}{2.362in}\cos(0°)\right] = 442.0Hz$$

Table 2: AH-64D Aft Hanger Bearing Fault Frequencies and Harmonics

Harmonic	CFF (Hz)	BSF (Hz)	BPFO (Hz)	BPFI (Hz)
1	31.95	182.9	287.6	442.0
2	63.90	365.8	575.1	883.9
3	95.85	548.7	862.6	1326
4	127.8	731.6	1150	1768

MSPU Bearing Energy CI Creation

To capture these frequencies and their first few harmonics, a CI that calculates the energy from 100 Hz to 1100 Hz, excluding the band from 152 Hz to 172 Hz, was created. The reject band centered at 162 Hz is used to exclude the second harmonic of the tail rotor drive shaft rotational speed. This shaft harmonic can be a valuable indicator of drive shaft alignment, but it is

captured by a different CI and does not provide useful information about the condition of the bearing itself. The frequencies that are captured by this bearing energy CI are highlighted in Table 2.

Normal and Faulted CI levels

The purpose of every CI is to distinguish between faulted and unfaulted components, so the effectiveness of a CI is based on its ability to separate these two populations. To determine how well a CI separates faulted cases from the healthy ones, one must first identify these two data sets. Two methods are used to make this identification: teardowns and seeded fault testing. Teardowns are used to determine the actual condition of components for which values of a CI have been calculated, usually components that are suspected of being faulted. In seeded fault testing, a component with a known fault is place on a test stand to determine how its CI values differ from unfaulted components. Since it is impractical to teardown every component, the going assumption is that the vast majority of components are unfaulted. A good CI should provide enough separation between known faulted components and the rest of the fleet that a threshold can be selected such that the known faulted components are above it and that the vast majority of the rest of the fleet is below it.

Two thresholds are commonly established for each CI. The lower threshold, or *caution* threshold, indicates that component's behavior is anomalous. Maintainers should inspect such a component and order a replacement or order a replacement. The higher threshold, or *exceedance* threshold, indicates that the component has a significant fault. Maintainers should replace such a component. The initial thresholds are set using engineering judgment and statistical analysis. As more data is collected and faults are found, the thresholds are revised to more accurately convey the condition of the component.

Figure 1 shows a section of a spectrum from a faulted AH-64D aft hanger bearing and an average of spectrums from the fleet. This is the section of the spectrum that is used for the Aft Hanger Bearing Energy CI. Note that the largest peaks in the faulted spectrum correspond to the fundamental BSF of 182.9 Hz and its harmonics. The average spectrum was calculated using 10 spectrums (or the maximum number available) from each monitored tail number.

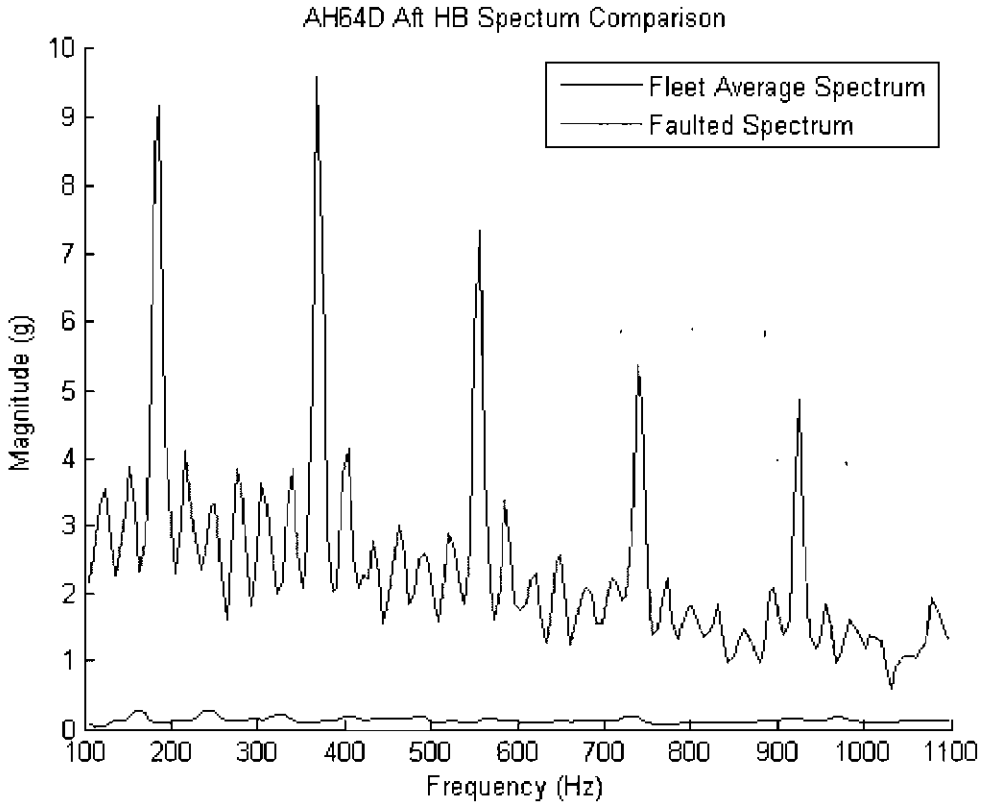


Figure 1: Comparison of AH-64D Aft Hanger Bearing Faulted Spectrum and Average Spectum

The faulted bearing that produced this data was sent to Corpus Christi Army Depot for teardown. It found that the grease was contaminated with dirt, and that spalling and corrosion pitting of one single ball initiated failure and caused secondary damage to the other balls and the races (Figure 2).

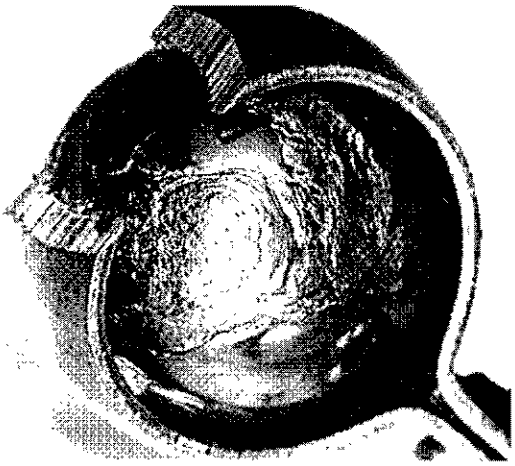


Figure 2: Damaged Ball from 01-05270 Aft Hanger Bearing

Figure 3 shows a comparative histogram for the AH-64D Survey FPG101 Aft Hanger Bearing Energy CI. The fleet data is a statistically representative sample of 6379 points and includes data from all other monitored tail numbers. The current yellow limit, 7 g, effectively separates this bearing from the rest of the fleet, and it is the only case from the fleet that has ever produced an Aft Hanger Bearing Energy CI value over the red threshold, 14 g.

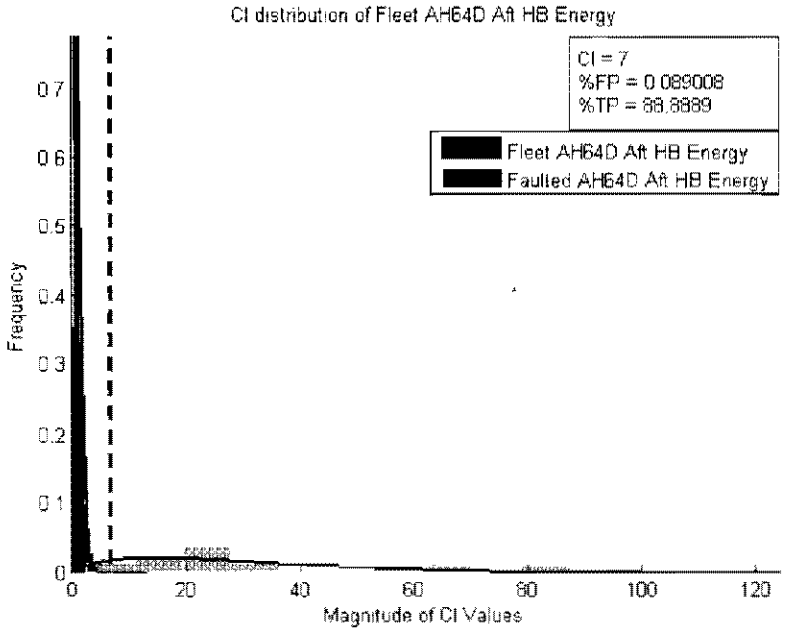


Figure 3: AH-64D Aft Hanger Bearing Energy

Figures 4, 5, and 6 show comparative histograms of the same CI from an Apache Tail Drive Train Test Stand seeded fault test. This CI effectively detected saltwater-corroded bearings and coarse sand contamination in the bearing grease, and the current yellow threshold, 7 g, provides excellent separation. The CI provided limited detection of fine sand grease contamination, and very few values were above the yellow threshold.

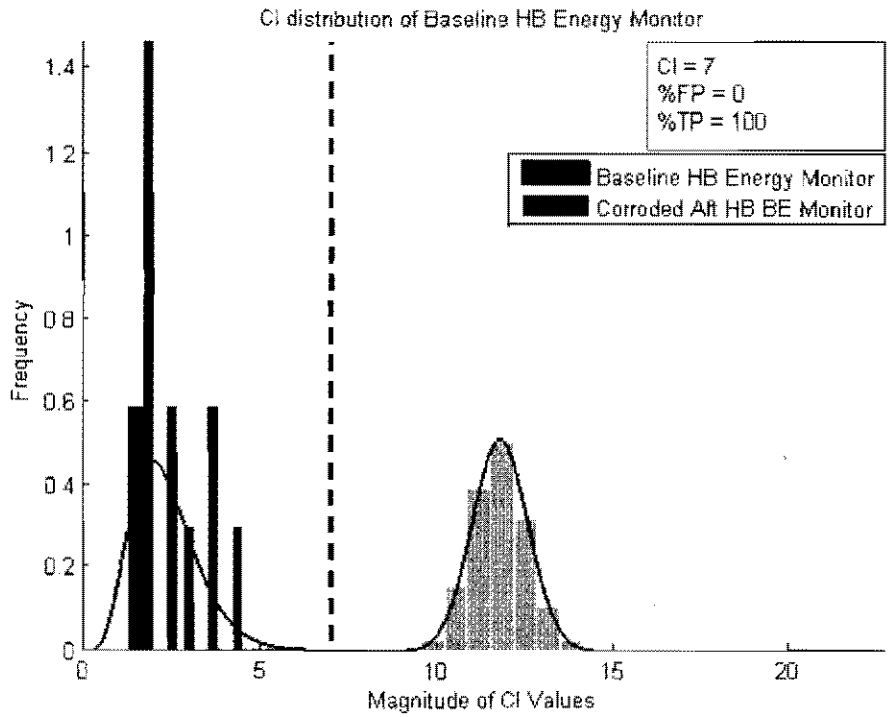


Figure 4: Aft Hanger Bearing Energy CI (Saltwater Corrosion Fault)

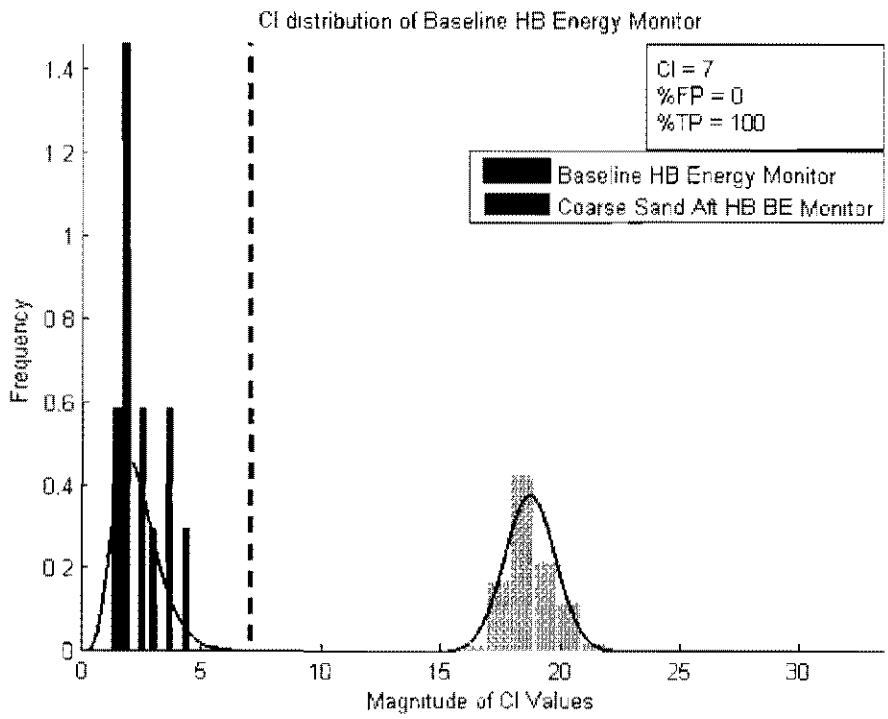


Figure 5: Aft Hanger Bearing Energy CI (Coarse Sand Fault)

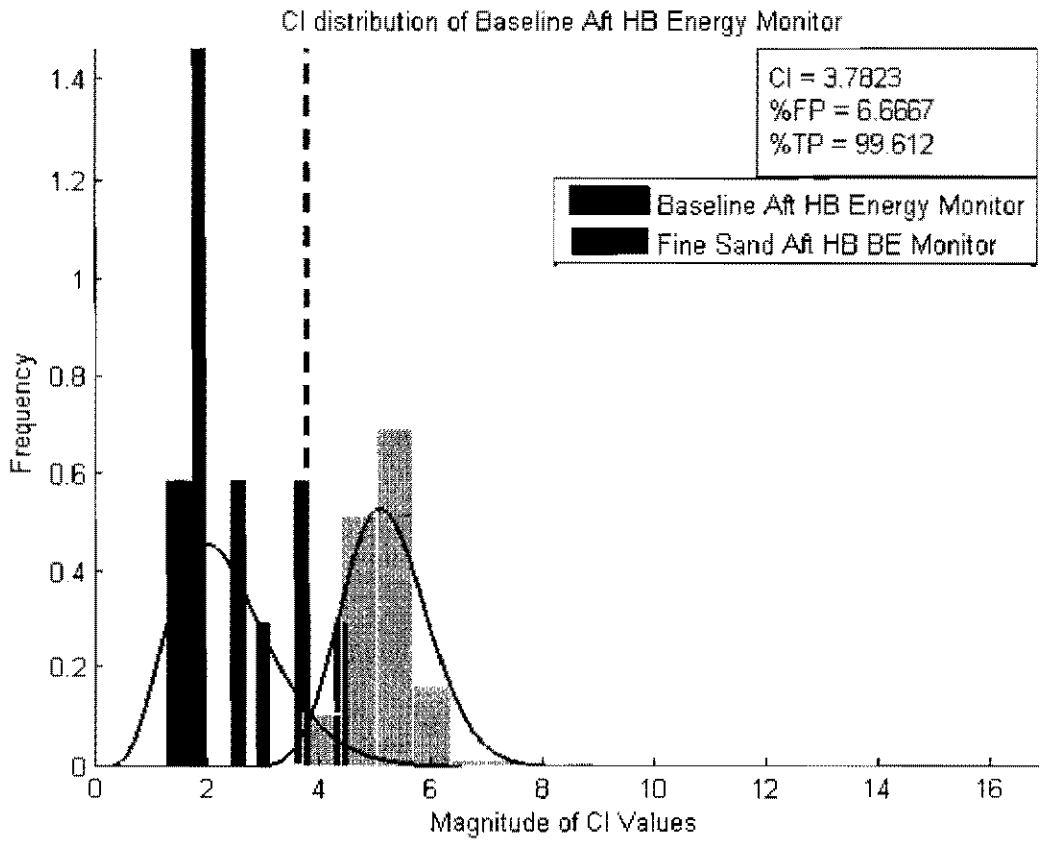


Figure 6: Aft Hanger Bearing Energy CI (Fine Sand Fault)

Summary

Bearing CI development starts with an examination of the physical properties of the bearing and the calculation of fault frequencies. Energy bands are selected based on this information, with attention paid to the frequencies of other vibration sources that should be excluded from the band. Once a band has been selected for a CI, its effectiveness must be tested and confirmed by seeded fault testing and teardowns from the fleet, or teardowns from the fleet. This approach was used to develop the AH-64 Hanger Bearing Energy CIs, and they have demonstrated their effectiveness in detecting faulted bearings.

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1. Girdhar, Paresh and Cornelius Scheffer. *Practical Machinery Vibration Analysis and Predictive Maintenance*, p. 112. Elsevier, 2004.

Appendix D:
Vibration Based Diagnostics

D.1 SCOPE

This Aeronautical Design Standard (ADS) Appendix addresses Vibration-Based Diagnostics. It covers the use of sensors, acquisition systems, and signal processing algorithms to detect, identify, and characterize faults in aircraft mechanical systems. The process involves extracting features from the vibratory data and comparing the feature characteristics to a baseline set of limits (or thresholds) which indicate the severity of a potential fault. The diagnostic algorithms should also indicate a recommended maintenance action.

Another application for vibration-based diagnostic systems is rotor track and balance, or rotor smoothing, to reduce rotor vibrations. Rotor smoothing is applicable to both the main and tail rotors. Tracking and balancing a rotor is done by adjusting weights, trim tabs, wedges and pitch link length to minimize the rotor's fundamental harmonic vibrations. Rotor smoothing is important to minimizing loads on life-limited dynamic components in the rotor system, improving aircrew human factors and reducing vibration in non-rotor system components which reduces vibration induced failures.

Vibration measurements are collected from sensors such as accelerometers at periodic intervals under specific aircraft operating conditions. For example, some diagnostic algorithms require that the data be collected while the aircraft is on the ground with blades at flat pitch and at full rotor speed. This is done to eliminate the effects of variations in aircraft loading and drive train torque on the characteristic vibration signatures. Raw vibration data from the sensors is collected in the time domain then typically transformed to the frequency domain to obtain the vibration spectrum. The vibration data may be synchronized with at least one tachometer that produces a pulse at the same rate as the fastest rotating component of interest (order ratio analysis). This synchronization process will permit effective filtration of spectral content from other components not of interest for the most accurate calculation of fault features. Features are then extracted from the spectrum and used to calculate the Condition Indicator (CI). One or more CIs may be used to calculate an aggregate Health Indicator (HI). The CIs and HIs, or HIs are then compared to thresholds to specify the component condition and maintenance status.

D.2 REFERENCES AND APPLICABLE DOCUMENTS

a. deSilva, Clarence, Control Sensors and Actuators, Prentice Hall, NJ, 1989.

b. Zakrajsek, J., Dempsey, P., Huff, E., Decker, H., Augustin, M., Safa-Bakhsh, R., Duke, A., and Grabill, P. "Rotorcraft Health Management Issues and Challenges." NASA/TM-2006-214022. February 2006.

c. CAP 753. "Helicopter Vibration Health Monitoring: Guidance Material for Operators Utilizing VHM in Rotor and Rotor Drive Systems of Helicopters." UK Civil Aviation Authority, Safety Regulation Group. June 2006. See also: <www.caa.co.uk>.

d. Ogata, K. "Discrete-Time Control Systems." Prentice Hall: Englewood Cliffs, NJ, 1987.

e. FAA AC 27-1B. "Part 27 Airworthiness Standards Normal Category Rotorcraft." FAA Advisory Circular 29-2C. 12 February 2003.

f. Roemer, M., Dzakowic, J., Orsagh, R., Byington, C., and Vachtsevanos, G. "Validation and Verification of Prognostic and Health Management Technologies." IEEEAC paper #1344. 27 October 2004.

g. Health and Usage Monitoring Metrics, Monitoring the Monitor. SAE Aerospace, Aerospace Recommended Practice ARP5783. 11 January 2008.

h. Bracewell, R.M. "The Fourier Transform and its Applications." McGraw-Hill, 1965.

i. McFadden, P.D. "Analysis of the Vibration of the Input Bevel Pinion in RAN Wessex Helicopter Main Rotor Gearbox WAK143 Prior to Failure." Aero Propulsion Report 169, Department of Defense, Defense Science and Technology Organization, Aeronautical Research Laboratories.

j. Keller, J.A., Branhof, R., Dunaway, D., and Grabill, P. "Examples of Condition Based Maintenance with the Vibration Management Enhancement Program." Presented at the American Helicopter Society 61st Annual Forum, Grapevine, TX. 1-3 June 2005.

D.3 TECHNICAL GUIDANCE.

The sensor specifications should be appropriate for the amplitude and frequency domain of the component being monitored. These specifications include its bandwidth, dynamic range, and sensitivity. With regard to signal processing, the system's sampling rate should be high enough to avoid aliasing which causes a distortion that can mask or alter a feature signature. If these parameters are not carefully matched to the component of interest, the algorithms which detect and identify the fault will not perform to the required specifications. The detection and identification algorithms themselves should be inexpensive to implement, explainable in physical terms, and be insensitive to extraneous inputs.

D.3.1 Sensor guidance. The characteristics of analog sensors include sensitivity, dynamic range, linearity, drift, and bandwidth (or useful frequency range). The following guidance is provided for sensors in a Vibration Monitoring System (VMS).

D.3.1.1 Sensitivity. Vibration sensors (accelerometers and velocimeters) should be sensitive enough to measure the smallest amplitude signal generated by an incipient fault at the threshold of detection by the diagnostic algorithm. The sensor should be able to detect this signal at the specified mounting location of the sensor. In addition, the sensor's cross-sensitivity (or "off-axis" sensitivity) should be 5% or less than the "on-axis" sensitivity.

Sensitivity is measured by the magnitude of the output signal corresponding to a unit input of the measured signal along the specified sensitive axis. It may be expressed as the ratio of the incremental output to incremental input, which is essentially a gain. Cross-sensitivity is the

sensitivity along axes that are orthogonal to the direction of the sensitive axis. High sensitivity and low cross-sensitivity are characteristics of good sensors.[14]

D.3.1.2 Dynamic Range. The dynamic range of the sensor should extend from the lowest signal amplitude required for detection to the largest expected amplitude such that the sensor signal does not saturate over the intended amplitude range of operation. If the amplitude range is dependent upon the location and orientation, or orientation at which the sensor is mounted, the determination of the required dynamic range should take this dependency into account.

The dynamic range of a sensor is determined by the largest and smallest input signals that can be detected or measured by the device. In most cases the lower limit is dictated by the amplifying electronics noise floor and the higher limit by the voltage rail used by the power supply.

D.3.1.3 Linearity. The sensor's amplitude linearity should be 1% or less of full scale. Any associated bracketry required to install the sensor on the component of interest should be considered in the measure of linearity.

Linearity is determined from the sensor's calibration curve which is a plot of the output amplitude versus the input amplitude under static conditions within the dynamic range of the sensor. The degree to which the calibration curve is a straight line is its linearity. Linearity is expressed as the maximum deviation of the calibration curve from the least squares straight-line fit of the calibration data in percent of the full scale range of the sensor.

D.3.1.4 Drift. Sensor drift should be less than 1% over the expected range of ambient operating conditions. If the sensor drift is greater than 1%, then the parameters inducing the drift should also be measured to permit compensation for the drift.

Over a period of time the characteristics of a sensor may change or drift with changes in temperature, pressure, humidity, the power supply, or with aging. Parametric drift is drift that results from parameter changes caused by instrument nonlinearities. Change in a sensor's sensitivity due to temperature changes is an example of a parametric drift.

D.3.1.5 Bandwidth. To ensure sufficient sensor response, the bandwidth or useful frequency range of the sensor should exceed the frequency range of interest for the component(s) being monitored.

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[14] deSilva, Clarence, *Control Sensors and Actuators*, Prentice Hall, NJ, 1989, pp. 51-53. [Reference not available.]

The bandwidth of a sensor is defined as the frequency range over which the magnitude of the ratio of the output to the input does not differ by more than ±3 dB from its nominal value (see Figure D-1. In the case of an accelerometer, for example, the input is acceleration while the output is volts. Thus the magnitude ratio is in the form of volts/g which varies by no more than 3 dB over its bandwidth.

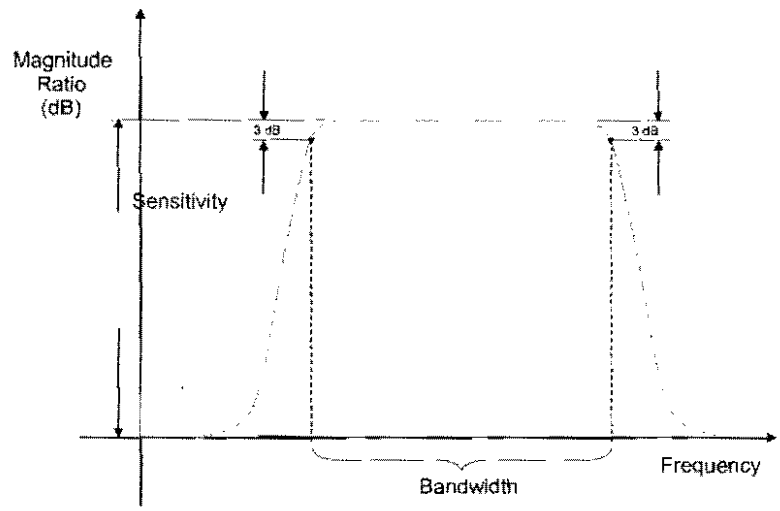

FIGURE D-1. Sensor response characteristics

D.3.1.6 Installation. Vibration sensors should be mounted as close as practical to the component(s) they are intended to monitor. In addition, they should be oriented such that their sensitive axis is aligned with the predominant axis of vibration. Each proposed mounting location should be tested (example rap test and during dynamic developmental testing) to characterize the natural structural response at the mounting location. Mounting locations should not be used when they have structural resonance frequencies that can mask the frequency modes of the dynamic components being monitored.

D.3.1.7 Built-in test capability. The VMS should have a capability for verifying the proper functioning of the sensor circuitry.

D.3.2 Data acquisition and signal processing guidance. Data acquisition deals with how frequently and under which conditions data sets are acquired. Signal processing is required to convert the sensor's analog signal to a digital signal for computation processing in the diagnostic algorithms. In addition, prior to conversion, the analog signal may require filtering to improve the signal to noise ratio, scaling to improve sensitivity, or adjustments to account for biases due to drift. Care should be taken in signal handling so as not to induce unwanted distortion of the signal.

D.3.2.1 Data acquisition conditions. Time series data should be acquired under operating conditions with the greatest signal stationarity. Stationarity denotes the consistency of

a signal's statistical properties over time. Conditions with the greatest stationarity may occur when the aircraft is on the ground with the main rotor at full speed and flat blade pitch or in the forward climb regime. [15] Collecting data under conditions of greatest stationarity minimizes the effects of loads variations on the quality of the signal. If the CI for a component requires conditions of high torque or a range of torque levels, this may affect the algorithm's ability to meet performance metrics related to false alarm rate, detect ability and accuracy.

D.3.2.2 Data acquisition frequency. At a minimum, at least one data set should be acquired for all monitored components for flights of 30 minutes or longer. This data should be acquired under stabilized conditions without the need for pilot action during the flight.[16] In addition, some components, such as high speed rotating parts, may experience a rapid onset of failure, on the order of a few hours. Data for these components should be acquired at frequent enough intervals to allow for fault detection and warning with preventative actions prior to the component's failure.

D.3.2.3 Analog to digital conversion. Range: The analog-to-digital converter (ADC) should be chosen to provide sufficient range for capturing the expected excursion in signal level without clipping. Clipping or compressing the input signal amplitude induces an artificial modulation into the measured data that can mask or alter the desired feature signature.

Resolution (Dynamic Range): The resolution of the ADC should be sufficient to detect the smallest change in the signal required by the corresponding vibration diagnostic algorithm in the presence of large amplitude background.

Resolution is the smallest change in a signal that can be detected and accurately indicated. It is usually expressed as a percentage of the maximum range of the instrument.

D.3.2.4 Sampling rate. To avoid aliasing of the sampled signal, the minimum sampling frequency (ω_s) should be at least twice as high as the highest frequency of interest (ω_I) in the signal. To preclude the influence of signal content above frequencies of interest, a prefilter should be used ahead of the sampler to modify the frequency content of the signal before it is sampled so that the frequency spectrum for $\omega > \frac{1}{2}\omega_s$ is negligible.[17]

Signal aliasing is the result of higher frequencies being folded into lower frequency signals due to the sampling rate being too low. While the minimum sampling rate is required to be twice as high as the highest frequency component present in the signal, this represents the theoretical

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[15] Zakrajsek, J., Dempsey, P., Huff, E., Decker, H., Augustin, M., Safa-Bakhsh, R., Duke, A., and Grabill, P. "Rotorcraft Health Management Issues and Challenges," NASA/TM—2006-214022, February 2006.

[16] CAP 753. "Helicopter Vibration Health Monitoring: Guidance Material for Operators Utilizing VHM in Rotor and Rotor Drive Systems of Helicopters." UK Civil Aviation Authority, Safety Regulation Group. June 2006. See also: <www.caa.co.uk>.

[17] Ogata, K., "Discrete-Time Control Systems," Prentice Hall, Englewood Cliffs, NJ, 1987, pp. 170-177.

minimum required to reconstruct the continuous signal from the sampled data. In practice, the sampling frequency is frequently chosen to be $10\omega_1$ to $20\omega_1$.

D.3.2.5 Data windowing. Digital processing is performed on a "window" of measured data that is often extracted from a continuously occurring event. Windows applied to data to prevent leakage error should be defined in the system performance specification.

Processing of a finite record length of data inherently induces a distortion, called leakage, which can perturb the feature signature and reduce the detected signal-to-noise ratio. Care should be taken in selecting a proper amplitude taper (window) to reduce these effects. Applying no window at all is to imply a rectangular window which can induce high levels of unwanted signal leakage, a redirection of the data into other spectral lines.

D.3.3 Diagnostic algorithm guidance. Vibration-Based Diagnostic Algorithms perform two basic functions: anomaly detection and fault isolation. Anomaly detection is the process of classifying the signal as either normal or anomalous. Fault isolation is the process of determining the root cause of an anomalous signal down to the component level.

As an example, if a diagnostic algorithm is intended to detect a crack of 10 mm or larger in a gear tooth, the accelerometer monitoring the transmission and its associated signal processing algorithms should be sensitive enough to measure the vibration caused by a 10 mm crack at the location at which the sensor is mounted.

The following paragraphs provide the guidance for Vibration-Based Diagnostic Algorithms.

D.3.3.1 Computational efficiency. In systems employing onboard fault state estimation the detection technique should be sufficiently computationally efficient so that all required algorithms can be executed without incurring system latencies.

In systems where processing is performed off-board the algorithms should be efficient, so that results are available in a timeframe acceptable to the maintainers making repair decisions. If the computational expense is too high for a particular algorithm, then an alternative technique should be used in order to arrive at a realizable implementation to meet the time requirement.

D.3.3.2 Physical description. The mathematical system of equations that describe the CI should be based on the Physics of Failure Modeling. In addition, the "signature feature" to which the matched filter is "tuned" for extraction should be describable with the physics of failure.

The spectral shape of a CI vibration in frequency domain should be firmly based on the Physics of Failure Characterization of the device or system. A CI selected in an ad hoc fashion based simply on historical observation without being grounded in the theoretical analysis can be risky and will ultimately lead to an implementation that is less than robust. For example, simply stating that, when a particular phenomenon is observed, it has been found experimentally that "X" is the fault and "Y" is the time to failure may not be stringent enough to yield an implementation that will work reliably in the field. The physical science behind the effect should typically be understood in order to develop a robust detection technique.

D.3.3.3 Confidence level. To ensure confidence in failure detection, CIs should be characterized by large interclass mean distance and a small infraclass variance. A class is representative of a specific failure mode or the base class of normal operation.

To meet small intraclass variance the effect should produce a signature that exhibits a parametric "clustering" in order to arrive at a matched filter that can reliably achieve a detectable signal-to-noise ratio. A feature that exhibits wide signature excursions induces a high degree of mismatch in the filter designed to extract it. A tight parametric clustering improves the confidence level in declaring a fault while a large interclass distance allows for fault classification by insuring that the feature signature will diverge from its normal operating regime as the fault progresses.

D.3.3.4 Algorithm validation. All vibration diagnostic algorithms should be validated. Algorithm's whose failure to detect the faults for which they were designed to would be hazardous to aircraft operation, and should be validated against direct evidence of a fault. Algorithms for components that are less important may be validated against indirect evidence of a fault. For both direct and indirect evidence, the whole system should be validated end-to-end. [18]

FAA Advisory Circular 29-2C (referenced above) defines "end-to-end" as intended to address the boundaries of the Health Usage Monitoring System (HUMS) application and the effect on the aircraft. As the term implies, the boundaries are the starting point that corresponds with the airborne data acquisition to the result that is meaningful in relation to the defined credit without further significant processing. In the case where credit is sought, the result should arise from the controlled HUMS process containing the 3 basic requirements for certification as follows:

1) Equipment installation/qualification (both airborne and ground)
2) Credit validation activities, and
3) Institutions for Continued Airworthiness (ICA) activities.

Direct Evidence: If failure of the vibration monitoring algorithm to detect a condition would be hazardous to aircraft operation, then direct evidence should be used to validate the diagnostic algorithm. Examples of highly critical applications include maintenance tasks such as vibration checks for imbalance/misalignment of high energy rotating equipment, fatigue life counting, or going "on-condition" for flight critical assemblies. Direct evidence of a specific fault may come from either Seeded Fault Testing or accelerated mission testing. In addition, actual field data from the entire system may be used if the detailed loading profiles are known and the parameters that are correlated with the progression of the failure are monitored.[19] Because these types of data sets may be costly to develop, they may be supplemented with data from subsystem or component rig tests.

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[18] FAA AC 29-2C. "Part 27 Airworthiness Standards Normal Category Rotorcraft" FAA Advisory Circular 29-2C, 12 February 2008.

[19] Roemer, M., Dzakowic, J., Orsagh, R., Byington, C., and Vachtsevanos, G., "Validation and Verification of Prognostic and Health Management Technologies," IEEEAC paper #1344, October 27, 2004.

Tests should be representative of the aircraft for which the credit is being sought and of test conditions representing the flight regime that would prevail when data is normally gathered (e g., cruise). Evidence gathered from on-aircraft ground trials or rig-based seeded tests should be valid for in-flight conditions.

Indirect Evidence: In less critical applications indirect evidence may be used. An example of using indirect evidence would be to analyze results from a number of potential failure modes collectively to determine the probability of an undetected failure. The failure criteria may be derived from proven analytical methods, such as finite element modeling and fracture mechanics, in conjunction with sound engineering judgment. The criteria may be validated by analogy with direct evidence gathered on other aircraft types or equipment.

D.3.3.5 False positive rate. CI and HI based maintenance actions on the aircraft should have a false alert rate of no more than 5%. A false positive is a warning that results in the unnecessary removal of a component or other unnecessary maintenance actions.

D.3.3.6 False Negative Rate. Vibration diagnostic algorithms should successfully detect at least 90% of significant (1 in 1,000,000 flight hours) failure modes occurring in the components that the system is designed to monitor. In applications where missed fault detection could be flight critical to the aircraft's operation, the missed detection rate should be no more than 1 in 1,000,000 occurrences of the fault.

D.3.3.7 Fault Isolation Rate. Once a fault has been detected, the fault should be correctly identified 95% of the time.[20] Since a component may fail in several ways, the system should identify the particular type of failure specifically within that component.

D.3.3.8 Software Development. Vibration diagnostic software should be developed, as the minimum, to the integrity level required by the system criticality assessment using RTCA/DO-178B. This system-determined level should be a result of the end-to-end criticality assessment and, in general, the same as the airborne software.

D.3.3.9 Recommended Maintenance Actions. A reliable alert generation process should be developed to advise maintenance personnel of the need to review data and determine what maintenance actions are required. Refer to Appendix C.

D.3.4 Prognostic Algorithm Guidance. Prognosis is the estimation of the time when maintenance action should be taken or when a component will fail within a specified confidence interval (see ADS paragraph 2.2, Remaining Useful Life).

D.3.4.1 Predictability. The feature to be detected and the CI that the detection updates and supports should be amenable to characterization by a mathematical function that enables

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[20] Health and Usage Monitoring Metrics, Monitoring the Monitor, SAE Aerospace, Aerospace Recommended Practice ARP5783, Jan. 11, 2008.

prediction of future condition. Prognostics based on this characterization will be updated with usage experience.

D.3.4.2 Time Horizon Guidance. Prognostic algorithms that predict the time remaining before a required maintenance action and the time until the component will fail should have time horizons of sufficient length to permit the scheduling of maintenance actions and to enhance the safe operation of the aircraft.

In some components incipient failures may be detectable only a few flight hours prior to component failure. This is particularly true of components operating under load at high rotational speeds. Consequently, vibration data acquisition for these components should be performed more frequently than for other components.

D.4 EXISTING VIBRATION BASED DIAGNOSTICS

Army aircraft mechanical systems are predominantly grouped in the engine, the drive system, the accessory subsystems, and the rotor systems. In the engine and drive system the critical faults typically include gear, bearing, and shaft failures. Accessory subsystems, such as electrical and hydraulic systems, also include components typically consisting of gears, shafts and bearings that derive power from the drive system through auxiliary gearing and shafts. The rotor system consists of main and tail rotor smoothing, or tail rotor smoothing (a.k.a. track and balance). The following paragraphs list the CIs that have been developed for the various mechanical system components.

D.4.1 Shaft Condition Indicators. Shaft CIs are mathematically simpler compared to gear and bearing CIs because the shaft faults are detected through simple harmonics of the shaft operating speed. The key indicators of shaft faults can be calculated through either asynchronous or synchronous means, using a synchronous time average (STA). The following is a non-exhaustive list of CIs for shaft faults that are proven both on test stands and in the field environment:

- Asynchronous Shaft Order ½ (SO½)
- Asynchronous Shaft Order 1 (SO1)
- Asynchronous Shaft Order 2 (SO2)
- Asynchronous Shaft Order 3 (SO3)
- Synchronous Shaft Order ½ (SO½)
- Synchronous Shaft Order 1 (SO1)
- Synchronous Shaft Order 2 (SO2)
- Synchronous Shaft Order 3 (SO3)
- STA RMS
- STA Peak to Peak
- STA Kurtosis

D.4.2 Shaft Balancing and Rotor Smoothing. Shaft balancing and rotor smoothing algorithms are required procedures. Shaft balance is typically accomplished with a magnetic or optical tachometer along with an accelerometer mounted close to the shaft coupling. Rotor smoothing is typically accomplished with an optical blade tracker, accelerometers mounted in the airframe, and magnetic tachometers.

D.4.2.1 Shaft Balance Techniques. Shaft balancing procedures are required on some aircraft platforms. The system may use permanently installed accelerometers to monitor the

condition of shafts throughout the drive train, especially shafts operating at very high frequencies (greater than 200 Hz). An example would be the engine output shaft.

Small mass imbalance on a high frequency shaft induces high vibration levels that can be destructive to the surrounding equipment, potentially causing the catastrophic loss of the aircraft. Shaft balance is achieved using a combination of the shaft condition indicators and balancing algorithms. The system should be capable of using linear balance coefficients and applying basic shaft balance techniques.

D.4.2.2 Rotor Smoothing Techniques. Rotor smoothing is required on all aircraft systems and is an essential maintenance operation. The system may use optical blade trackers to minimize blade track split and accelerometers mounted near the swashplates or in the cockpit in conjunction with a tachometer to reduce once per revolution (1/R) vibration.

Rotor smoothing is accomplished in a step-by-step procedure that involves ground or hover track and lateral balance, and forward flight vibration smoothing. Rotor smoothing algorithms should provide maintainers rotor adjustments such as pitch link changes, hub or blade weight changes, wedges and trim tab changes specific to each aircraft type. Once per revolution (1/R) vibration should be reduced at the most common ground, hover, and forward flight regimes. For aircraft with 4 rotor blades, track should be minimized to reduce the potential for split track conditions typically associated with twice per revolution (2/R) vibration. Rotor smoothing should be accomplished in an average of three flights following phase maintenance.

D.4.3 Bearing Condition Indicators. Bearing faults are typically associated with the rolling elements, cages, and races which make up the bearing and their associated fundamental fault frequencies. Faults also appear as increases in energy bands. In current practice, there are two distinct methods for calculating CIs that use energy based algorithms. The methods differ in their use of an enveloping technique.[21,22] The following CIs are for bearings:

- Envelope Ball Energy
- Envelope Cage Energy
- Envelope Inner Race Energy
- Envelope Outer Race Energy
- Envelope Tone Energy
- Envelope Base Energy
- Envelope High Frequency Energy (15 – 20 kHz)
- Peak Pick
- Frequency Band Energy

[21] Bracewell, R.M. "The Fourier Transform and its Applications", McGraw-Hill, 1965. [out of print.]

[22] McFadden, P.D. "Analysis of the Vibration of the Input Bevel Pinion in RAN Wessex Helicopter Main Rotor Gearbox WAK143 Prior to Failure" Aero Propulsion Report 169, Department of Defense, Defense Science and Technology Organization, Aeronautical Research Laboratories.

D.4.4 Gear Condition Indicators. The following CIs are laboratory proven on gear test stands operated by various commercial and Government organizations.[23]

- Residual Kurtosis
- Residual RMS
- Sideband Modulation
- Narrowband Crest Factor
- Gear Distributed Fault
- G2-1
- Residual Peak to Peak
- Energy Operator
- Sideband Index
- Sideband Level Factor
- FM0
- FM4 & FM4*
- Energy Ratio
- M6A & M6A*
- M8A & M8A*
- NA4 & NA4*
- NA4 Reset
- Amplitude Modulation
- Phase Modulation
- Instantaneous Frequency
- NB4 & NB4*
- NP4

[23] Vachtsevanos, G., Lewis, F.L., Roemer, M., Hess, A., and Wu, B. Intelligent Fault Diagnosis and Prognosis for Engineering Systems. Wiley & Sons: New York, 2006.

Appendix E:
Data Integrity

E.1 SCOPE

This Aeronautical Design Standard (ADS) Appendix establishes the guidance for ensuring the Integrity of Data Collection and Storage as a component of any Condition Based Maintenance (CBM) system.

E.2 APPLICABLE DOCUMENTS. The documents listed below are not necessarily all of the documents referenced herein, but are those needed to understand the information provided by this handbook.

The following specifications, standards, and handbooks (available at < www.rtca.org>) form a part of this appendix to the extent specified herein.

(a) RTCA DO-178B. "Software Considerations in Airborne Systems and Equipment Certification." 1 December 1992.

(b) RTCA DO-200A. "Standards for Processing Aeronautical Data." 28 September 1998.

(c) RTCA DO-278. "Guidelines for Communication, Navigation, Surveillance, and Air Traffic Management (CNS/ATM) Systems Software Integrity Assurance." 5 March 2002.

(d) RTCA Report: "Future Flight Data Collection Committee Final Report." Issued 4 December 2001.

In addition to these documents, Section 2.1.1 of the basic ADS (of which this is Appendix E) contains others that have general pertinence to the CBM process and should be reviewed.

E.3 DEFINITIONS

E.3.1 Data Availability. Data Availability refers to the provisions taken to ensure that the data is available to the maintenance user at the time of need. These provisions include the use of a reliable delivery mechanism as well as storage media.

E.3.2 End-to-end. This term is used within the context of this appendix to mean encompassing the mechanisms from the point at which the data is collected (acquired) to the point in which the data is destroyed including transmission, computation, storage, retrieval, and disposal.

E.3.3 Data security. Data Security refers to the provisions taken to ensure that the data is protected from corruption by malicious acts.

E.3.4 Data reliability. Data Reliability refers to the assurances that the data can be used for its purposes in the CBM system as a result of steps taken to ensure its integrity and availability.

E.3.5 Data integrity. Data Integrity refers to the assurances that the data is unchanged (missing or corrupted) from when it was initially acquired by the CBM system.

E.3.6 Data verification. Data Verification refers to the steps taken to confirm the integrity of data retrieved from a storage system. These techniques include the use of hash functions on data read-back or the use of a Message Integrity Code (MIC0) or Message Authentication Code (MAC).

E.3.7 Data reduction. Data Reduction refers to any action taken to reduce the volume of the measured data without compromising the value of the data with regard to its intended purpose. Data reduction is often performed as part of the acquisition process in order to reduce the burden on storage capacity and may be broadly interpreted to actions ranging from down sampling (volume reduction) to filtering (smoothing).

E.3.8 Data mining. Data Mining refers to reviewing or processing the data in order to obtain information or knowledge. Depending on the format of the stored data, this process can range from signal processing of sampled measurements to queries performed on database tables.

E.4 GENERAL GUIDANCE

CBM systems require the processing and storage of digital data in both aircraft onboard and ground station systems. This data is used to make often critical maintenance decisions regarding the airworthiness and remaining useful life (RUL) of the vehicle, its subsystems, assemblies, and components, or components and therefore, should be trustworthy. This appendix describes the system end-to-end design practices to be used to ensure the integrity, reliability, and security of CBM flight data from its onboard acquisition to its ground station storage and usage.

Precautions should be taken at each stage of a CBM system implementation as data integrity can be compromised at any point in the chain from acquisition to storage and retrieval for use. Corruption and loss of data, or corruption or loss of data may occur during:

a. Acquisition
b. Onboard computation
c. Transmission
d. Storage
e. Retrieval and use

In addition, the loss of data integrity may be either inadvertent or the result of willful malicious attacks and, therefore, care and handling should include prudent practices that guard against both forms of corruption and loss.

The degree to which data integrity should be ensured is ultimately governed by the severity of the resulting failure or malfunction being prevented by the CBM system. The failure event

severity is graded in accordance with the criticality levels prescribed by RTCA DO-178B.[24] The higher the criticality of the failure event being prevented, the more stringent the processes and procedures are to ensure that lack of data integrity is not the cause of poor performance by the CBM system.

E.5 SPECIFIC GUIDANCE

E.5.1 Criticality. The measures and procedures taken to ensure data integrity in an airborne CBM system should be determined by the resultant severity of the safety effects caused by a compromise in data integrity. The severity of effects should be determined in accordance with the guidance provided in RTCA DO-178B Section 2.2.1 on Failure Condition Categorization (FCC). These levels are defined as:

a. Catastrophic: Failure conditions which would prevent continued safe flight or landing.

b. Hazardous/Severe-Major: Failure conditions which would reduce the capability of the aircraft or the ability of the crew to cope with adverse operating conditions to the extent that there would be:

i. A large reduction in safety margins or functional capabilities,

ii. Physical distress or higher workload such that the flight crew would not be relied on to perform their tasks accurately or completely, or

iii. Adverse effects on occupants including serious or potentially fatal injuries to a small number of those occupants.

c. Major: Failure conditions which would reduce the capability of the aircraft or the ability of the crew to cope with adverse operating conditions to the extent that there would be, for example, a significant reduction in safety margins or functional capabilities, a significant increase in crew workload or in conditions impairing crew efficiency, or discomfort to occupants, possibly including injuries.

d. Minor: Failure conditions which would not significantly reduce aircraft safety, and which would involve crew actions that are well within their capabilities. Minor failure conditions may include, for example, a slight reduction in safety margins or functional capabilities, a slight increase in crew workload such as routine flight plan changes, or some inconvenience to the occupants.

e. No Effect (Non-hazardous class): Failure conditions which do not affect the operational capability or safety of the aircraft, or the crew's workload.

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[24] RTCA DO-178B: Software Considerations in Airborne Systems and Equipment Certification.

Criticality may be determined by performing a Functional Hazard Assessment (FHA). The FHA may be a preliminary document to the Preliminary Safety Assessment (PSA) or a part of the PSA. The FHA is a top down analysis that starts with the hazards to the aircraft and traces these hazards to the system, subsystem, and component level in the areas affected by the CBM system.

For each topic in the following subsections, prevention of corruption and loss, or corruption or loss should be mandatory for data in which failure of that facet of the CBM system could result in Catastrophic, Hazardous/Severe-Major, or Major consequences. The prevention of corruption and loss of data, or corruption or loss of data should be recommended for data in which failure of that facet of the CBM system could result in Minor consequences. No special recommendations on data integrity are made in data for which the failure of the CBM system has no effect. Note, however, the mandated guidance does not preclude implementing a conservative practice which is more stringent than that required to meet the criticality requirement. For example, a design may include password protection and perform routine storage backup of data used in making maintenance decisions on aircraft systems whose failure would not result in catastrophic safety events.

E.5.2 Data acquisition. Data corruption and loss, or corruption or loss may occur during collection at the point of data initiation; therefore, the necessary precautions should be taken to ensure that data is protected during acquisition. For example, as part of an aircraft onboard data collection system, these precautions will take the form of proper shielding from electromagnetic interference (EMI) in the vicinity of an analog, electrical sensor. Also, any action performed as part of the acquisition process in an effort to reduce the volume of collected data should not compromise the data with respect to its purpose in the CBM system. For example, data should be pre-filtered and sampled at appropriate rates in order to avoid aliasing and prevent distortion. Also any filtering or smoothing should not mask features or characteristics.

In most CBM systems persistent data will ultimately reside in a relational database. Further data acquisition will occur at the ground station as technicians access the data and annotate the records with maintenance actions taken; therefore, the appropriate input protection should be implemented to ensure data integrity. For example, a good data acquisition design will incorporate the use of a finite number of selectable options, where possible, as opposed to operator-typed entries. For operator-typed entries the CBM system should perform input data validation in the form of error checking against the defined data schema before presenting input to the database. This would include testing for operator input correctness and completeness, such as preventing entry of a character where a numeric is expected. In addition, the system will perform the appropriate rejected item handling for improper operator entries.

In addition to the user interface of the CBM system software, the Relational Database Management System (DBMS) should be used to ensure data integrity. Data integrity is enforced in a DBMS through the use of integrity constraints and database triggers. An integrity constraint is a declarative method of defining a rule within the DBMS for the column of a table. Examples of integrity constraints are:

a. Null Rule: Columns (fields) will disallow INSERTs or UPDATEs to rows (records) containing a NULL (absence of a value) entry.

b. Primary Key Rules: Column (field) is identified for containing a "primary key" value that is unique to each row (record). Data entries are disallowed for INSERTs and UPDATEs to rows (records) containing non-unique primary key fields.

c. Relational Integrity Rules: A rule defined on a key (column or set of columns) in one table that guarantees that the values in that key match the values in a key in a related table (the reference value). Referential integrity also includes the rules that dictate what types of data manipulation are allowed on referenced values and how these actions affect dependent values. An example of a referential integrity rule is "Set to Default" where when referenced data is updated or deleted, all associated dependent data is set to a default value.

A database trigger is an integrity enforcement rule that refers to a set of database procedures which are automatically invoked on INSERT, UPDATE, or DELETE query operations. Trigger functions performed by the DBMS serve to augment the input testing performed by the user interface of the application software. They are capable of performing more complex tests of the input fields in the course of a database transaction than a simple integrity constraint.

E.5.3 Data computation. Data corruption and loss, or corruption or loss may occur during computation; therefore, the design should incorporate the necessary precautions to ensure that data is protected during data processing. Typically, integrity tests conducted as part of data processing involve the implementation of "traps" within the application software for error and exception handling. These software traps will include tests for zero divide as well as the improper operator entry and input rejection due to the integrity constraints and database triggers in data acquisition.

Computational data integrity tests will incorporate "try" software blocks (or their syntactic equivalent, depending on software language) for accessing a relational database. In addition to trapping integrity tests, "try" blocks ensure that data is not overwritten while being simultaneously accessed by multiple users in the ground station.

E.5.4 Data transmission. Data corruption and loss, or corruption or loss may occur during transmission; therefore, the design should incorporate the necessary precautions to ensure data integrity during aircraft onboard and off-board data transmittal. This, for example, will range from EMI shielding of cables used to transmit analog data to procedures for ensuring the integrity of digital information transmitted over a data bus. Digital transmission procedures will range from the use of embedded checksums to the use of error correcting codes for recovering corrupted data. Unrecoverable data lost in the course of transmission may be resolved with protocols such as automatic re-transmission and transmit/receive handshaking.

E.5.5 Data storage. Data corruption and loss, or data corruption or loss may occur during storage; therefore, the design should incorporate the necessary precautions to ensure data integrity during aircraft onboard and off-board storage.

In addition, the design should incorporate proper database administration (DBA) procedures and policies to ensure stored data integrity. These procedures should include the use of routine system-wide data backups performed by the database administrator to prevent catastrophic data

loss. Also, the database administrator should perform routine maintenance using a set of database consistency check (DBCC) queries. These queries will include relational integrity checks that identify and fix orphaned records, confirm known record counts within tables, and identify and resolve the existence of multiple primary keys within damaged tables.

E.5.6 Security. In addition to accidental data corruption and loss, or corruption or loss during storage, data integrity may be compromised as a result of malicious attacks on the CBM system. Therefore, the proper design should ensure that security measures and procedures are implemented to prevent the willful, malicious destruction of maintenance data. These measures should include the implementation of either or both physical security and logical security. Physical security refers to the physical placement of the data storage system in a secure area where only authorized administrators have access. Logical security refers to the implementation of user passwords or other authentication for data access. User passwords offer the ability of implementing a layered security by allowing different levels of access, including the ability to change or delete data, to different users.

E.5.7 Data retrieval. Data corruption and loss, or corruption or loss may occur during data retrieval; therefore, the design should incorporate the necessary precautions to ensure data integrity during data recall from storage and use. For example, modifications to the originally acquired data on retrieval and use should be documented with a date stamp before being returned to storage.

E.5.8 Data mining. Stored data may be called upon at any time in its lifecycle for processing to obtain information about the observed event. Depending on the nature of the stored data, this could involve filtering of sampled measurements or queries of records in a database of processed measurements. Therefore, the data should be oriented and formatted in a manner that allows access to the variety of authorized Army maintenance and analysis systems (see FIGURE E-1).

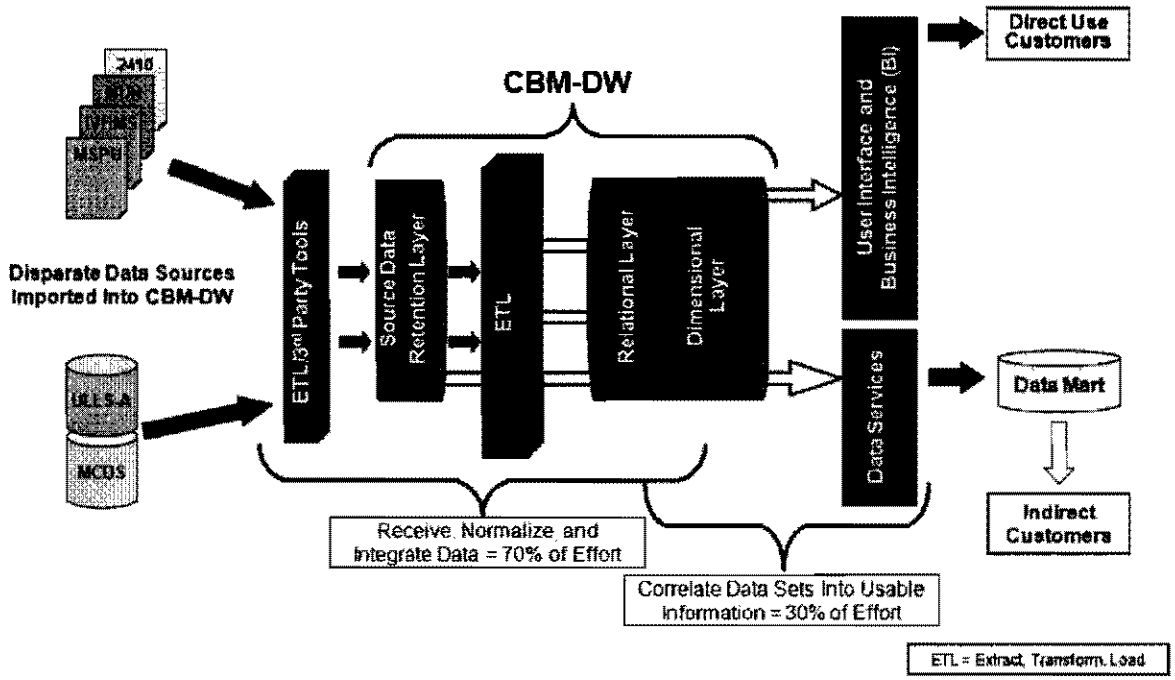

FIGURE E-1. Data orientation and formatting

However, as discussed as part of Data Retrieval (E.5.7), measures should be taken to insure that data is not lost or corrupted as a product of data analysis. For example, the data storage system may limit data mining to being performed on a copy of the archived data while retaining the original in order to guarantee integrity.

E5.9 Data error correction and notification. Steps should be taken to provide information that ensures that data is traceable back to the source. Traceability information provides a record of any actions/changes made to the data from acquisition to end user and is used to determine the causes of data errors. If data errors occur at any point in the chain from acquisition to retrieval, an error correction and notification process should be employed. Users should be informed if there are suspected errors in the data and a process that corrects errors at the source of the errors should then be exercised.

Appendix F:
Seeded Fault Testing

F.1 SCOPE

This Aeronautical Design Standard (ADS) Handbook Appendix provides guidance for the development and performance of component Seeded Fault Testing programs for the purposes of validating the accuracy and robustness of condition indicators (CIs) and health indicators (HIs) used as part of a condition based maintenance (CBM) system.

F.2 APPLICABLE DOCUMENTS.

The documents listed below are not necessarily all of the documents referenced herein, but are those most useful in understanding the information provided by this document.

The following references form a part of this appendix to the extent specified herein.

(a) "Spiral Bevel Pinion Crack Detection in a Helicopter Gearbox", NASA Glenn Research Center, US Army Research Laboratory, H.J. Decker, D.G. Lewicki, June 2003.

(b) "Inserted Fault Vibration Monitoring Tests for a CH-47D Aft Swashplate Bearing", US Army RDECOM, J.A. Keller, P. Grabill, June 2005.

(c) "CBM Test Requirements," US Army Aviation Engineering Directorate (AED) Condition Based Maintenance (CBM) Office, June 2009.

F.3 DEFINITIONS

Probability of Detection (P_D): The probability that a true fault signature is detected by the CBM sensors. For CBM aircraft systems, the target probability of detection is 90% for both condition and health indicators; however, this target value may be increased or decreased pending the level of criticality associated with the fault.

Probability of False Positive (P_{FP}): The probability that a sensor detects a fault that is not found by inspection. For CBM systems, the target probability of false positive is 10% for both condition and health indicators; however, this target value may also be increased or decreased pending the level of criticality associated with the fault.

Probability of a False Negative (P_{FN}): The probability that a sensor fails to detect a fault that is found by inspection. P_{FN} is equal to one minus P_D, and P_{FN} and P_{FP} are inversely related. For CBM systems, the target probability of false negative is 10% for both condition and health indicators; however, this target value may also be increased or decreased pending the level of criticality associated with the fault.

Component Failure: In the context of this appendix, component failure may refer to either "complete" or "near" failure. "Complete" failure is defined as the condition in which the article under test can no longer perform its intended function and may happen as either a slow progression or a sudden, catastrophic event. "Near" failure is defined as the point where the component under test reaches a degraded condition where complete failure is imminent.

F.4 GENERAL GUIDANCE

Test stand Seeded Fault Testing provides a means to acquire the empirical information needed to verify the fault indication(s) in support of on-aircraft CBM validation. In measuring and observing a component in a controlled laboratory environment as it degrades to failure, condition and health indicators can be tested for their ability to reliably and accurately recognize fault signatures.

Fault testing can be used for a variety of reasons. One purpose could be to down select among a candidate list of sensors. Another purpose of fault testing could be to tune selected sensors for achieving an acceptable tradeoff between probability of a false positive (P_{FP}) and the probability of a false negative (P_{FN}). Furthermore, Seeded Fault Testing should be used to demonstrate that fault signatures and their detection by CIs are suitably insensitive to variations in test specimen and operating environment. CIs should deliver consistent results across all available test specimens over the full range of expected on-aircraft operating conditions (examples: temperature, vibration). To consider variability of fielded aircraft, CIs should also be tested on multiple aircraft. CIs should not be tuned to the degree that they are tailored for specific test configurations.

Note that laboratory testing may confirm that some failure modes and fault conditions are not reliably detectable by measured indicators and should not be transitioned to a CBM system. An essential benefit of laboratory Seeded Fault Testing is the ability to accurately measure the rate of failure progression (crack growth) and the corresponding changes in measured indicators. Laboratory testing may reveal that an impending fault may not exhibit any measurable indication prior to complete failure, and therefore, it may also not be a good CBM candidate.

Seeded Fault Testing involves all of the steps normally associated with the aircraft part qualification testing. Figure F-1 and reference F.2(c) outline example seeded fault testing and qualification processes used by the Army.

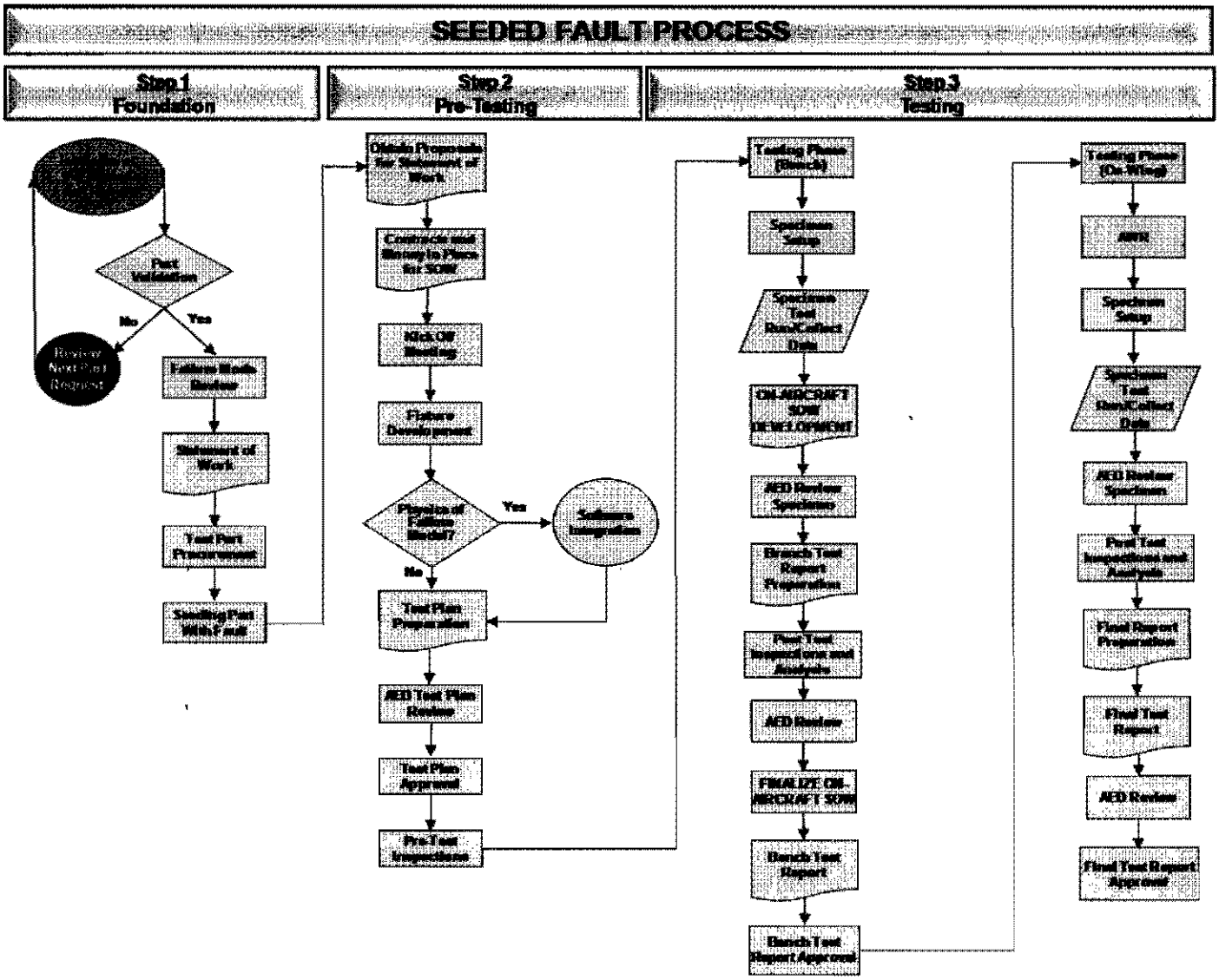

FIGURE F-1. Example seeded fault testing and qualification process

As shown in the figure, the process is organized into four general steps:

Step 1: Foundation – Initial test planning begins by determination of the goals and objectives of the seeded part experiment clearly defined in a Statement of Work (SoW) for the effort. In addition, this step also includes acquiring and preparing the test specimen.

a. **Failure mode review** – The test planning review should clearly identify, (1) the fault under test, and (2) the indicators being evaluated. The laboratory nature of seeded fault testing allows for the careful isolation of a specific failure condition without interference from other fault conditions. Also, the decision should be made in this initial test planning and review stage as to whether the component should be taken to complete or near failure in the course of the test. It is acceptable to define end of test at component near failure when the article under test reaches a condition where it is no longer safe to operate in the test fixture.

b. **Seeding the part with a fault** – The test should be provided with a specimen that will progress to failure and in such a way that the rate of degradation can be accurately measured during the course of the experiment. Introduction of a specimen which is degraded or deformed in a known or controlled way will ensure that only the desired fault condition will occur during the test and that failure will occur within a reasonable test timeframe. The part may be seeded either by manually deforming the part in the laboratory, for example, scoring or cutting the part in order to induce a crack at a desired location or, by accepting a part returning from the field which is worn or deformed in a way which will ultimately result in the desired fault condition under the induced stress of the laboratory test setup.

i) Failure can be classified as a complete failure if the article under test can no longer perform its intended function. This can happen in a slow progression or quickly as in a component failure. The point at which it is possible to detect the fault will also determine how much time is remaining before progressing to a complete failure. If it is not observed early enough it may not be an appropriate CBM candidate.

ii) A second classification of failure can be referred to as a near failure or when a failure is imminent. This occurs when the component under test reaches a point when it is no longer safe to operate in the test fixture. Safe operating limits are imposed on the test fixture to ensure that the test article does not cause harm to equipment or personnel.

iii) It should also be noted that prior to introducing a faulted component on an aircraft for on-aircraft testing, the safe-life of the component (time to failure) should be accurately measured in the laboratory. This will provide some assurance that the test article will not progress to a component failure during this test phase.

Step 2: Pre-Testing – Test planning continues with evaluation of vendors and final vendor selection. The vendor should clarify all test objectives before initiating test fixture development.

a. **Test fixture development** – The laboratory test jig should be configured to induce enough stress to produce the desired fault condition in the seeded (deformed or worn) test part. Typically, the test stand should be designed to simulate on-aircraft operating conditions so that fault progression and condition indicator fault tracking can proceed as it would in a normal environment. However, at times it may be necessary to exceed normal component operating conditions in order to achieve a reasonable time limit on the experiment. It is important though that the test conditions do not call for operation outside of the safety limits of the test fixture. Test design should not call for exceeding test stand operating thresholds which might expose equipment or personnel to a safety risk. Also, if possible, automated monitoring equipment should be designed into the test fixture to maintain continuous, real-time observation and monitoring of not only the condition indicators but damage progression in the test specimen.

b. **Physics of failure model** – A complete analysis would include development of a physics of failure model. A rigorous mathematical characterization of the experiment enables a complete post-test analysis of all observable fault symptoms. In addition, the modeling effort could help explain any unexpected, observed failure phenomenon encountered during the test.

c. **Test plan review / approval** – The overall Seeded Fault Test should be reviewed by peers prior to execution. This review should again confirm the expected, controlled fault to be induced in the part as well as the manner in which it is generated in the laboratory. Therefore, the review should cover both selected test specimen and the configured test fixture as well as any conducted pre-test analysis, such as physics of failure.

d. **Pre-test inspections** – Both the test specimen part and the test fixture should be carefully inspected prior to test start. It should be confirmed that the test part is of acceptable quality and that the controlled deformity is the only compromise to integrity so that the part will fail as expected in the test. Also, a final inspection of the text fixture should be performed to insure that it will operate properly over the entirety of the test and that it will impose the controlled stress needed to induce the expected fault.

Step 3: Testing – The testing phase proceeds from specimen setup through conducting the experiment to documenting results in a test report. Ideally, the seeded fault laboratory experiment should be followed by confirmation with on-aircraft testing of the implemented CBM approach derived from the laboratory experiment. This step, referred to as on-aircraft testing, allows for a proposed CBM approach demonstrated in the laboratory to be monitored in a normal operating environment before fielding the technique on aircraft.

a. **Specimen setup** – All minor modifications and servicing to the specimen should be made before installation in the fixture to minimize interruption of the test run. For example, the part should be cleaned prior to installing in the jig to allow for better test observation.

b. **(Bench) specimen test run / collect data** – On completion of all pre-test analysis, review and setup, the seeded test specimen should be stressed until the fault condition is reached. If available in the test fixture design, the automated monitoring equipment should be used to maintain a continuous observation of the test specimen condition; however, it may also be acceptable to periodically stop the test to perform visual inspection.

c. **(Bench) test report preparation** – On completion of all specimen test runs, a bench test report should be created to document all observed events and findings of the laboratory experiment. The findings should include summary conclusions concerning the detectability of the fault, as well as the general impression of the condition and health indicator's ability to reliably detect and track the phenomenon. The report should also document the original condition of the test specimen, test fixture, and all pre-test analysis.

During the laboratory bench testing phase, plans should be developed to conduct on-aircraft testing of the proposed CBM technique. The purpose of on-aircraft testing is to confirm that the implemented technique is robust enough to detect the fault and monitor fault progression in the noisy environment of normal aircraft operation. Following a review of the bench testing results and the decision to validate the CBM hardware on aircraft, an Airworthiness Release (AWR) should be developed to allow limited testing of the CBM hardware on a specific number of aircraft. The on-aircraft testing is essentially a repeat of the bench testing with a seeded specimen placed in a test aircraft for evaluation. Data is again collected and evaluated with a test report documenting the results of the experiment.

For the on-aircraft test, a specimen should be chosen that has already reached or is very close to reaching the desired failure condition. This will allow for a reasonable amount of normal operation to induce the failure and provide data for again evaluating the condition or health indicator's ability to measure fault progression. Because a failed or compromised component part is being introduced into the airframe, the on-aircraft testing should be conducted with ample consideration given to vehicle and operator safety. In fact, the earlier laboratory testing should provide as accurate an estimate as possible as to the remaining safe-life (time to complete failure) for the specimen prior to installation on the aircraft. This will provide some assurance that the test article will not progress to a component failure during the on aircraft test. However, to provide meaningful results the test should obtain fault and indicator data over the full range of aircraft operating regimes. Naturally, therefore, the aircraft testing should be performed as part of an experimental flight with a trained test pilot.

Step 4: Follow-on efforts: Pending the conclusions and results found in the bench and on-aircraft test reports, an AWR may be generated to alert the fleet as to any changes required in a fielded CBM system. Depending on the purpose and intended goals of the Seeded Fault Test, this information can range from the introduction of a new condition or health indicator; retirement of an existing, prescribed condition or health indicator; or change in threshold value of an indicator for inspection or replacement of a part.

F.5 SPECIFIC GUIDANCE

To further illustrate and provide specific guidance in executing the seeded fault test process, example references are cited in Section F.2, Applicable Documents, of this appendix. While these examples do not specifically utilize the Figure F-1 process, both of these references are good examples of where following a rigorous experimental process led to obtaining a conclusion as to the effectiveness of a CBM technique.

In the NASA / US Army Research Laboratory study on crack detection[25], thirteen vibration-based diagnostic metrics were compared for their ability to detect tooth fracture and progression to tooth separation in a spiral bevel pinion of a Bell OH-58 main rotor gearbox. The specific fault condition under test was identified, and the test specimen was prepared by manually placing a notch into the fillet region of one spiral bevel pinion tooth using electro-discharge machining (ultimately, trial and error determined the minimum notch size used to induce the intended fault). The test specimen was installed in an OH-58 transmission and mounted in a Helicopter Transmission Test Stand at NASA's Glenn Research Center. Bench testing commenced with the pinion operated at the design speed and at various percentages of maximum design torque, with the overall goal of the testing to initiate a crack in the pinion at the lowest possible torque. Three metric indicators proved sensitive enough to detect the damage while not being overly sensitive to torque fluctuations. The other diagnostic metrics either could not reliably detect the fault condition or were too noisy in their indications to be used as a viable fielded solution.

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[25] "Spiral Bevel Pinion Crack Detection in a Helicopter Gearbox", NASA Glenn Research Center, US Army Research Laboratory, H.J. Decker, D.G. Lewicki, June 2003.

The US Army RDECOM report "Inserted Fault Vibration Monitoring Tests for a CH-47D Aft Swashplate Bearing"[26], involving the detection of swashplate bearing failure in a CH-47D, offers an acceptable alternative to Reference 21 for obtaining a test specimen for initiation of seeded fault testing. In Reference 22, heavily worn, used components returning from field operation with the fleet were hand-selected by researchers at the Corpus Christi Army Depot (CCAD). The parts were inspected and selected for their anticipated ability to produce the desired fault condition in the laboratory test stand. These defective bearings, therefore, provided a natural source alternative to manually degrading a new part.

The intention of these reference articles was to document the methods and results of laboratory seeded fault testing; therefore, follow-up, on-aircraft seeded fault testing or the need for an AWR was not addressed by the articles. It would be expected, however, following the example process guidance in Figure F-1, that, in situ, on-aircraft seeded testing be used to validate any laboratory findings before issuing a flight/fielding AWR for CBM system on Army aircraft.

[26] "Inserted Fault Vibration Monitoring Tests for a CH-47D Aft Swashplate Bearing", US Army RDECOM, J.A. Keller, P. Grabill, June 2005.

FIGURE F-2 EXAMPLE: Spiral bevel pinion crack detection in a helicopter gearbox.

Spiral Bevel Pinion Crack Detection in a Helicopter Gearbox

Harry J. Decker
Harry.J.Decker@grc.nasa.gov
Mechanical Engineer

David G. Lewicki
David.G.Lewicki@grc.nasa.gov
Aerospace Engineer

US Army Research Laboratory, Cleveland, Ohio

ABSTRACT

The vibration resulting from a cracked spiral bevel pinion was recorded and analyzed using existing Health and Usage Monitoring System (HUMS) techniques. A tooth on the input pinion to a Bell OH-58 main rotor gearbox was notched and run for an extended period at severe over-torque condition to facilitate a tooth fracture. Thirteen vibration-based diagnostic metrics were calculated throughout the run. After 101.41 hours of run time, some of the metrics indicated damage. At that point a visual inspection did not reveal any damage. The pinion was then run for another 12 minutes until a proximity probe indicated that a tooth had fractured. This paper discusses the damage detection effectiveness of the different metrics and a comparison of effects of the different accelerometer locations.

INTRODUCTION

Since 1988, the NASA Glenn Research Center has been working on improving gear damage detection using vibration monitoring. Most of the effort has focused on pitting and other surface distress failures. Later, the testing expanded into oil debris monitoring-based HUMS, vibration-based crack detection, and data fusion. Gear cracks, although potentially more catastrophic, are much less common, thus more difficult to study.

There have been several studies [1-6] to determine the onset of a gear tooth fracture in a helicopter gearbox. Some of these studies have been planned and others have been the result of unplanned faults. There have been few attempts to detect a fracture at its onset and then simulate a mission profile in order to determine the remaining life of the component.

There have been studies on gear fault detection for a spiral bevel pinion [7-9]. These studies have primarily focused on the surface contact mode of failure (pitting). The higher contact ratio of a spiral bevel pinion makes the detection of a small fault even more difficult. Some argue that the metrics that are readily available are sufficient to detect, and even in some cases, predict the remaining life of the gear.

The objective of this study was to evaluate vibration-based diagnostic metric to detect gear crack initiation. To accomplish this, seeded fault tests were conducted using a helicopter main rotor transmission. Various over-torque conditions were run to facilitate crack initiation. A visual inspection was performed before each change in torque.

FAULT DETECTION METHODS

Thirteen metrics that are available in the open literature were evaluated in this study. They were applied to the vibration signals of a relatively simple helicopter main rotor gearbox.

All of the diagnostic techniques discussed in this paper require time synchronous averaging. Time synchronous averaging has two desirable effects: (1) it reduces the effects

of items in the vibration signal that are not synchronous with shaft and mesh frequencies; (2) because of this, the amplitudes of the desired parts of the signal are effectively amplified relative to the noise.

A once per revolution tachometer pulse is required to synchronize different parts of the vibration signal. The tachometer signal is used to divide the digitized vibration signal into blocks representing exactly one revolution of the gear being studied. The beginning and end data points are interpolated to provide more accurate and consistent averages. Each block's data record is then interpolated to provide a convenient number of equally spaced points (typically a power of two, such as 1024) for the feature detection and extraction process. By interpolating each revolution into an equal number of points, slight changes in the rotational speed can be accommodated. Since each point in the signal now refers to the same angular position for all the sampled rotations, the blocks are simply averaged. A simple linear average is used since experience has shown that the interpolation method is not significant [10].

The traditional methods of gear failure detection methods are typically based on some statistical measurement of vibration energy. The primary differences are based on which of the characteristic frequencies are included, excluded, or used as a reference [11].

Root Mean Square

The root mean square (RMS) is defined to be the square root of the average of the sum of the squares of an infinite number of samples of the signal (Equation 1). It is also sometimes referred to as the standard deviation of the signal average. For a simple sine wave, the RMS value will be defined to be approximately 0.707 times the amplitude of the signal.

$$\mathrm{RMS} = \sqrt{\frac{1}{N}\left[\sum_{i=1}^{N}(S_i)^2\right]} \qquad (1)$$

Crest Factor

The Crest Factor (CF), shown in Equation 2, is calculated by dividing the maximum positive peak value by the RMS value of the signal [12]. This makes the metric a normalized measurement of the amplitude of the signal. A signal that has a few, high amplitude peaks would produce a greater Crest Factor as the numerator would increase (high amplitude peaks), as the denominator decreases (few peaks means lower RMS).

$$\mathrm{CF} = \frac{S_{0\text{-}pk}}{\mathrm{RMS}} \qquad (2)$$

Energy Operator

For the Energy Operator [13], the input signal for each point in time is squared and the product of the point before and after is subtracted. In the case of the endpoints, the data is looped around. Specifically, when calculating the first point, use the last point and vice versa. The normalized kurtosis of the resultant signal is then taken and reported as the energy operator.

Kurtosis

The kurtosis (Equation 3) is simply the normalized fourth moment of the signal [14]. The moment is normalized to the square of the variance of the signal. The kurtosis is a statistical measure of the number and amplitude of peaks in a signal. That is, a signal that has more and sharper peaks will have a larger value. A Gaussian distribution has a kurtosis value of very nearly three. It should be noted that some investigators subtract 3 from this calculated value.

$$\mathrm{Kurtosis} = \frac{N\sum_{i=1}^{N}(S-\bar{S})^4}{\left[\sum_{i=1}^{N}(S-\bar{S})^2\right]^2} \qquad (3)$$

where

S	signal
$\bar{S}$	mean value of signal
i	data point number in time record
N	number of data points

Energy Ratio

Heavy uniform wear can be detected by the energy ratio [12]. The difference signal (d) is the resultant signal after the regular meshing components (r) (mesh and harmonic frequencies) are removed. It compares the energy contained in the difference signal to the energy contained in the regular components signal. The theory is that as wear progresses, the energy is moved from the regular signal to the difference signal (Equation 4).

$$\mathrm{ER} = \frac{\mathrm{RMS}_d}{\mathrm{RMS}_r} \qquad (4)$$

M6

The M6 metric [15], shown in Equation 5, is a continuation of the kurtosis. In this particular case, it is the sixth moment that is used. It is normalized in a similar manner as the kurtosis, except that the variance now has to be raised to the third power. In general, the characteristics of the spread of the distribution show up to be even (as opposed to odd) functions of the statistical moment. The odd functions relate the position of the peak density distribution with respect to the mean.

$$\mathrm{M6} = \frac{N^2\sum_{i=1}^{N}(d-\bar{d})^6}{\left[\sum_{i=1}^{N}(d-\bar{d})^2\right]^3} \qquad (5)$$

where

- d difference signal
- $\bar{d}$ mean value of difference signal
- i data point number in time record
- N number of data points

FM4

The FM4 vibration diagnostic metric (Equation 6) is one of the most popular metrics used [16]. This metric detects changes in the vibration resulting from damage limited to several teeth. The FM4 metric is non-dimensional and is calculatcd by dividing the fourth statistical moment about the mean by the square of the variance of the difference. As long as damage propagates locally, the FM4 metric will increase. When damage starts becoming generalized, the value decreases.

$$FM4 = \frac{N\sum_{i=1}^{N}(d_i - \bar{d})^4}{\left[\sum_{i=1}^{N}(d_i - \bar{d})^2\right]^2} \qquad (6)$$

where

- d difference signal
- $\bar{d}$ mean value of difference signal
- N total number of points in time record
- I data point number in time record

NA4

The NA4 metric (Equation 7) was developed to overcome a shortcoming of the FM4 metric [11]. As the oecurrences of damage progresses in both number and severity, FM4 becomes less sensitive to the new damage. Two changes were made to the FM4 metric to develop the NA4 metric as one that is more sensitive to progressing damage. One change is that FM4 is calculated from the difference signal while NA4 is calculated from the residual signal. The residual signal includes the first order sidebands that were removed from the difference signal. The second change is that trending was incorporated into the NA4 metric. While FM4 is calculatcd as the ratio of the kurtosis of the data record divided by the square of the variance of the same data record, NA4 is calculated as the ratio of the kurtosis of the data record divided by the square of the average variance. The average variance is the mean value of the varianee of all previous data records in the run ensemble. These two changes make the NA4 metric a more sensitive and robust metric. The NA4 metric is calculated by

$$NA4 = \frac{N\sum(r_i - \bar{r})^4}{\frac{1}{M}\sum_j\left[\sum_i(r_{ij} - \bar{r}_j)^2\right]^2} \qquad (7)$$

where

- r residual signal
- $\bar{r}$ mean value of residual signal
- N total number of points in time record
- M current time record in run ensemble
- i data point number in time record
- j time record number in run ensemble

NB4

The NB4 metric is the time-averaged kurtosis of the envelope of the signal that is bandpass filtered about the mesh frequency [17]. An estimate of the amplitude modulation caused by the sidebands of the mesh frequency, is calculated using the Hilbert Transform. The Hilbert transform creates a complex time signal in which the real part is the bandpassed signal and the imaginary part is the Hilbert transform of the signal.

NA4*

As damage progresses from being localized to distributed, the variance of the kurtosis increases dramatically. Since the kurtosis is normalized by the variance, this results in the kurtosis decreasing to normal values even with damage present. To counter this effect, NA4* was developed [18]. While the kurtosis for a data record is normalized by the squared average variance for the run ensemble for NA4, with NA4* the kurtosis for a data record is normalized by the squared variance for a gearbox in good condition. This is a change in the trending of the data and was proposed to make a metric that is more robust as damage progresses.

In order to estimate the variance for a gearbox in good condition, a minimum number of data records of a run ensemble is chosen to ensure a statistically significant sample size. The variance of the residual signal for all data records is calculated, as well as the mean and standard deviation. The mean is used as the current estimate of the variance for a gearbox in good condition. When the next data record is available, a judgment is made as to whether to include that data record as representative of a good gearbox. A gearbox with damaged gears will have a larger variance that one in good condition. The decision is based on an upper limit L (Equation 8), which in turn is dependent on the choice of a probability coefficient Z, and is calculated by

$$L = \bar{x} + \frac{Z}{\sqrt{n}}\sigma \qquad (8)$$

where

- $\bar{x}$ mean value of previous variances

Z value for a normal distribution

σ standard deviation of previous variances

n number of samples ($n \geq 30$)

The value for the Z parameter can be found in introductory statistics books. If the current variance exceeds this limit, then it is judged that the gearbox is no longer in "good" condition and the previous estimate of the variance is used for the remainder of the run ensemble. If the variance for the new data record does not exceed this limit, then the new data record is included into the data representing the gearbox in good condition.

The decision of what probability coefficient is chosen is based on many factors. The most difficult trade-off is that of Type I or Type II errors. A Type I errors is an undetected defect. A Type II error, on the other hand, reports damage when none is present. The choice of the probability coefficient is a compromise between having too many Type II errors and not detecting damage.

M6*

This metric is based on the M6 metric with the exception that it includes the averaging effect of NA4* and the variance comparison present in the denominator.

FM4*

The diagnostic metric FM4*metric is, like NB4*, the addition of the run ensemble averaging and the statistical limitation of the growth of the square of the variance. The calculation of the numerator of this metric remains the same as in FM4. The denominator has the averaging effect of NA4*, and also determines if the current variance is of sufficient probability to be contained in the previous samples.

NB4*

The diagnostic metric NB4* is the addition of the run ensemble averaging and the statistical limitation of the growth of the square of the variance first introduced in the development of NA4*. The calculation of the numerator of this metric remains the same as in NB4. The denominator does have the averaging effect of NA4*, and determines if the current variance is of sufficient probability to be contained in the previous samples.

EXPERIMENT CONFIGURATION

OH-58 Main Rotor Transmission

The OH-58 is a single-engine, land-based, light, observation helicopter. The helicopter serves both military (OH-58 Kiowa) and commercial (Bell Model 206 Jet Ranger) needs. The design maximum torque and speed for the OH-58A main-rotor transmission (Figure 3) is 350 N-m (3100 in-lb) input torque and 6060 rpm input speed [19]. This corresponds to 222 kW (298 HP). The transmission is a two-stage reduction gearbox. The first stage is a spiral bevel gear set with a 19-tooth pinion that meshes with a 71-tooth gear. Triplex ball bearings and one roller bearing support the bevel-pinion shaft. Duplex ball bearings and one roller bearing support the bevel-gear shaft in an overhung configuration.

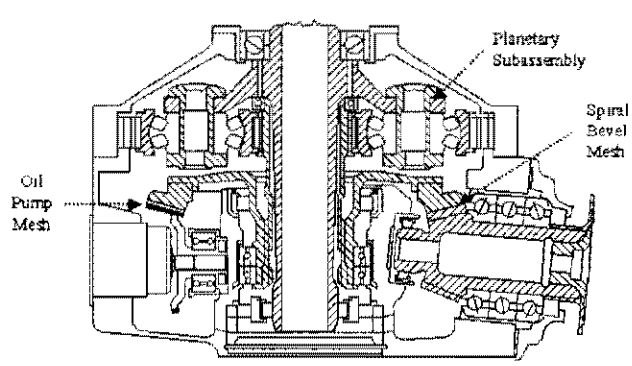


Figure 3. OH-58 Main Rotor Transmission

A planetary mesh provides the second reduction stage. The bevel-gear shaft is splined to a sun gear shaft. The 27-tooth sun gear drives three or four 35-tooth planet gears, depending on the model. The planet gears mesh with a 99-tooth fixed ring gear splined to the transmission housing. Power is taken out through the planet carrier splined to the output mast shaft. The output shaft is supported on top by a split-inner-race ball bearing and on the bottom by a roller bearing. The overall reduction ratio of the main power train is 17.44:1.

The 71-tooth bevel gear also drives a 27-tooth accessory gear. The accessory gear runs an oil pump, which supplies lubrication through jets and passageways located in the transmission housing.

NASA 500 HP Helicopter Transmission Test Stand

The OH-58 transmission was tested in the NASA Glenn 500 HP Helicopter Transmission Test Stand (Figure 4). The test stand operates on the closed-loop, or torque-regenerative, principle. Mechanical power circulates through a closed loop of gears and shafts, one of which is the test transmission. The output of the test transmission attaches to the bevel gearbox, whose output shaft passes through a hollow shaft in the closing-end gearbox and connects to the differential gearbox. The output of the differential attaches to the hollow shaft in the closing-end gearbox. The output of the closing-end gearbox connects to the input of the test transmission, thereby closing the loop.

A 149-kW (200 HP) variable speed direct-current (DC) motor powers the test stand and controls the speed. The motor output attaches to the closing-end gearbox. Since power circulates around the loop, the motor replenishes only friction losses.

An 11-kW (15 HP) DC motor provides the torque in the closed loop. The motor drives a magnetic particle clutch. For the OH-58 application, the clutch output does not turn but exerts a torque. This torque transfers through a speed-reducer gearbox and a chain drive to a large sprocket on the differential gearbox. The torque on the sprocket puts a torque in the closed loop by displacing the gear attached to the bevel gearbox output shaft with the gear connected to the input shaft of the closing-end gearbox. This is done within the differential gearbox by a compound planetary system where the planet carrier attaches to the sprocket housing. The magnitude of torque in the loop is adjusted by changing the electric field strength of the magnetic particle cluteh. For applications other than the OH-58 transmission where the speed ratio of the test transmission is slightly different or when slippage occurs (i.e., traction drives), the planet/sprocket/chain assembly rotates to make up for the speed mismatches that occur in the closed loop.

A mast-shaft loading system in the test stand simulates rotor loads imposed on the OH-58 transmission output mast shaft. The OH-58 transmission output mast shaft connects to a loading yoke. Two vertical load cylinders connected to the yoke produce lift loads. A single horizontal load cylinder connected to the yoke produces shear loads. A 13,790-kPa (2000-psig) gas nitrogen system powers the cylinders. Pressure regulators connected to each loading cylinder's nitrogen supply adjust the magnitude of lift and shear forces.

The test transmission input and output shafts have speed sensors, torquemeters, and slip rings. All three load cylinders on the mast yoke are mounted to load cells. The test transmission internal oil pump supplies lubrication. An external oil-water heat exchanger cools the test transmission oil.

The 149-kW (200 HP) motor has a speed sensor and a torquemeter. The magnetic particle clutch has speed sensors and thermocouples on the input and output shafts. A facility oil-pumping and cooling system lubricates the differential gearbox, the closing-end gearbox, and the bevel gearbox. The facility gearboxes have accelerometers, thermocouples, and chip detectors, for health and condition monitoring.

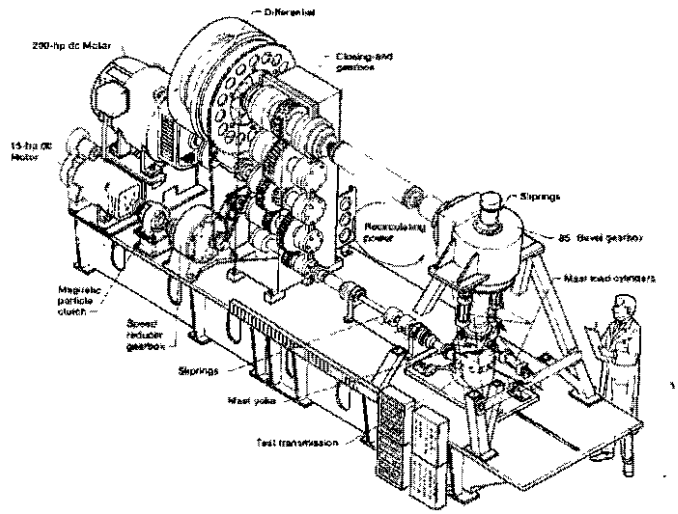


Figure 4. 500 HP Helicopter Transmission Test Stand

Test Gear

The spiral bevel pinion has 19 teeth, a diametral pitch of 6.940 teeth/ineh, a face width of 1.28 inch, a bevel angle of 15 degrees 16 minutes, and a spiral angle of 30 degrees left hand, clockwise. It meshes with a 71 tooth gear to form the first stage of reduction. Triplex ball bearings and one roller bearing support the pinion.

Notch Geometry

A notch was machined into the fillet region of one spiral bevel pinion tooth using electro-discharge machining. The dimensions were approximately 0.1 inch wide, 0.005 inch tall and 0.005 inch deep. After a significant amount of run time at extreme torques, it was determined that the notch was not of sufficient size to facilitate crack initiation. A second notch (Figure 5) was machined in the same area and measured 0.12 inch wide by 0.01 inch tall by 0.08 inch deep. This notch geometry was sufficient to initiate a crack although an extreme over-torque condition was required.

Figure 5. Photograph of notch

Sensors

A suite of sensors were mounted to facilitate the detection of crack initiation and propagation. It consisted of a tachometer, five accelerometers, and a proximity probe.

The once per revolution tachometer signal is generated using an infrared optical sensor that is located on the input shaft to the test gearbox. The sensor detects a change in the reflectivity of an infrared light. The connecting shaft has a piece of highly reflective silver colored tape cemented to the black oxide coated shaft. This provides a reliable signal that has good dynamic performance.

The five accelerometers were located at various locations around the gearbox as shown in Figure 6. Accelerometer 1 is located on the input bevel gear housing immediately above where the input shaft connects to the pinion and is oriented to be most responsive in the vertical direction. Accelerometer 2 is at the same location and is aligned to the rotational axis of the input shaft. Accelerometers 3 and 4 are mounted around the circumference of the ring gear housing and are located 45 and 225 degrees from the input pinion gear. Accelerometer 5 is mounted to one of the attachment bolts near accelerometer 4. Accelerometers 3, 4, and 5 are mounted in the axial-transverse plane and have sensitivities in both directions. The accelerometers are linear to 20 kHz and have a resonance frequency of 90 kHz.

Accelerometer positions 1, 2, and 3 were chosen based on previous experience [20]. In previous testing, accelerometers 1 and 2 had the spiral bevel harmonics as the dominant components. Accelerometer 3 produced the highest levels of vibration where the dominant vibration sources were the spiral bevel mesh and the planetary mesh. Accelerometer locations 4 and 5 also had significant spiral bevel mesh frequency components. The transfer path through the ring gear provides an excellent source for gear mesh vibrations.

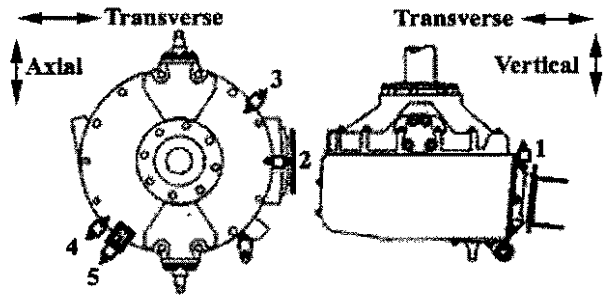


Figure 6. Accelerometer locations

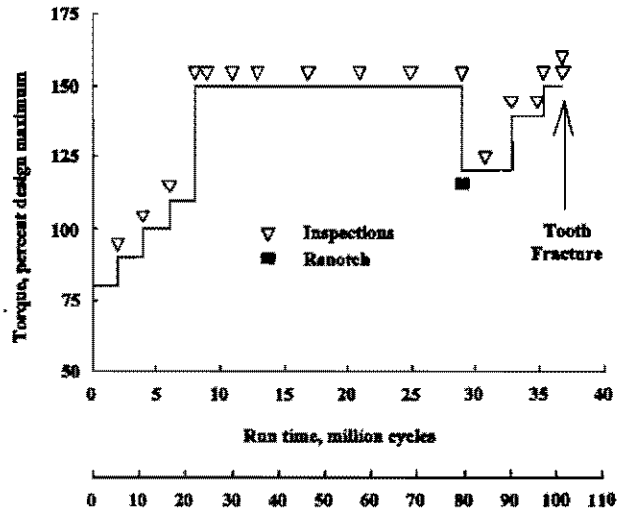
A radio frequency (RF) eddy current proximity probe was mounted inside the transmission on one of the support webs. The probe coil radiates a small RF field near the tip of the probe. If there is no conductive material within this field, there is no power loss in the RF signal. When the top land of the pinion approaches the probe tip, eddy currents are generated on the surface of the pinion, resulting in a power loss in the RF signal. This allows the proximity probe to detect the passing of the top land of the teeth.

Test Procedure

The pinion was run at the design speed of 6060 rpm and at percentages of the maximum design torque according to Figure7. The goal was to initiate a crack in the pinion at the lowest possible torque. Thus, the pinion was initially run at 80% toque. The torque was gradually increased. The inverted triangles represent the periods where an inspection occurred. Inspections were visual using a 60X microscope. At 80 hours run time, the notch was deepened (solid square symbol). This paper deals with the vibration acquired during the last 150% torque cycle between 97 hours runtime and the end of the test.

The vibration, speed and proximity probe signals were passed through a low-pass elliptical anti-aliasing filter with a cutoff frequency of 56 kHz. This data was then acquired using a personal computer equipped with an analog to digital converter capable of digitizing 8 channels at 150 kHz each. A record length of 1.5 seconds was taken every 15 seconds and analyzed. The analysis was performed and displayed near real time.

Figure7 .Loading history of pinion



RESULTS

From the vibration data, there was an indication of potential damage at 101.15 hours run time. A visual inspection with a 60X microscope was performed after 101.41 hours and no crack initiation was detected. The pinion was reinstalled into the gearbox and run for another 12 minutes until the proximity probe indicated a spike corresponding to damage to one of the teeth. Upon disassembly, a tooth was found to be fractured off as shown in Figure 8. The proximity probe had detected the missing top land when the signal caused by the passing of the damaged tooth produced a differing output signal. The fractured tooth was the one with the notch. Close examination shows that the notch surfaces were evenly distributed between the two pieces.

Detailed analysis of the proximity probe data indicates that at 101.4723 hours of run time (approximately 9 minutes before complete fracture), a once per revolution spike was continuously observed. The most probable explanation for this is that at this point massive deflection was taking place. At 101.621 hours into the test, the damaged tooth separated from the remainder of the gear.

It is believed that the pinion was cracked at the 101.15 hour run time inspection. It is also believed that the crack was not visually detected due to two factors. First, adhesive from strain gages installed in the pinion tooth root could have masked the surface to affect the visual inspection. Second, the pinion was inspected under no load and might require tension to open the crack for successful visual inspection.

The RMS of the time synchronous average is shown in

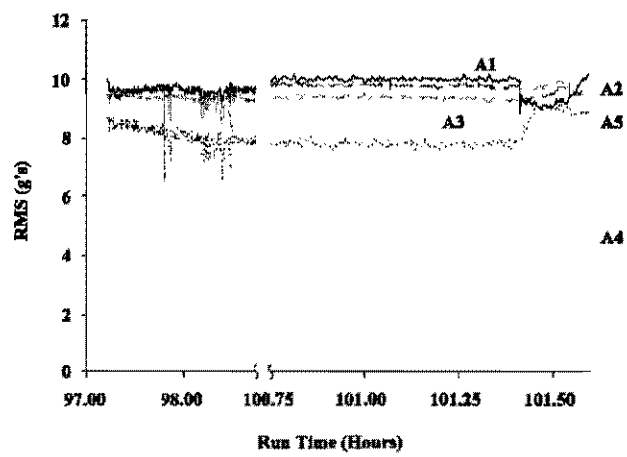


Figure 9. Brief periods of the run encountered torque fluctuations (between 98.3964 and 98.4931 hours of run time). The exact cause of the fluctuations are unknown, but they may have been caused by facility electrical power variations or instrumentation noise. The RMS was very sensitive to the torque fluctuations. Overall, there was no definitive indication of damage from the RMS metric except for at the end of the run where tooth fracture occurred.

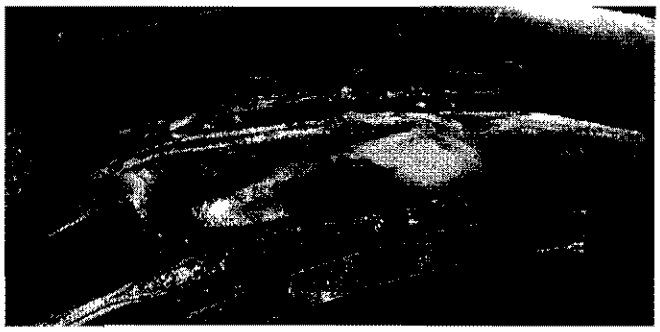


Figure 8. Photograph of fracture

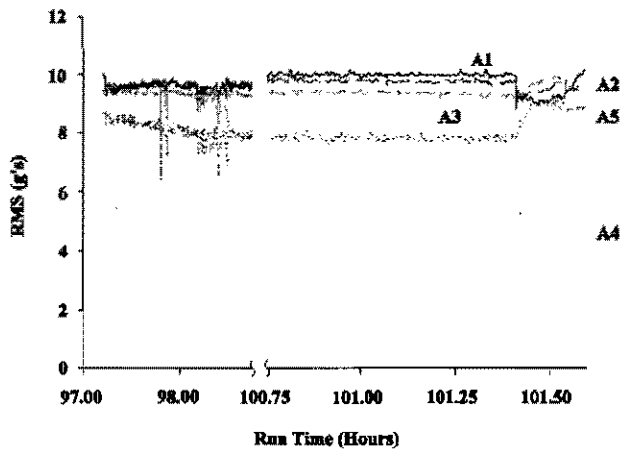


Figure 9. RMS of Synchronous Average

The Crest Factor in Figure 10 shows apparent damage. This indication of damage is after the shutdown and inspection which was prompted by other metrics. This metrie is not as sensitive to the torque spikes as was the RMS.

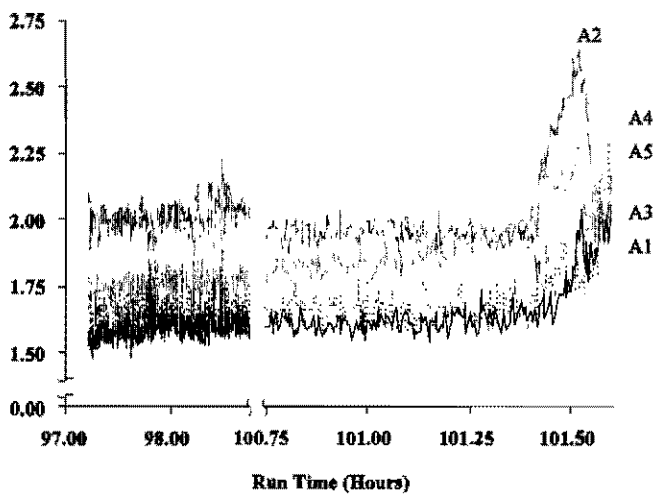


Figure 10. Crest Factor

The responsiveness of the Energy Operator (Figure 11) to the torque spikes casts some uncertainty to its ability to detect the onset of damage. Once the damage has progressed, it becomes a good metric as its value does not decrease to a value indicative of an undamaged state.

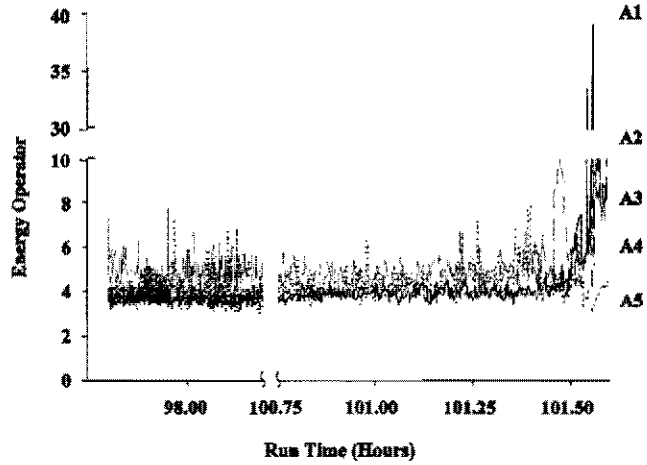


Figure 11. Energy Operator

The Kurtosis (Figure 12) is one of the most responsive of the metrics to the torque fluctuations early in the run. There is some possible indication of damage before the shutdown. This is tempered by the sensitivity to outside influences. Once the damage has progressed, there is an absolute indication of damage.

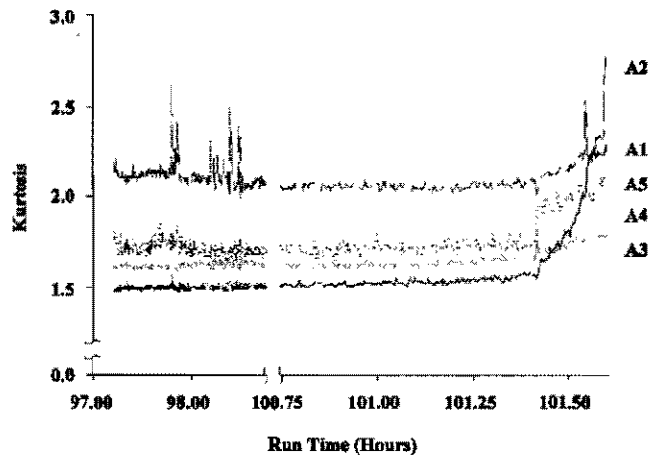


Figure 12. Kurtosis

Figure 13 shows how the responsiveness to the torque spikes makes the Energy Ratio less useful. The uncertainty caused by the torque excursions cast doubt on the metric's suitability until well after other metrics have demonstrated the existence of damage.

The M6A metric (Figure 14) shows less response to the torque fluctuations and exhibits a general upward trend starting almost 30 minutes before shut down and inspection. Once the damage has become a total fracture, the metric shows a definite upset from its normal value.

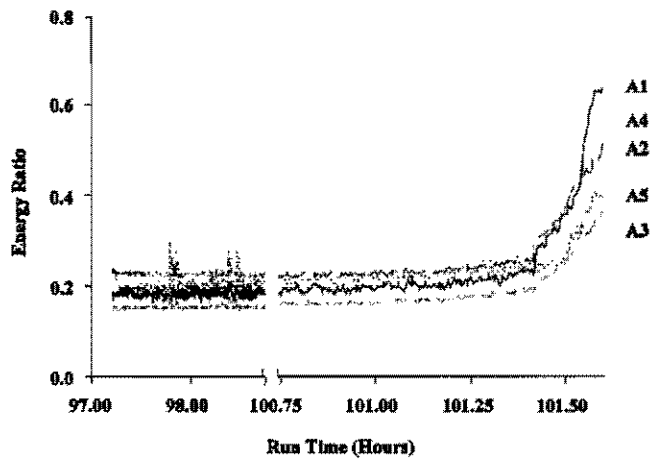


Figure 13. Energy Ratio

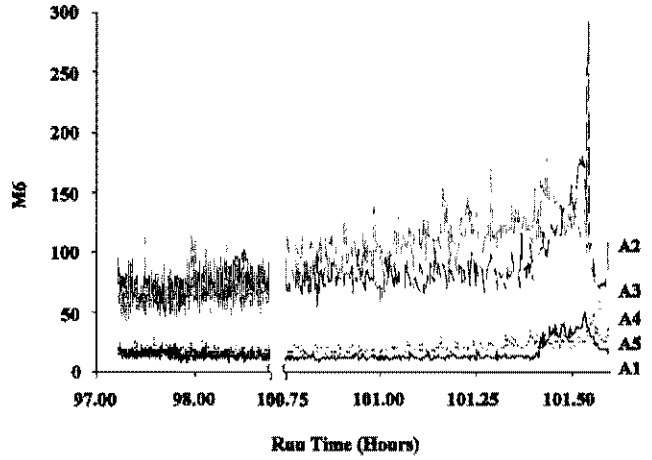


Figure 14. M6A

The torque fluctuations did not have a significant effect on the FM4 metric (Figure 15). There is some gradual upward trending of the metric starting at about 3.5 hours into the run. The real indication of damage occurs after the inspection. This metric also shows one of the potential drawbacks of many of the metrics in its ability to return to a value indicative of a no fault condition.

The metric that best provided indication that damage was occurring or imminent was NA4 (Figure 16). It appears to have indicated damage 15 minutes before the shutdown or 35 minutes before the loss of the tooth. Unfortunately, the torque excursions have a tendency to reduce the confidence in this metric until other metrics confirm the existence of damage.

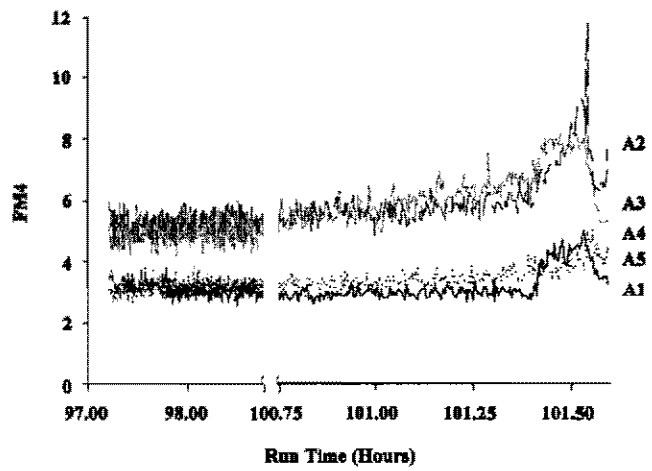


Figure 15. FM4

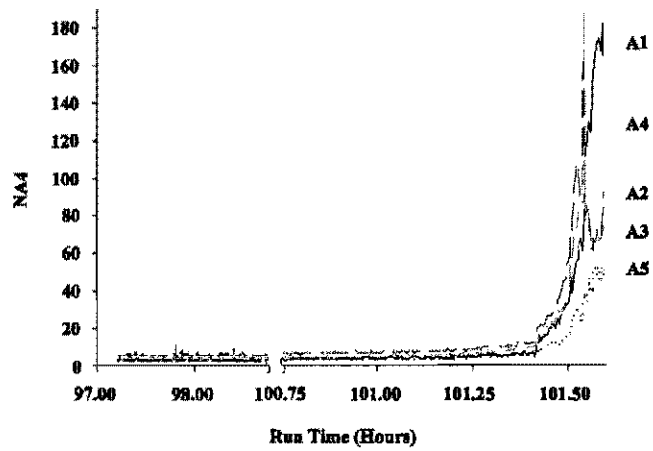


Figure 16. NA4

The NB4 metric (Figure 17) did not exhibit any of the detrimental torque sensitivities of some of the other metrics. It also only started to indicate damage about 4.8 minutes before the inspection shutdown. The stability of the metric during the run does tend to increase its usefulness. The metric also had a definite response to the actual damage.

The NA4* metric (Figure 18) exhibited a time delay relative to the NA4 metric on which it is based. The metric was designed to be more responsive. It also appeared to be more responsive to torque fluctuations than NA4.

Figure 19 shows the M6A* metric. This metric displays the undesirable characteristic of being too responsive to torque variations and also returning to a condition that can be misconstrued as being in a no damage condition.

The FM4* metric (Figure 20) was only slightly responsive to the torque fluctuations. It did appear to reveal that damage was occurring before the shutdown and inspection. After restarting the test, the metric responded in a manner that gave no doubts about whether there was damage or not.

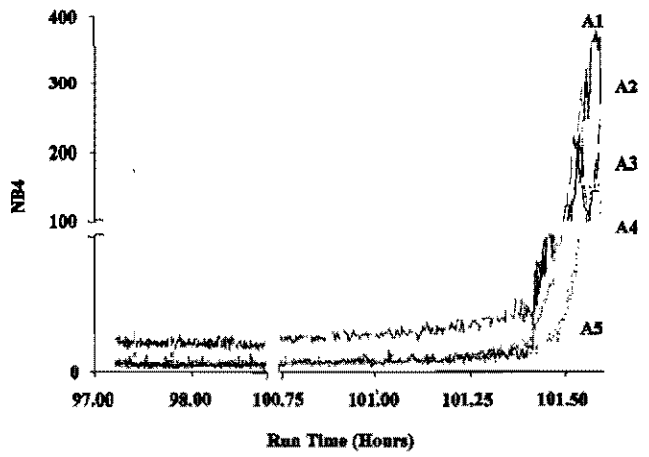


Figure 17. NB4

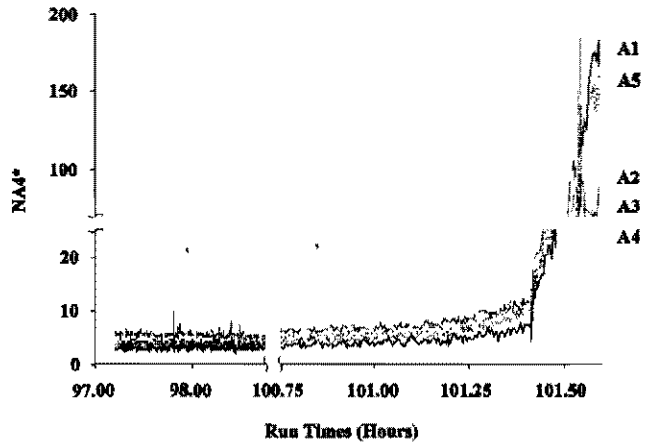


Figure 18. NA4*

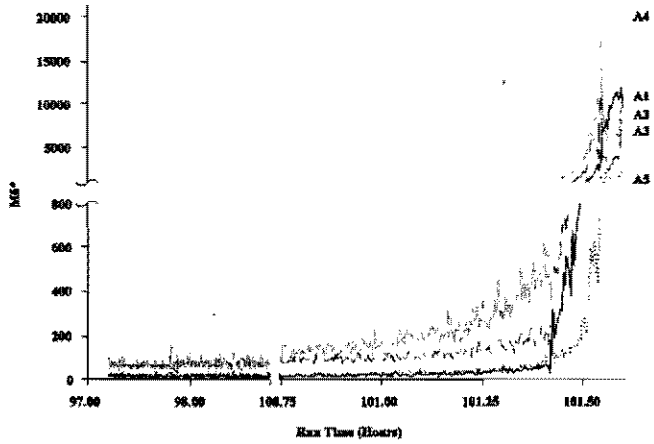


Figure 19. M6A*

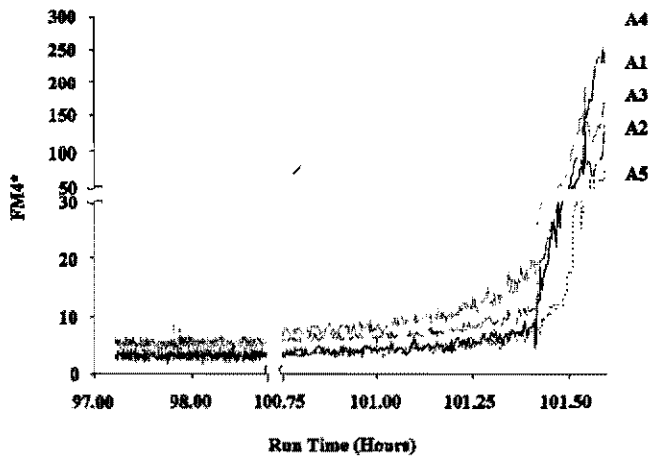


Figure 20. FM4*

The NB4* metric (Figure 21) is much like the NB4 metric in that it is relatively insensitive to torque while still responding to damage nicely.

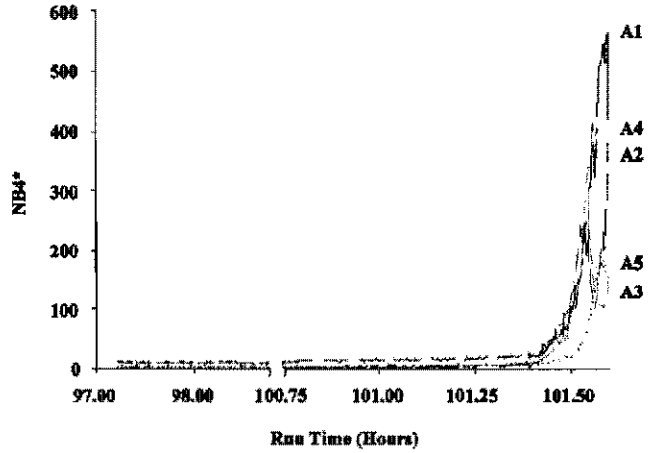


Figure 21. NB4*

If the contributions from each of the accelerometers are examined and compared, it is interesting to note that the A1 and A5 accelerometers provided the least information. Specifically, these accelerometers provided indication of damage after the other accelerometers. The A1 accelerometer is the only one that is most sensitive in the vertical direction. The A5 accelerometer was the only accelerometer that was mounted in a different manner and resulted did not have a major effect on any of the metrics. This may be due to the mounting block that was used.

The A2, A3 and A4 accelerometers produced the majority of the remaining best responses. It is interesting to note that these accelerometers all have their primary sensitivities with a component aligned in the axial direction. A2 is primarily axial in direction and located on the most direct path. This would account for its high degree of performance. A3 and A4 both provided significant indication of damage, but due

to their more distant location from the pinion mesh, their signals were most likely more attenuated.

CONCLUSIONS

This study evaluated vibration-based diagnostic metric to detect gear crack initiation. Seeded fault tests were conducted using a helicopter main rotor transmission. Various over-torque conditions were run to facilitate crack initiation. A visual inspection was performed before each change in torque. Some conclusions are

1. The most effective metrics (in decreasing order) were M6A*, FM4*, and NB4. They were sensitive enough to pick up the damage while not being overly sensitive to the torque fluctuations.
2. Some metrics such as RMS, Energy Ratio, Energy Operator, Kurtosis, and NA4 are very sensitive to torque fluctuations and thus may not be effective.
3. Accelerometer, location and orientation appear to be critical in effectively detecting the damage early.
4. Despite examining the gear with a 60X microscope, it was not possible to detect whether a tooth crack was occurring, even when some of the metrics indicated that one might exist.

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FIGURE F-3 EXAMPLE: INSERTED FAULT VIBRATION MONITORING TESTS FOR A CH-47D AFT SWASHPLATE BEARING

INSERTED FAULT VIBRATION MONITORING TESTS FOR A CH-47D AFT SWASHPLATE BEARING

Jonathan A. Keller
jonathan.a.keller@us.army.mil
U.S. Army RDECOM
Aviation Engineering Directorate
Redstone Arsenal, AL 35898-5000
Paul Grabill
paul.grabill@iac-online.com
Intelligent Automation Corporation
Poway, CA 92064

An incident involving a failure of a CH-47D swashplate bearing has motivated interest within the US Army for vibration monitoring of these bearings. Because of the incident, a series of vibration tests were sponsored by the Army and were conducted using a special, preexisting test rig in which vibration measurements were acquired on swashplate bearings in good and degraded states. This paper discusses the experiment and the resnlts using traditional vibration-based techniques for bearing fault detection. The results demonstrate that corrosion, pitting, and spalling are all detectable through vibration measurements, but cage defects were not.

Notation

a Ball contact angle
BPFI Inner race fault frequency
BPFO Outer race fault frequency
BSF Ball spin frequency
c Ball center
CFF Cage fault frequency
d Rolling element diameter
D Bearing pitch diameter to ball center
f_s Shaft frequency
N_b Number of balls

Introduction

In October of 2002, the swashplate bearing failed in the aft rotor head of a CH-47D during a ground run. Although there were no injuries to the flight crew, this caused a Class-A accident and resulted in the loss of the aircraft. The post-accident investigation determined that failure of the cage of the duplex ball bearing between the rotating and non-rotating swashplates caused the accident. The design of the cage for the swashplate bearing is unique in that it is comprised of two cage segments each of which spans half the circumference of the bearing. Each cage is simply a relatively small-diameter wire which wraps around the balls and holds them in place. For the incident it is suspected that one end of the cage was displaced out from its position between the races and un-caged the balls, eventually resulting in a bearing failure. The failed bearing is shown in Figure 1. This phenomenon has since been termed cage "popping".

The incident resulted in manual inspections of all Army CH-47D and MH-47E swashplates, which required significant manpower. A total of 795 swashplates were inspected and approximately 10 percent failed visual inspection or oil analysis. The majority of deficiencies were due to corrosion; but there were several bearings with raised or broken cages and un-caged ball bearings and a few bearings with pitted/spalled ball bearings and races. Since the incident, an additional maintenance check has been required on the Chinook to maintain airworthiness. The check involves visually inspecting both swashplate bearings and is required every 50 flight-hours. It is likely that vibration monitoring equipment, if it had been installed on the swashplate, would have detected such a failure in advance of the accident. Clearly, vibration monitoring of swashplate bearings would be beneficial to the Chinook fleet in terms of both safety and reduction in maintenance.

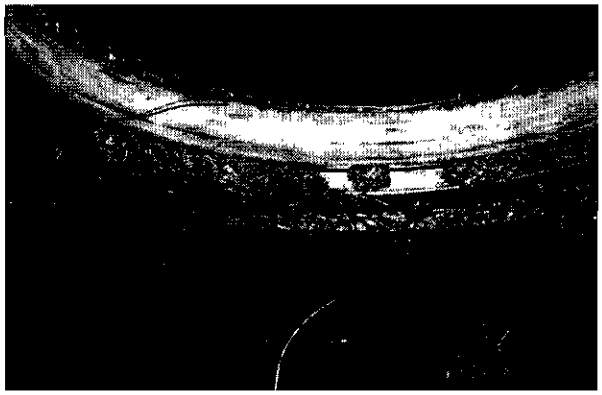


Figure 1. Failed Swashplate Bearing

The Vibration Management Enhancement Program (VMEP) program is currently in use by the US Army National Guard, US Army Special Operations, and US Army TMDE demonstration program. There are over 80 aircraft equipped with the VMEP system on the UH60A/L, MH-60L/K, AH-64A/D and CH-47D. VMEP is a permanently installed on-board system that performs rotor smoothing functions and monitors vibration levels of rotating components on the aircraft using diagnostic algorithms for typical gearbox and bearing faults [1-4]. In October of 2003, a VMEP system was installed on CH-47D tail number 81-23381, called "Bearcat-1", at the US Army Aviation Technical Test Directorate (ATTC) at Ft. Rucker, Alabama. Since its installation, the VMEP system has been used to gather baseline vibration diagnostic measurements on Bearcat-1 including monitoring of both the forward and aft swashplate bearings. However, Bearcat-1 is a well-maintained test aircraft so no swashplate bearing anomalies have occurred to date. Thus, the swashplate bearing data gathered on Bearcat-1 so far has only useful in establishing "normal" vibration levels. This establishment of a normal vibration level is a necessary step, but it is unknown exactly how vibration levels will change due to a faulty bearing.

Presented at the American Helicopter Society 61stAnnual Forum, Grapevine, TX, June 1 – 3, 2005. Copyright ©2005 by the American Helicopter Society International, Inc. All rights reserved.

Related Research

Sikorsky has developed a Bearing Monitoring System (BMS) for H-53E and S-80M aircraft, consisting of accelerometers and temperature sensors mounted to the non-rotating swashplate [5]. Periodic inspections of H-53E bearings have been eliminated and bearings are being removed based solely on the BMS indication. The BMS has been highly successful, with zero missed alarms and just one false indication due to a loose sensor. Three swashplate bearings have been removed based on the BMS and each removal was found to be justifiable based on wear.

Currently, the Aviation Vibration Analyzer (AVA) is the fielded US Army system used to perform periodic vibration measurements on dynamic components and rotor track and balance on all Army aircraft. Although the AVA performs these periodic tasks well, it has several shortcomings with respeet to bearing monitoring. Firstly, it is not permanently installed on the aircraft so it is not capable continuously monitoring components of interest. Also, the signal processing methods available in the AVA are limited to simple spectral analysis and synchronous averaging. The size of the accelerometer included in the AVA kits fielded to Army units, the Wilcoxon 991D, is over 2 inches high (including connector) and ¾ inches in diameter [6]. So the 991D is not easily placed near the swashplate bearing, which reduces the measurability bearing faults traditionally detected at high frequencies. Thus, the AVA is probably not a satisfactory solution for swashplate bearing monitoring which often requires reliable high frequency vibration data on a continuously monitored basis.

Preliminary AVA and VMEP vibration measurements were discussed for a temporary "proof of concept" CH-47D installation [7]. Like Bearcat-1, the test aircraft was well-maintained and had passed the manual swashplate bearing inspection so the measurements acquired have only been useful as additional baseline data. For the Apache

and Blackhawk aircraft, VMEP has had success detecting degraded UH-60A oil cooler fan bearing, AH-64A main rotor swashplate bearing, AH-64D tail rotor drive shaft aft hanger bearing, and AH-64A nose gearbox [8].

Objectives

The U.S. Army Aviation Engineering Directorate, Aeromechanics Division conducted an investigation to acquire vibration measurements for H-47 swashplate bearings in both good and degraded conditions on a special test rig. Vibration measurements were acquired at several simulated flight conditions and at a steady flight condition over a 24-hour run. The results from these tests are the subject of this paper. The primary objective of the test was to determine whether faulted bearings could be detected using the VMEP system.

Analytic Approach

Swashplate Bearing Test Rig

All tests were conducted on a specialized swashplate test rig located at Boeing Helicopters Philadelphia (BHP), originally intended to qualify swashplate bearings. The rig, shown in Figure 2, is a back-to-back design in which the upper swashplate assembly is the drive and the lower swashplate assembly houses the test bearing. The swashplates are driven by an electric motor, which for the test was made to closely approximate H-47 main rotor speed. The rotor speed is measured with a standard AVA magnetic-pickup main rotor tachometer, identical to the one use in the field. Approximate ground-hover-forward flight loads are applied to the swashplate using three hydraulic actuators below the test rig. The actuators are capable of producing enough force to simulate loads up to steady, level flight at 140 knots airspeed. The loads on two of the three pitch links are measured by strain gages channeled through a slip ring and are used to set the hydraulic actuators. Approximate bearing temperature was also measured with two sensors. A proximity probe was also used to shut down the test rig in the event of excessive test stand vibration.

Figure 2. Swashplate Bearing Test Rig

The non-rotating swashplate of the test housing was instrumented with two VMEP Dytran 3077A accelerometers and a standard AVA-type magnetic pickup tachometer. The accelerometers were bolted to a steel mounting pad, which was in turn epoxied to the underside of the swashplate. The accelerometers were located as close as possible to the swashplate bearing and main rotor shaft just above the stationary swashplate to longitudinal cyclic trim (LCT) actuator connection point. One accelerometer was in the vertical (parallel to the shaft) direction as shown in Figure 3 and the other accelerometer was in the radial direction. The tachometer and both accelerometers were then wired to a VMEP Vibration Management Unit (VMU) inside a control room for data acquisition.

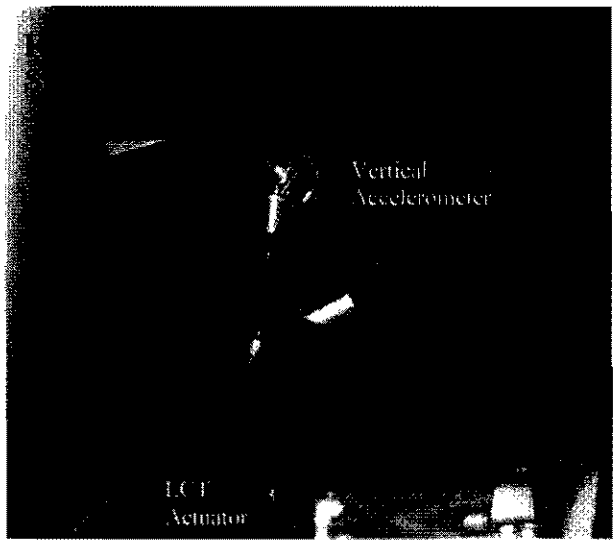
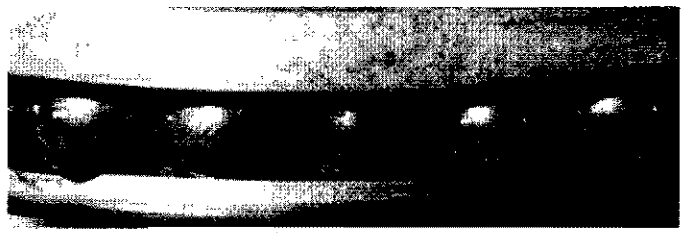


Figure 3. Dytran 3077A Vertical Accelerometer

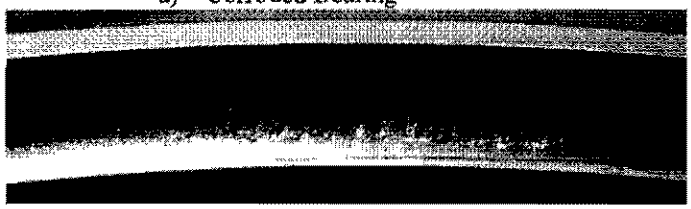
Selection of Swashplate Bearings

In many other tests of this nature, bearing faults must be artificially introduced, or "seeded", by injecting debris into the bearing or by running the bearing at high torque, preload and/or speed conditions. However, that was not the case for this test. As previously stated, after the incident manual swashplate bearing inspections were required of all Army H-47 swashplate bearings. The bearings that failed the visual and oil analysis were sent to the Corpus Christi Army Depot (CCAD) where they were degreased, inspected and cataloged by damage type and severity by the Analytical Investigation Branch (AIB). This set of defective bearings provided a natural source for these "inserted" fault tests.

The predominant failure mode of the swashplate bearing was corrosion of the balls and races. A few specimens were pitted and spalled and a few had raised cage ends. Three faulted bearings were chosen – one corroded, one spalled and one with a raised cage end – and re-greased and sent to Boeing. While this process was occurring, a fourth bearing specimen was identified from a Quality Deficiency Report (QDR). This bearing had two cage segments that were not only raised, but overlapping each other. Pictures of the four faulted bearings are shown in Figure 4. In addition to the faulted bearings, two bearings in good condition were selected to establish a normal baseline. One bearing was a relatively low-time specimen, with only 132 accumulated flight hours. The other bearing had 1199 accumulated flight hours, which is just under the Army's to-be-overhauled (TBO) limit of 1200 flight hours for the swashplate assembly.

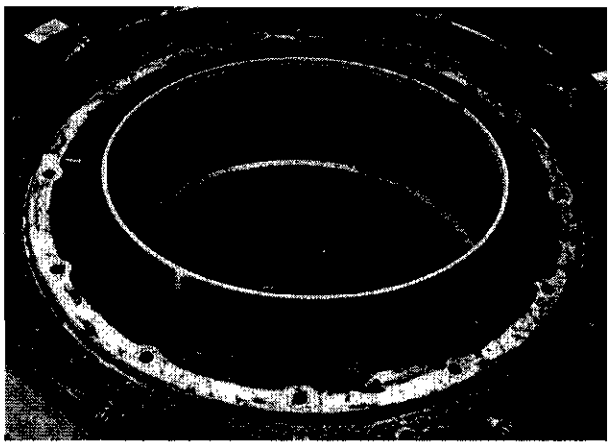


a) Corroded Bearing



b) Spalled Bearing

c) Bearing with Popped (i.e. Raised) Cage



d) Bearing with Overlapped Cage (in Swashplate Assy)

Figure 4. Faulted Bearings

Data Acquisition

Using the flexible iMDS Setup Design Tool, the VMU was programmed to acquire and process data in two different modes. In the "Flight" mode the data was acquired through a manual button push. For each flight acquisition, the VMU was programmed to save the tachometer pulses (for calculation of rotor speed), 275000-point raw accelerometer data records, 8192-point 24 kHz spectra calculated with 10 spectral averages, and selected condition indicators. In the "Monitor" mode a data set was acquired automatically once every two minutes. For each monitor acquisition, the VMU calculated the same parameters as the "Flight" mode, but only archived the selected condition indicators. In all cases, the sampling rate was fixed at 48 kHz.

The test for each bearing specimen was split into two phases. In the first phase, single "Flight" mode vibration measurements, hereafter referred to as "snapshot" measurements, were acquired at 5 different load conditions applied to the swashplate assembly through the test rig. Each condition was representative of the loads experienced by a normal CH-47D swashplate in actual flight conditions. The load conditions were flat-pitch-ground 100% rotor speed (FPG100), hover, and 80 knots, 120 knots and 140 knots steady level forward flight speeds. The first phase was intended to establish a baseline and examine changes in vibration with loading. In the second phase, the bearing was run for a total of 24 hours (three 8-hour segments) at the FPG100 load condition. "Flight" mode vibration measurements were acquired every 2 hours during this phase, while "Monitor" mode measurements were acquired automatically once every two minutes. The second phase was intended to examine any changes in the vibration with time, i.e. fault progression.

Data Analysis

Vibration analysis can be used to detect bearing faults at a relatively early stage in the fault progression. Rolling element bearings generate characteristic vibration signatures in several ways. A typical ball bearing, shown in Figure 5, consists of an inner and outer race separated by the rolling elements, which are usually held in a cage. If a roller or a ball has a defect such as a pit, each revolution will result in a brief impact that is transmitted to the bearing housing. The fundamental frequency of these impacts is called the ball spin frequency (BSF). If the bearing inner race has a defect, then each ball will produce a shock as it passes giving rise to a fundamental vibration frequency called the ball-pass frequency, inner race (BPFI). Likewise a fault on the bearing outer race will produce a frequency at the ball-pass frequency, outer race (BPFO). The last frequency of interest is the frequency at which the bearing cage itself rotates. This frequency is called the cage fault frequency (CFF) or fundamental train frequency (FTF).

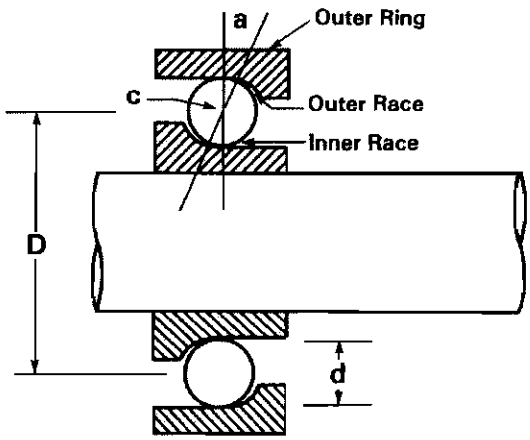


Figure 5. Basic Rolling Element Bearing Geometry

For the H-47 swashplate bearing, the inner race is fixed to the non-rotating swashplate while the outer race rotates with rotating swashplate and main rotor. This geometry has no effect on the ball spin, inner race and outer race fault frequencies but does change the calculations of the cage fault frequency. The defect frequencies can be easily calculated from the bearing geometry as follows:

$$\mathrm{BPFI} = \frac{N_b}{2} f_s \left[1 + \frac{d}{D} \cos(a) \right] \qquad (1)$$

$$\mathrm{BPFO} = \frac{N_b}{2} f_s \left[1 - \frac{d}{D} \cos(a) \right] \qquad (2)$$

$$\mathrm{BSF} = \frac{D}{2d} f_s \left[1 - \left(\frac{d}{D} \right)^2 \cos^2(a) \right] \qquad (3)$$

$$\mathrm{CFF} = \frac{1}{2} f_s \left[1 + \left(\frac{d}{D} \right) \cos(a) \right] \qquad (4)$$

The Chinook swashplate bearing can be assembled with 102, 103, or 104 balls. All bearings used for this test were assembled with 102 balls. Using Eqns. (1) through(4), the fault frequencies were calculated and are listed in Table 2.

Vibration "pulses" are created by the faults at the frequencies in Eqns. (1) through(4), but these pulses tend to be short in duration. Different fault detection techniques are employed to enhance the known vibration characteristics from the fault and their resultant effect on the measured vibration. The simplest technique is the narrow band spectrum analysis, which can be used to identify the fundamental fault frequencies in the measured vibration spectrum. The cepstrum analysis evaluates the harmonic content in the measured vibration spectrum created by the short duration of the fault impulses. The shock pulse method quantifies the high frequency, short impulse nature of the fault by measuring the vibration of the swashplate or accelerometer-bracket structure at a resonance frequency. Lastly, the amplitude demodulation technique identifies the fault frequencies that are present in the high frequency spectrum, often the same frequency region as the shock pulse method.

Table 2. Chinook Swashplate Bearing Properties

Rotor Speed (f_s)	3.75 Hz (100%NR)			
Ball Diameter (d)	0.4375 in			
Pitch Diameter (D)	15.75 in			
Contact Angle (a)	30°			
No. of Balls (N_b)	102			
	Fault Frequency (Hz)			
	1x	2x	3x	4x
CFF	1.9	3.8	5.8	7.7
BSF	68	135	202	267
BPFO	187	373	560	747
BPFI	196	392	588	783

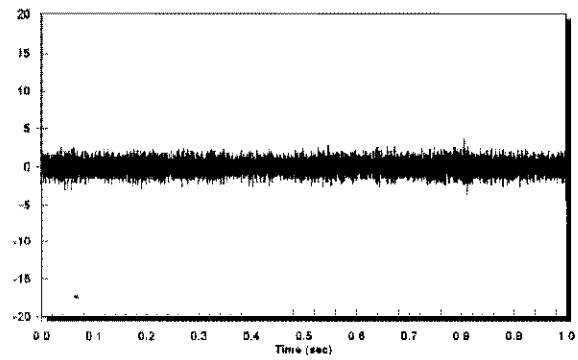
Rolling element bearings typically pass through four stages of degradation [9], characterized by the size of the fault and the frequencies associated with it. The frequency ranges listed here are approximate ranges for Army rotorcraft, meant only for reference purposes. The first stage of degradation is characterized by microscopic defects that result in vibration and acoustic noise, or vibration or acoustic noise at frequencies on the order of 100 kHz. Accelerometers are not well suited to detect energy in this frequency range. As the defects grow larger and start to become visible to the naked eye, the bearing is in the second stage of degradation. The vibration in this stage often manifests itself in the resonances of the structure in the range of 2 to 20 kHz, which is within the range of most accelerometers utilized by helicopter vibration monitoring systems. As the faults increase in size and severity, the vibration can be detected at the fundamental fault frequencies of the bearing, typically in the range of 100 Hz to 2 kHz. In the fourth and final stage of degradation, the faults are often severe enough to allow the shaft to flail, resulting in vibration at the shaft once-per-revolution frequency on the order of 4 to 400 Hz, and eventual breakage or seizure of the shaft. Each of the analysis techniques discussed in the previous paragraph have advantages and disadvantages with respect to the progression of bearing faults, which is described in greater detail in Ref. 7.

Results

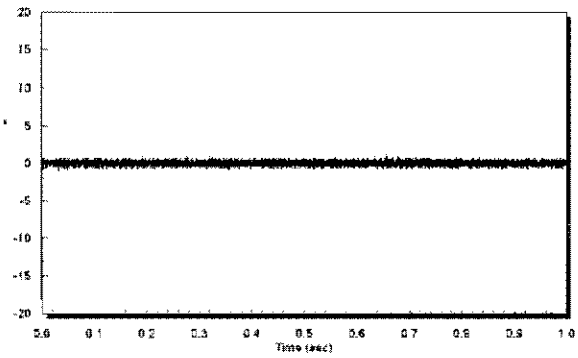
Performing vibration diagnostics for bearings is often an iterative process. The frequency range of the structural resonances described in the previous section is often not known *a priori*. The best method to construct the algorithms yielding the most useful CIs is to compare the differences in vibration for bearings with known faults to bearings in good condition. Hence, seeded or inserted fault testing is highly useful for vibration diagnostics. In the following section, the results of this process are discussed.

Snapshot Vibration Measurements

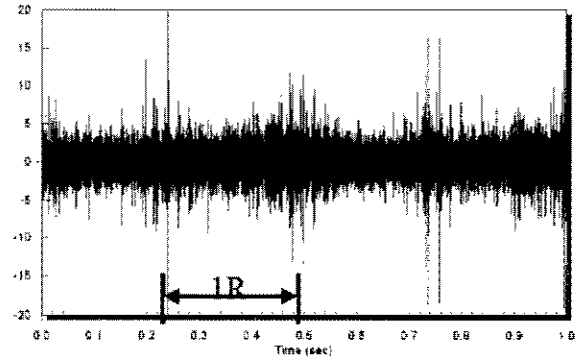
Although not necessarily a good indicator of a fault, a cursory examination of the raw time domain vibration signal is often instructive. The raw vertical accelerometer signal, measured in g's, for each bearing is shown in Figure 6. In each case, the load condition was FPG100. Only the first second of the record, which is about 4 revolutions of the main rotor shaft, is shown. Several conclusions can be made. First, there is little difference between the low-time and TBO bearings. The TBO bearing (with 1199 hours) has lower magnitude than the low time bearing (with 132 hours). Secondly, it is blatantly obvious that the corroded and spalled bearings are vastly different than the bearings in good condition. The corroded bearing has peaks greater than ±20g that appear once-per main rotor shaft revolution. The spalled bearing, on the other hand, appears to have smaller peaks but greater overall vibration. It should also be noted that the spalled bearing was run for 50 hours at the FPG100 condition before the actual test measurements were made. This was done in an attempt to re-generate metal particles which would have existed in the grease had the bearing not been degreased and inspected at CCAD. The last point of interest is that the cage pop and overlap specimens do not appear any different from the specimens in good condition. Clearly, one can assume that detecting a bearing with only a cage defect will be a challenge.



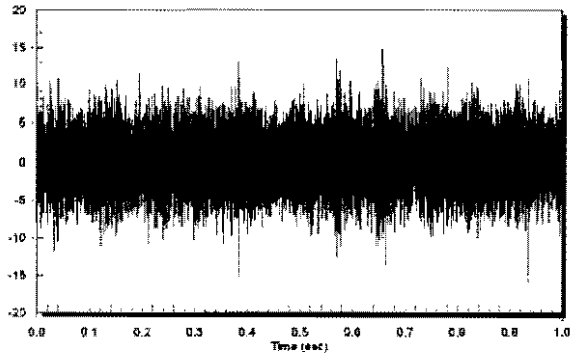
a) Low-time Bearing



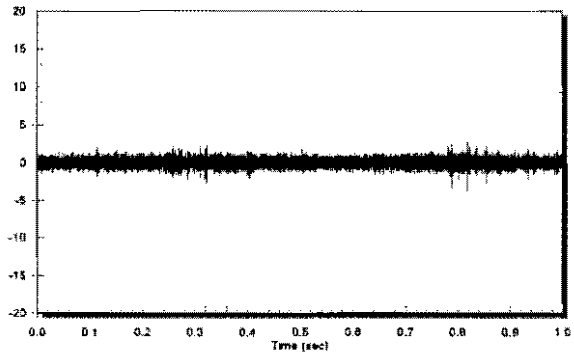
b) TBO Bearing



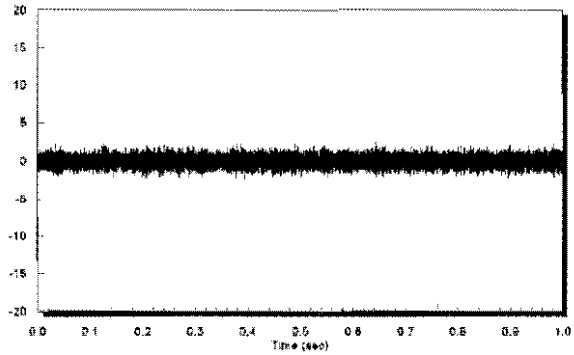
c) Corroded Bearing



d) Spalled Bearing



e) Cage Pop Bearing

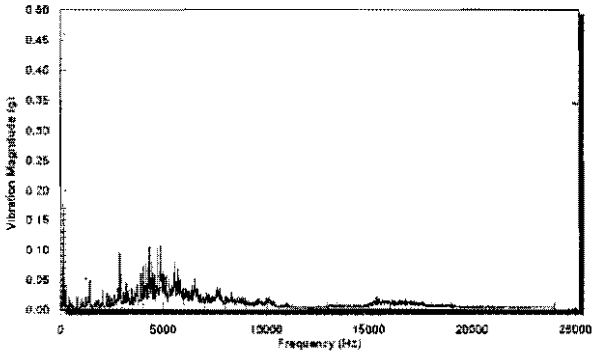


f) Cage Overlap Bearing

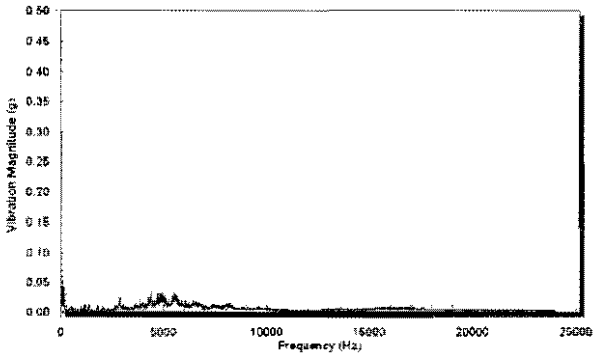
Figure 6. Raw Vibration Signals

significant amplitude under 1 kHz, which is in the range of the fundamental fault frequencies. In contrast, the spectra for the cage pop and cage overlap specimens appear to be very similar to the specimens in good condition.

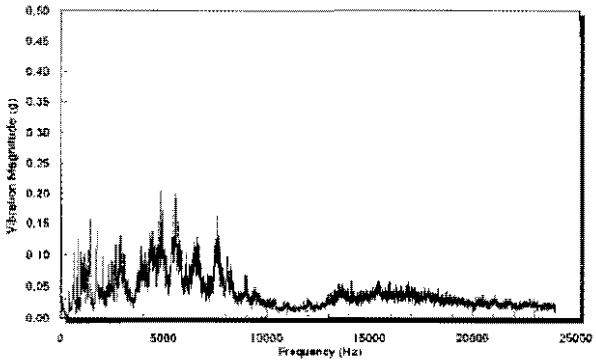
The next logical step is to examine the vibration spectra for each bearing as shown in Figure 7. Aside from having slightly larger vibration amplitude around 5 kHz, there is very little difference between the low-time and TBO bearings. As expected, there is a significant difference for the corroded and spalled bearings. Each of them has significant peaks in the neighborhood of 5 kHz and broadband energy in the 12 to 24 kHz range, with a peak at about 16 kHz. It is unknown whether these regions contain natural frequencies of the swashplate assembly, test stand, or accelerometer/bracket combination. In addition, the spalled bearing appears to have



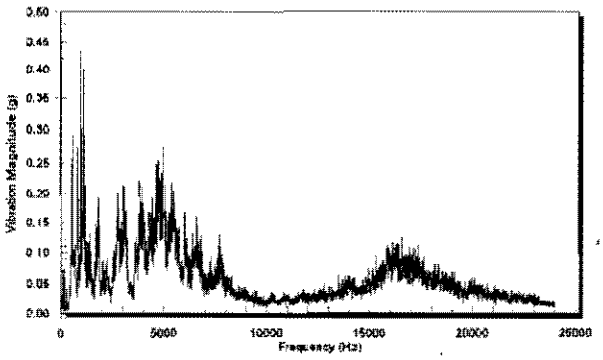
a) Low-time Bearing



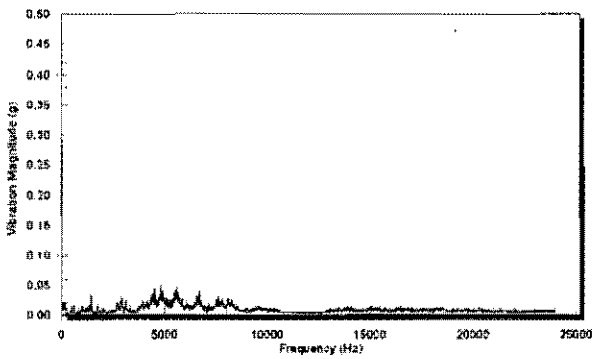
b) TBO Bearing



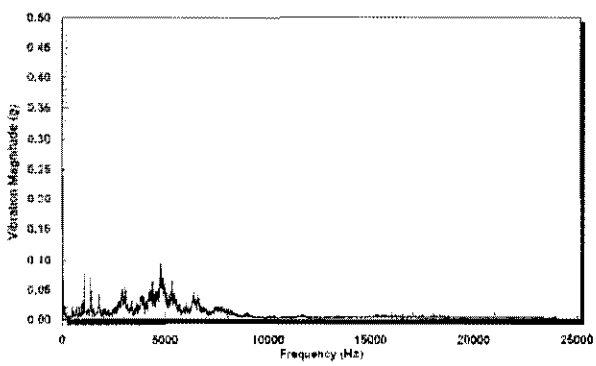
c) Corroded Bearing



d) Spalled Bearing



e) Cage Pop Bearing



f) Cage Overlap Bearing

Figure 7. Vibration Spectra

From the examination of the spectra, three frequency regions are of interest: a low (< 1 kHz) region containing the fundamental fault frequencies, a mid (3 to 6 kHz) region and a high (12 to 24 kHz) region. The results, although not shown here for brevity, are very similar for the radial accelerometer.

Comparison of Condition Indicators

The vibration signatures discussed in the previous section lead naturally to the development of several sets of condition indicators. For the low frequency (< 1 kHz) region, CIs were developed from the shock-pulse and narrow band spectral analyses. For the shock pulse method, the largest amplitude peak in g's is selected and the signal power in g's is calculated from 0 to 1 kHz, which contains up to the fifth harmonic of the bearing fault frequencies. For the mid frequency region, CIs were developed from the shock-pulse and amplitude demodulation methods in a region from 3 to 6 kHz. For the amplitude demodulation method, both the largest peak in g's and the signal power from 0 to 1 kHz was then selected. For the high frequency region, the same shock pulse and amplitude demodulation CIs were calculated in ranges from 14 kHz to 18 kHz and 10 kHz to 24 kHz. In addition, the signal RMS value was also calculated from the raw vibration data for comparison purposes.

The following paragraphs present some of the most effective individual CI values for each bearing specimen. The general trend is that most of the methods are capable of detecting the corroded and spalled bearings, but not either of the bearings with a cage fault. In addition, in most cases the low-time bearing actually has higher CI values than the TBO bearing.

Figure 8 shows a summary of the results for the signal RMS. In this test cell application, a metric as simple as this is capable of identifying corroded and spalled bearings; though in on-aircraft applications it may not be as useful due to other vibration sources such as the rotors and transmissions.

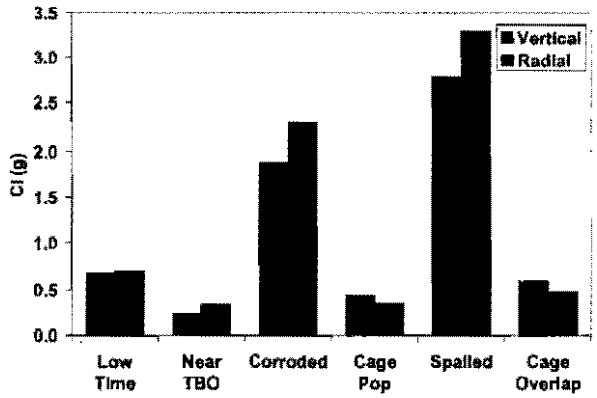
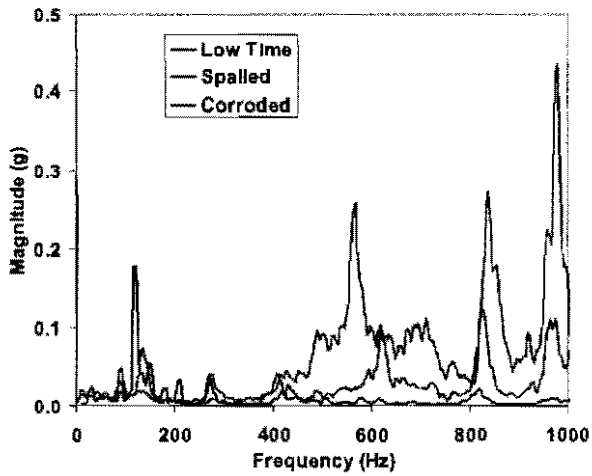
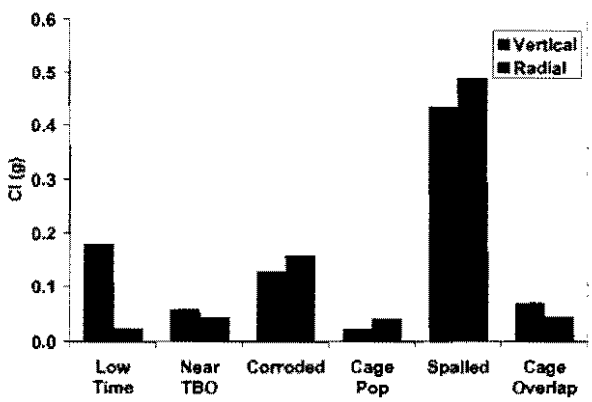


Figure 8. Signal RMS Summary

Figure 9a shows the low frequency vibration spectra for the vertical accelerometer that covers the first few harmonics of the bearing fault frequencies. Figure 9b shows the largest peak in that region in g's for both the radial and vertical accelerometers. Simple spectral analysis does a relatively good job of detecting the spalled bearing, but it did not definitively identify the corroded bearing. There are several large peaks at the harmonics of the inner and outer race frequencies. This is likely due to the fact that the spalling was relatively localized to a few regions on the bearing races, but the corrosion was widely distributed throughout the bearing. Also, there is a large peak at the ball spin frequency in the low-time bearing data for the vertical accelerometer. The fact that the spectral analysis data shows mixed results is not a surprising fact.



a) Low Frequency Vibration Spectra (Vertical)



b) Spectral Analysis

Figure 9. Spectral Analysis Summary

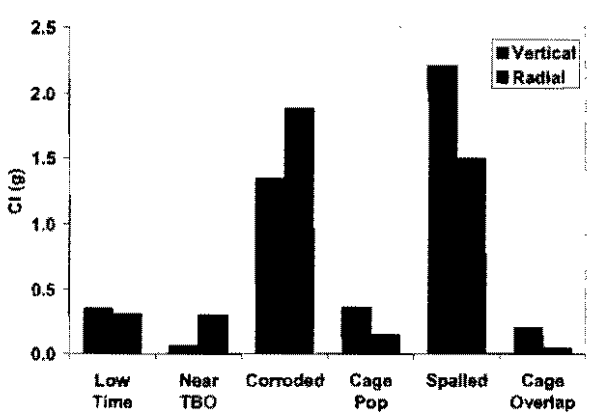
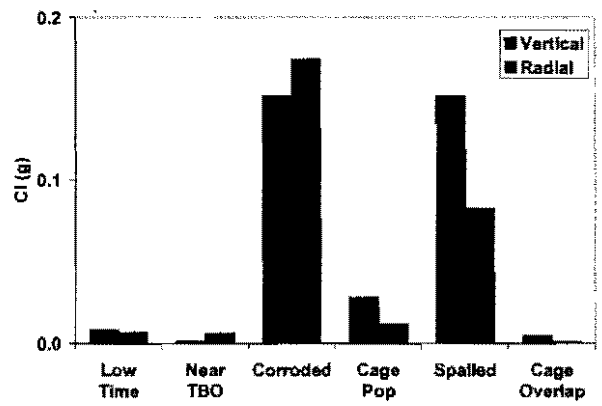


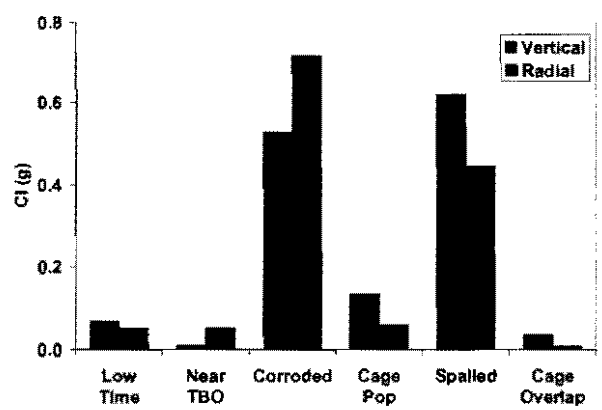
Figure 10. Shock Pulse Analysis Summary

Figure 10 shows the results of the shock pulse method for a frequency band from 10 to 24 kHz. Both the spalled and corroded bearings are easily distinguishable using both the vertical and radial accelerometers. However, the cage fault bearings are not distinguishable.

Figure 11 shows two results from the amplitude demodulation method. In this method, the signal was band passed from 10 to 24 kHz, full-wave rectified, and then the spectrum was calculated. In Figure 11a, the largest peak from the spectrum in the range of the fundamental bearing fault frequencies is presented. In Figure 11b, the energy in the



a) Largest Peak



b) Signal Energy

Figure 11. Spectral Analysis Summary

same frequency band is presented. Once again, in both methods the spalled and corroded bearings are easily distinguishable but the cage defect bearings are not.

Conclusions

A Class-A incident involving seizure of a CH-47D aft swashplate bearing has motivated interest within the US Army for vibration monitoring of these bearings. Because of the incident, a series of vibration tests were sponsored by the US Army Cargo Program office and were conducted using a special, preexisting test rig. In the test vibration measurements were acquired on swashplate bearings in good and degraded states. Traditional vibration-based techniques for bearing fault detection including spectral, cepstral, shock-pulse and amplitude demodulation analyses were applied to the data to determine whether faulted bearings could be detected. The results demonstrate that corrosion, pitting, and spalling are all easily detectable through vibration measurements, but bearings with only cage defects were not detectable.

Acknowledgments

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Appendix G:
Acronyms

G.1 ACRONYMS

ADC	Analog-to-Digital Converter
ADS	Aeronautical Design Standard
AED	Aviation Engineering Directorate
AG	Advisory Generation
ATTC	Aviation Technical Test Center
AWR	Airworthiness Release
BIT	Build-In Test
BITE	Build-In Test Equipment
CBM	Condition Based Maintenance
CBM+	Condition Based Maintenance Plus
CI	Condition Indicator
CLOE	Common Logistics Operating Environment
CNS/ATM	Communication, Navigation, Surveillance, and Air Traffic Management
COTS	Commercial Off-the-Shelf
CRT	Component Retirement Time
DA	Data Acquisition
DAD	Detection Algorithm Development
DBA	Database Administration
DBMS	Database Management System
DM	Data Manipulation
DMWR	Depot Maintenance Work Requirement
DoD	Department of Defense
DSC	Digital Source Collector
EMI	Electromagnetic Interference
FAA	Federal Aviation Administration
FCC	Failure Condition Characterization
FFT	Fast Fourier Transform
FHA	Functional Hazard Assessment
FMECA	Failure Modes Effects Criticality Analysis
HA	Health Assessment
HI	Health Indicator
HUMS	Health and Usage Monitoring System

IETM	Interactive Electronic Technical Manual
ISO	International Standards Organization
LG	Landing Gear
LIS	Logistics Information Systems
MAC	Message Authentication Code
MIC0	Message Integrity Code
MIMOSA	Machinery Information Management Open Systems Architecture
OEM	Original Equipment Manufacturer
PA	Prognostics Assessment
PCA	Principle Component Analysis
PEO	Program Executive Office (r)
PSA	Preliminary Safety Assessment
RCM	Reliability Centered Maintenance
RFP	Request For Proposal
RIMFIRE	Reliability Improvement through Failure Identification and Reporting
RTC	Redstone Test Center
RTCA	Radio Technical Commission for Aeronautics
RUL	Remaining Useful Life
SARSS	Standard Army Retail Supply System
SCORECARD	Structural Component Overhaul Repair Evaluation Category and Remediation Database
SD	State Detection
SEP	Systems Engineering Plan
SGU	Symbol Generator Unit
S-N	Stress-to-Cycles
STA	Synchronous Time Average
STAMIS	STandard Army Management Information System
TAMMS-A	The Army Maintenance Management System-Aviation
TBO	Time Between Overhauls
TDA	Tear-Down Analysis
TMDE	Test, Measurement and Diagnostic Equipment
UAS	Unmanned Aerial System
ULLS	Unit Level Logistics System
WUC	Work Unit Code
WoW	Weight-on-Wheels